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Development of the Ranger Block III Spacecraft Propulsion System

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ABSTRACT

This Report describes the design, development, and flight operation of the midcourse propulsion system utilized on the *Ranger* Block III spacecraft. This monopropellant-hydrazine-fueled system delivers 50 lb_f of thrust and is capable of imparting a variable impulse to the spacecraft in order to accomplish required trajectory corrections resulting from launch-vehicle dispersion errors. These corrections are required so that the time-of-arrival and impact point can be optimized in order to obtain high-resolution television pictures of the lunar surface. The propellant feed system utilizes a conventional high-pressure gas-storage system, a unique pressure regulator, and a propellant tank containing a bladder for positive expulsion. Engine ignition is accomplished through the injection of a small quantity of a hypergolic oxidizer, nitrogen tetroxide. All valving functions are accomplished by explosively actuated valves. A detailed description of component and system testing is included, along with the in-flight performance of the system during the *Ranger VI, VII, VIII,* and *IX* missions.

I. INTRODUCTION

The successful accomplishment of the exploration of the Moon requires that the trajectory of a spacecraft be precisely controlled such that the point of impact, or nearest approach, is optimized for the desired scientific mission. For lunar missions, a launch vehicle velocity error of 1 m/sec (0.01% of total injection velocity) in the most sensitive direction would result in about a 200-km miss of a lunar target, *even if no velocity vector pointing errors had occurred.* For the *Ranger* project, a detailed analysis of *Atlas–Agena* launch vehicle guidance errors indicated that an on-board propulsion capability to accomplish a postinjection trajectory correction was a necessity if the mission requirements were to be met (Ref. 1).

The requirements of such a propulsion system could be summarized as follows: (1) it must have the capability of delivering a variable total impulse to the spacecraft, imparting a velocity increment of 0.5–40 m/sec; (2) it must be of relatively low thrust, such that the acceleration to the spacecraft would be approximately 0.1 *g*; and (3) it must have the potential of extremely reliable operation. Both liquid- and solid-fueled rocket systems were considered for the application; the final choice was a liquid monopropellant-hydrazine system which develops 50 lb of thrust and utilizes jet vanes for thrust vector control.

Monopropellant-hydrazine had been under development at JPL for several years (Refs. 2–5), and its state of development was such that it could be readily employed in such an application. The conception, design, and original development of the midcourse propulsion system for *Ranger* is described in Ref. 6. This system was employed on the Block II series of spacecraft (*Ranger III, IV,* and *V*), although it was given an opportunity to operate only on *Ranger III,* where it performed normally, as described in Ref. 6. This Report concerns the development, testing, and operation of the Block III propulsion system utilized on *Rangers VI–IX.*

During the formative stages of the *Ranger* design, the decision was made to employ the modular concept for the propulsion system design. This was a significant and practical decision since the size of the system was small enough to allow the use of high-safety-factor tanks, permitting the system to be fueled and pressurized several days prior to installation into the spacecraft. It also eliminated the necessity for umbilical lines and quick disconnects. In addition, the interfaces were minimized by utilizing this modular approach. The final system design required only alignment and bolting into place, and the mating of electrical connectors to the telemetry harness, the squib harness, and the jet vane actuator harness in order to integrate the propulsion module into the spacecraft.

A disadvantage of this modular approach is that the propulsion system must be fueled and pressurized while personnel are working around it. A pressure-vessel safety factor of 2.2 (ratio of burst pressure to maximum working pressure) was determined to be adequate to ensure personnel safety. For a small system such as this one, the increase in weight associated with this safety factor was found to be very small.

A test philosophy was adopted which provided for (1) complete inspection of all parts composing the propulsion system, (2) selective assembly of these parts into the complete system, and (3) extensive pressure-leak testing of the completed system. At no time was a *flight* propulsion system fueled, pressurized, or fired prior to the actual mission.

This particular approach to reliability was, in part, necessitated by the presence of explosively-actuated valves which were capable of a single operation only. Thus, test firing of a system would expend those valves intended for flight use. In addition, it was felt that the conduction of extensive test operations, including handling and pressurizing operations, could contaminate the system such that the net reliability would be lower than if the system remained in a clean condition.

This limited test program was compensated for by an extensive type-approval (TA) test series, wherein all system components were subjected to severe environmental extremes prior to their assembly into a TA propulsion system. This particular system was then exhaustively tested, as described in Section IV of this Report.

The propulsion system utilized on Block III (*Rangers VI, VII, VIII,* and *IX*) was very similar to that utilized in Block II, although a larger fuel tank was incorporated in the new design. The other major area receiving change was the oxidizer start cartridge. In the case of Block III a bellows-reservoir type of cartridge was utilized in lieu of the long-tube-burst disc approach of Block II. This improvement was implemented to increase the potential reliability of the system, although the original design apparently worked successfully in at least two cases during flight (*Ranger III* and *Mariner II*).

II. DESCRIPTION AND OPERATION OF THE SYSTEM

A. Description

The *Ranger* propulsion system, shown schematically in Fig. 1, consists of a small rocket engine which develops 50 lb of thrust and utilizes anhydrous hydrazine as a fuel. The rocket is a constant thrust device, whose injection pressure is derived from compressed nitrogen gas which passes through a pressure regulator and forces the fuel from a bladdered propellant tank into the rocket engine. The engine contains a quantity of JPL Type H-7[1] catalyst to accelerate the decomposition of anhydrous hydrazine.

[1]Available commercially from Harshaw Chemical Company, Cleveland, Ohio.

COMPONENTS

1 ROCKET ENGINE
2 IGNITION-CARTRIDGE GN_2 FILL VALVE
3 IGNITION-CARTRIDGE GN_2 FILL RESERVOIR
4 IGNITION-CARTRIDGE ACTUATION VALVE
5 IGNITION-CARTRIDGE OXIDIZER RESERVOIR
6 IGNITION-CARTRIDGE OXIDIZER FILL VALVE
7 PROPELLANT SHUTOFF VALVE
8 PROPELLANT START VALVE
9 PROPELLANT-TANK FILL VALVE
10 PROPELLANT TANK
11 PROPELLANT-TANK BLADDER
12 PROPELLANT-TANK PRESSURIZATION VALVE
13 NITROGEN PRESSURE REGULATOR
14 NITROGEN FILTER
15 NITROGEN START VALVE
16 NITROGEN-TANK FILL VALVE
17 NITROGEN-TANK
18 NITROGEN SHUTOFF VALVE
19 VISUAL PRESSURE GAGE IGNITION-CARTRIDGE 0-500 psi

INSTRUMENTATION

PRESSURE

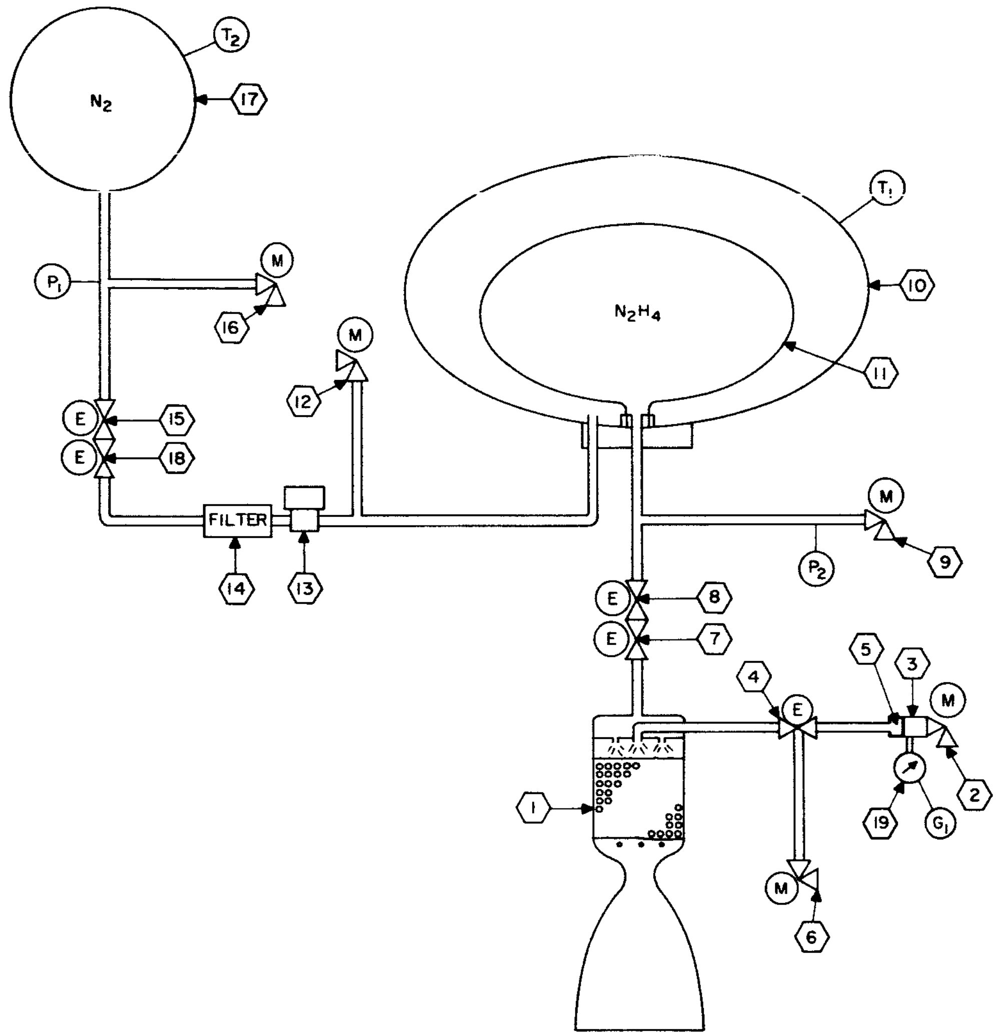
P_1 NITROGEN TANK
P_2 PROPELLANT TANK

PRESSURE MONITORING GAGE

G_1 IGNITION-CARTRIDGE RESERVOIR

TEMPERATURE

T_1 PROPELLANT TANK
T_2 NITROGEN TANK



SYMBOLS

E TWO-WAY VALVE, EXPLOSIVE-OPERATED
M ANGLE VALVE, MANUALLY OPERATED
BELLOWS RESERVOIR
FILTER FILTER
PRESET REGULATOR
PRESSURE GAGE (VISUAL)
17 COMPONENT NUMBERS
P_1 INSTRUMENTATION NUMBERS

Fig. 1. *Ranger* Block III propulsion system, schematic

Ignition is accomplished through the injection of a small quantity of nitrogen tetroxide. Three simultaneously operating, explosively actuated valves initiate operation of the system, while two simultaneously operating explosive valves are required to terminate system operation. Dual bridgewire pyrotechnics are used in each explosively actuated valve for the purpose of attaining a higher degree of reliability. Tables 1–3 list the propulsion system nominal engine performance, pressures, and temperatures, while Table 4 lists the transducer ranges. Component weights are shown in Table 5.

Table 1. Nominal engine performance summary

Vacuum specific impulse I_{vac}[a]	231.5	lb_f-sec/lb_m
Vacuum thrust F_{vac}	50.0	lb_f
Propellant flow rate $\dot{\omega}$	0.215	lb_m/sec
Characteristic velocity c^*	4306	ft/sec
Vacuum thrust coefficient $C_{F_{vac}}$	1.7295	
Chamber stagnation pressure L/C	189	PSIA
Hot throat area A_t	0.152	in.^{2}
Expansion ratio ε	44:1	

[a]Without jet vanes.

Table 2. Nominal system pressures

Item	Nominal pressure, psia
Nitrogen reservoir, at ignition	3300
Nitrogen reservoir, at termination (maximum duration run)	700
Propellant tank, prelaunch pressurization	275
Propellant tank, operating	305
N_2O_4 ignition cartridge, at ignition	375
N_2O_4 ignition cartridge, at termination	335
Chamber pressure	189

Table 3. Nominal system temperatures

Item	Nominal temperature, °F
Nitrogen reservoir, at ignition	+70
Nitrogen reservoir, at termination (maximum duration run)	+30
Propellant tank, at ignition	+70
Thrust chamber wall, during firing	1800 to 1900

Table 4. Transducer ranges

Nitrogen tank pressure	0–3600 psia
Propellant tank pressure	0–460 psia
Nitrogen tank temperature	0 to +165°F
Propellant tank temperature	+20 to +165°F

Table 5. *Ranger* propulsion system weight breakdown

Unit	Weight, lb_m
Jet vane actuator	2.5
Cabling	2.1
Structure assembly	4.9
Engine and catalyst	2.5
Propellant tank	5.5
Propellant bladder	1.0
Propellant valve	0.6
Propellant tank pressure transducer	0.3
Nitrogen tank	1.6
Nitrogen valve	0.5
Nitrogen regulator	1.2
Nitrogen tank pressure transducer	0.3
Oxidizer start cartridge	0.9
Tubing and fittings	3.0
Squibs	0.25
Blast shields	0.50
Fuel (N_2H_4)	21.50
Oxidizer (N_2O_4)	0.06
Nitrogen gas	0.75
Total weight	49.96

B. Operation

During a normal mission sequence, the trajectory correction maneuver is performed approximately 16 hr after launch. The spacecraft is commanded to roll and pitch in such a manner that the thrust vector lies in the proper direction in order to accomplish a velocity vector correction such that both the desired time-of-arrival and point-of-impact can be obtained. During the *Ranger* Block II seismograph missions, the time-of-arrival was controlled only grossly in order that lunar impact would occur during the Goldstone Space Communications Station view period. The control of arrival time became extremely important during the Block III television missions, since one channel of the television system was automatically turned on by an on-board timer that was started at the spacecraft–booster separation event.

The required velocity increment to be furnished by the on-board propulsion system is transmitted through the Earth–spacecraft command link and is stored in the central computer and sequencer (CC&S). Initiation of system operation occurs when the squib firing assembly provides power to fire the normally-closed propulsion valves, allowing rocket engine ignition to occur. As the spacecraft is accelerated, the output of the accelerometer is routed to the CC&S. When the commanded increment is obtained, the CC&S commands the squib firing assembly to fire the normally-open propulsion valves, thus terminating engine burn. During the rocket engine firing, spacecraft attitude is maintained by the autopilot-controlled jet vanes.

The sequence of events for the propulsion system after proper spacecraft orientation and up through thrust termination is as follows (Fig. 1):

1. At the command signal from the CC&S and squib firing assembly to ignite the rocket engine, normally-closed explosive valves 15, 8, and 4 are activated, allowing regulated nitrogen pressurization of the propellant tank, propellant flow to the rocket engine, and injection of a small quantity of nitrogen tetroxide to the rocket engine.

2. Hypergolic ignition ensues, followed by continuous catalytic decomposition of the anhydrous hydrazine.

3. At the command signal from the CC&S and squib firing assembly to terminate rocket thrust, normally-open, explosively-actuated valves 18 and 7 are activated, thereby terminating propellant flow to the rocket engine and isolating the remaining pressure in the nitrogen sphere from the propellant tank.

C. Description of Subassemblies

Section III contains a detailed description of each component. This section discusses the major subassemblies, which are identified in Figs. 2 and 3.

The heart of the propulsion system is the monopropellant rocket engine. The design of this efficient, minimum size decomposition chamber for monopropellant hydrazine required determination of the optimum combination of injector design, catalyst bed sizing, and ignition system. The nozzle expansion area ratio ε is 44:1. This is not the optimum ratio; however, space limitations established this configuration.

The requirement of zero-gravity ignition for the propulsion system necessitated the development of a unique ignition system. This system utilizes 13–17 cm^3 of nitrogen tetroxide contained within a metallic bellows which is pressurized by gaseous nitrogen. The outlet of the "cartridge" is sealed with a normally-closed, explosively-actuated valve. Energizing this valve allows the nitrogen to force the oxidizer out of a bellows reservoir and into the decomposition chamber where hypergolic ignition with the fuel occurs.

The pressure regulator which regulates the flow of nitrogen into the fuel tank was developed by JPL, while the explosive valves were manufactured commercially[2] In the fuel circuit, a normally-closed and a normally-open valve are combined into a single body. The inlet port of the normally-closed valve is bolted directly to the propellant tank outlet manifold, while the normally-open section is connected directly to the engine flexible line inlet. The high-pressure nitrogen circuit utilizes a similar set of valves contained within a single body. In this case the valve bolts directly to the high-pressure tank manifold. The oxidizer start valve is a normally-closed unit. Since all the oxidizer is expended during engine start, there is no requirement for a normally-open unit.

The jet vane actuators provide thrust vector control by use of vanes installed in the nozzle exit. The vanes are capable of a maximum pitch and yaw restoring moment about the vehicle center of gravity of 3.2 ft-lb, and a minimum roll moment of 0.1 ft-lb.

The fuel tank and nitrogen tank temperature transducers are bonded to the tank with epoxy cement. The purpose of the temperature transducers is to facilitate reduction of in-flight data in the calculations of gas pressure. The fuel tank and nitrogen tank pressure transducers are threaded into their respective manifolds. The purpose of the pressure transducers is to transmit, via the spacecraft telemetry link, the pressure levels in the nitrogen and fuel tanks. These pressure levels are required prior to launching and prior to committing the spacecraft to a midcourse maneuver.

[2]Conax Corporation, Buffalo, New York.

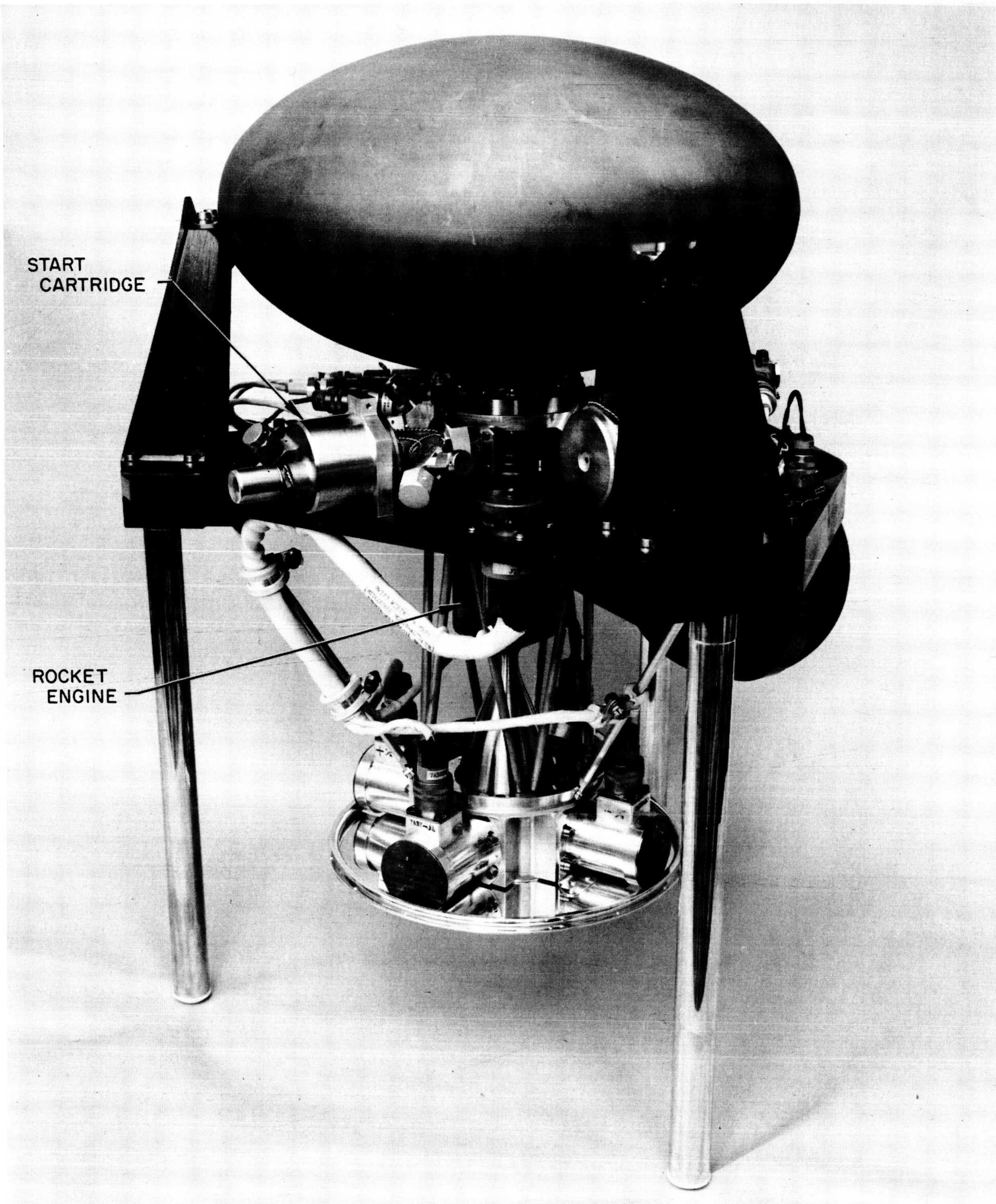


Fig. 2. *Ranger* Block III propulsion system, view 1

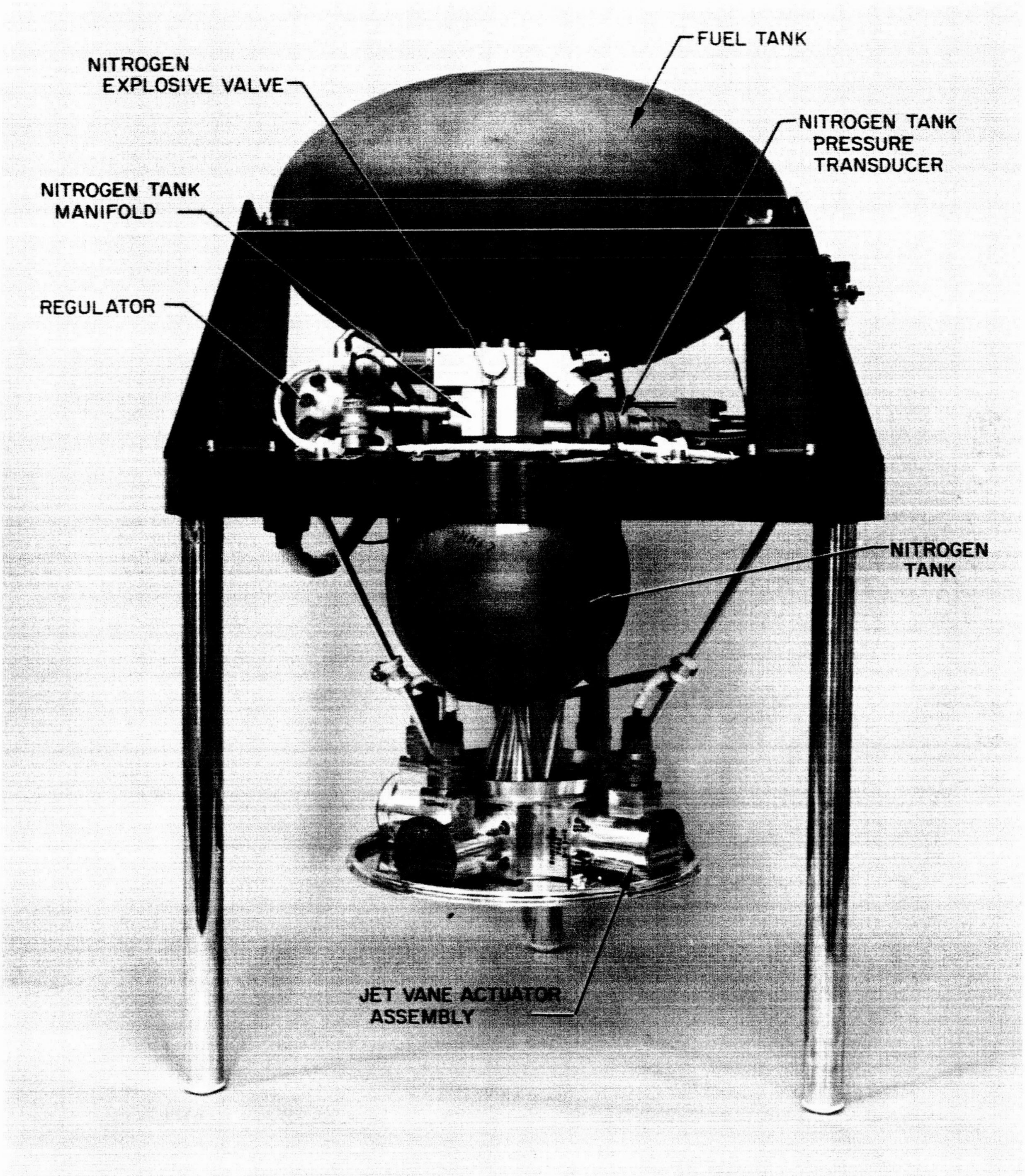


Fig. 3. *Ranger* Block III propulsion system, view 2

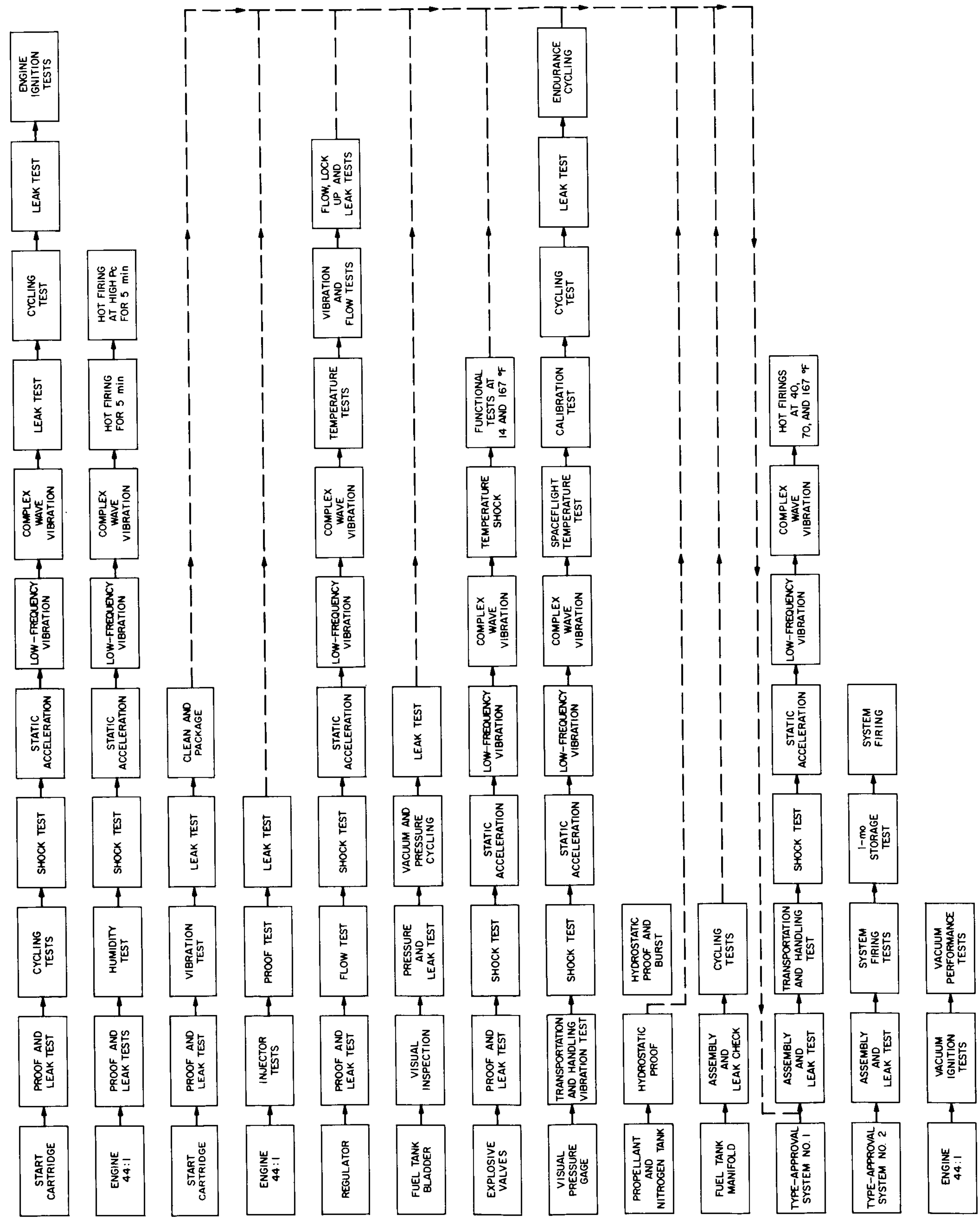


Fig. 4. Type-approval test sequence

III. COMPONENT DEVELOPMENT AND TESTING

A. General

The component designs were directed toward maximizing reliability and obtaining performance reproducibility. Rigorous development tests were performed on the components to assure that the design goals were achieved. The requirement that personnel work in close vicinity of the loaded propulsion system led to the use of a minimum safety factor of 2.2 for the component design burst pressure. The propulsion system components described in this section include the rocket engine, start cartridge, nitrogen tank, propellant tank, bladder, pressure regulator, and explosively actuated valve assemblies.

Four component design changes were incorporated in the Block III midcourse propulsion system when compared with the Block II system: (1) a new pressure regulator was designed and built by JPL to replace a regulator manufactured by an industrial concern, (2) the oxidizer start cartridge was redesigned to incorporate a positive-displacement bellows assembly, (3) protective shields were incorporated around the valve squibs, and (4) the fuel tank was increased in size to accommodate a larger propellant load. Figures 4–5 and Tables 6–7 show the test sequences and environmental levels to which all the components were subjected.

B. Rocket Engine

The rocket engine is a monopropellant device which decomposes anhydrous hydrazine. It consists of an injector for spraying the hydrazine into the chamber, a chamber which contains a quantity of catalyst to accelerate the decomposition of the hydrazine, and a nozzle for expanding the gases. Ignition is achieved by injecting a small quantity of nitrogen tetroxide into the chamber. Figure 6 is a schematic diagram of the device. This engine is identical to that used on the Block II system,

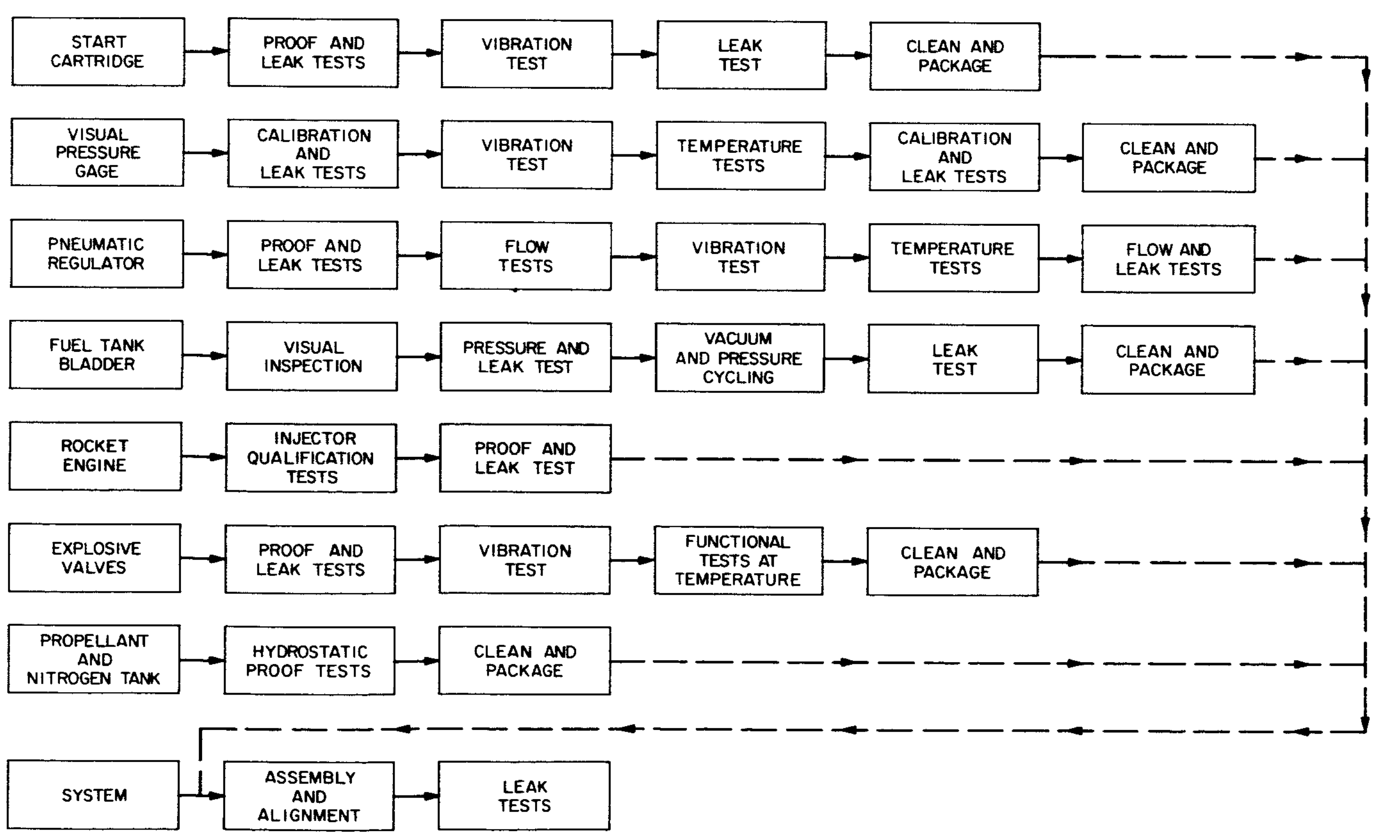


Fig. 5. Flight-acceptance test sequence

Table 6. Type-approval component test levels

Environmental test	Description
Bladder cycling	50 cycles: pressure and vacuum
Start cartridge cycling	100 cycles: pressure and vacuum
Transportation shock	42-in. drop, six times
Transportation vibration	1.3-g peak from 2 to 35 cps; 3.0-g peak from 35 to 48 cps; 5.0-g peak from 48 to 500 cps (The above vibration sweeps are sinusoidal for one hour in each of three orthogonal directions)
Humidity	Soak at 21°C and 95% humidity, 38°C and 95% humidity, 30 minutes
Shock	100 g, 0.5 to 1.5 msec, five times in each of three orthogonal directions
Static acceleration	14 g in three orthogonal directions, for 5 min each
Vibration	Low frequency— ±1.5 in., 1 to 4.4 cps, 3 min 3-g peak from 4.4 to 15 cps Complex wave— 14-g-RMS noise, 18 sec 5.0-g-RMS noise } 2.0-g-RMS sinusoid* } 600 sec 9.0-g-RMS sinusoid } 14-g-RMS noise, 18 sec
Temperature test	−10°C, 4 hours +75°C, 116 hours 10^{-4} mm Hg
Thermal shock	+75°C to −10°C, return to +75°C
Engine injector test	C^* = 4306 ft/sec, ±0.3% at 30 sec steady state P_c variation limits = ±2% Flow constant (K) = 6.2×10^8 ft^{-4} minimum 6.8×10^8 ft^{-4} maximum

*Sinusoidal wave superimposed on noise.

Table 7. Flight-acceptance component test levels

Environmental test	Description
Bladder cycling	1 complete cycle: pressure and vacuum
Vibration	Complex wave: 9.0-g-RMS noise, 6 sec 3.0-g-RMS noise } 1.5-g-RMS sinusoid* } 200 sec 6.0-g-RMS sinusoid } 9.0-g-RMS noise, 6 sec
Temperature tests	0°C, 2 hr 55°C, 40 hr 10^{-4} mm Hg
Injector qualification test	C^* = 4306 ft/sec, ±0.3% at 30 sec steady state P_c variation limits = ±2% Flow constant (K) = 6.2×10^8 ft^{-4} minimum 6.8×10^8 ft^{-4} maximum

*Sinusoidal wave superimposed on noise.

except for minor modification of the oxidizer spray nozzle (2 gal/hr increase) and modification of the oxidizer and fuel line flange design. Tests indicated that the new oxidizer spray nozzle was effective in producing a better start transient and, consequently, smoother steady-state decomposition.

The fuel injector consists of four 25-gal/hr, 30-deg hollow cone spray nozzles,[3] equally spaced on a 1.40-in. diam. circle. A single 14-gal/hr, 60-deg hollow cone spray nozzle is located in the center for injecting nitrogen tetroxide into the engine for ignition. The injector is made of corrosion-resistant stainless steel (AISI 347) with the interior surface coated with a 0.020-in. layer of zirconium oxide.

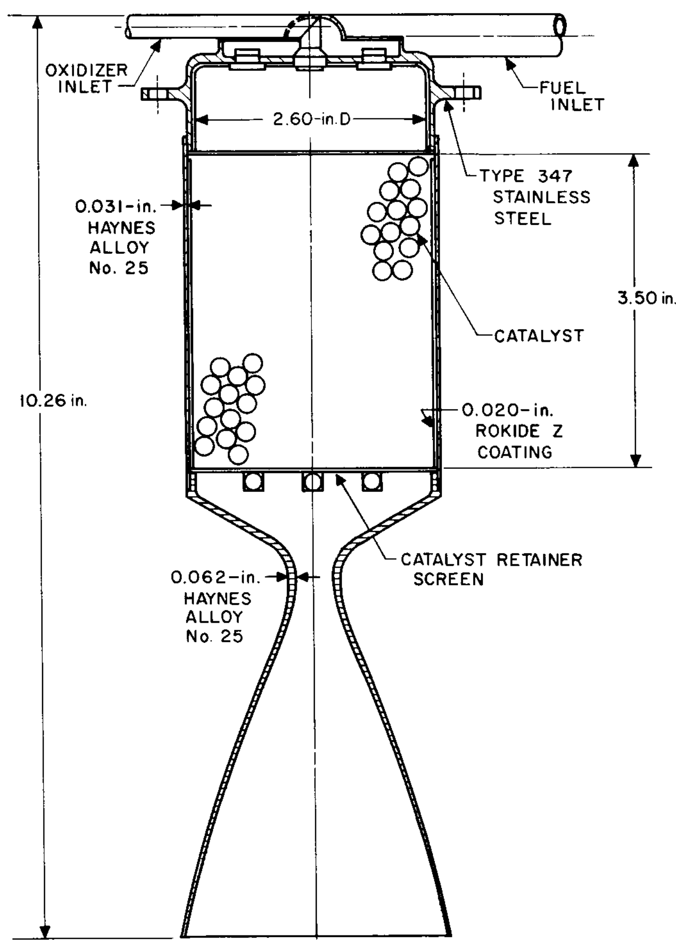


Fig. 6. Monopropellant-hydrazine 50-lb thrust rocket engine, schematic

[3]Delavan Manufacturing Company, Des Moines, Iowa.

The catalyst bed is 3.5 in. long and has a 2.7-in. diameter. The catalyst bed consists of JPL Type H-7[4] spherical particles ($^{3}/_{16}$-in. diameter) which are packed in a random-dense manner. It is prepared from an aluminum oxide support impregnated with iron, nickel, and cobalt. The decomposition chamber is 0.031 in. thick and is made of Haynes Alloy No. 25. A 0.020-in. coating of zirconium oxide is applied on the inside of the chamber.

The contour of the divergent portion of the exhaust nozzle was optimized to deliver a maximum thrust coefficient consistent with the physical boundary conditions imposed by the spacecraft configuration. A computer program, based on the method of characteristics, was employed to determine the optimum nozzle contour for an allowable length of 4.05 in. for the divergent nozzle. The nozzle contour selected resulted in a nozzle exit diameter of 2.904 in., with an associated theoretical Mach number of 4.49 and thrust coefficient of 1.793.

This rocket engine design was developed as a result of numerous tests. In these tests the following areas were investigated: (1) injector parameters, including single-spray jet designs and multispray jet designs, spray coarseness, distance of injector from the catalyst bed, and positions of jets in relation to chamber wall; (2) catalyst types, particle sizes, and bed volumes; (3) chamber pressure; and (4) quantity and injection pressure of oxidizer used for ignition. An external view of the engine is shown in Fig. 7.

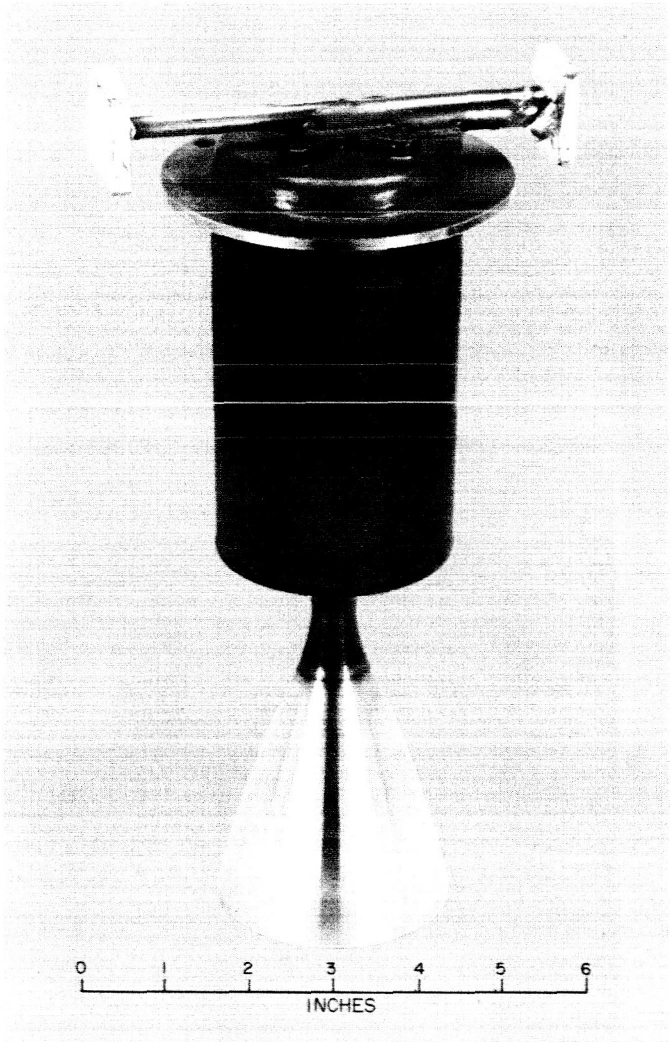


Fig. 7. Fired rocket engine, external view

Two rocket engines were subjected to TA testing. The first engine underwent the following environmental tests: shock, static acceleration, boost phase vibration, and temperature–humidity. After these tests were completed, two engine firings of 300-sec duration each and one of 100-sec duration were conducted. The first 300-sec firing was at nominal chamber pressure (190 psia) and the second at 1.5 times nominal chamber pressure (285 psia). The purpose of these tests was to demonstrate the adequacy of the design by conducting firings for extended durations (three times the normal 100-sec burn duration) and at high chamber pressure in order to impose a severe stress-time condition for the materials. The third firing was a humidity-test firing.

The second engine was subjected to flight-acceptance (FA) tests consisting of an injector performance test and a boost-phase vibration test. One firing of 60-sec duration was made with the engine installed in an inverted position. The purpose of this test was to demonstrate that, after shutdown, the propellant contained within the injector will vaporize without undergoing explosive decomposition as a result of heat soak into the injector from the hot catalyst bed. All tests were successfully completed and the engine was considered qualified for flight.

C. Oxidizer Start Cartridge

The oxidizer start cartridge, shown schematically in Fig. 8, is a device which injects 15 cm^3 of nitrogen tetroxide (N_2O_4) into the rocket engine for ignition. It consists of a cylindrical reservoir which contains a metallic bellows. This unit is fabricated of corrosion-resistant stainless steel (AISI 347). The oxidizer is filled

[4]Available commercially from Harshaw Chemical Company, Cleveland, Ohio.

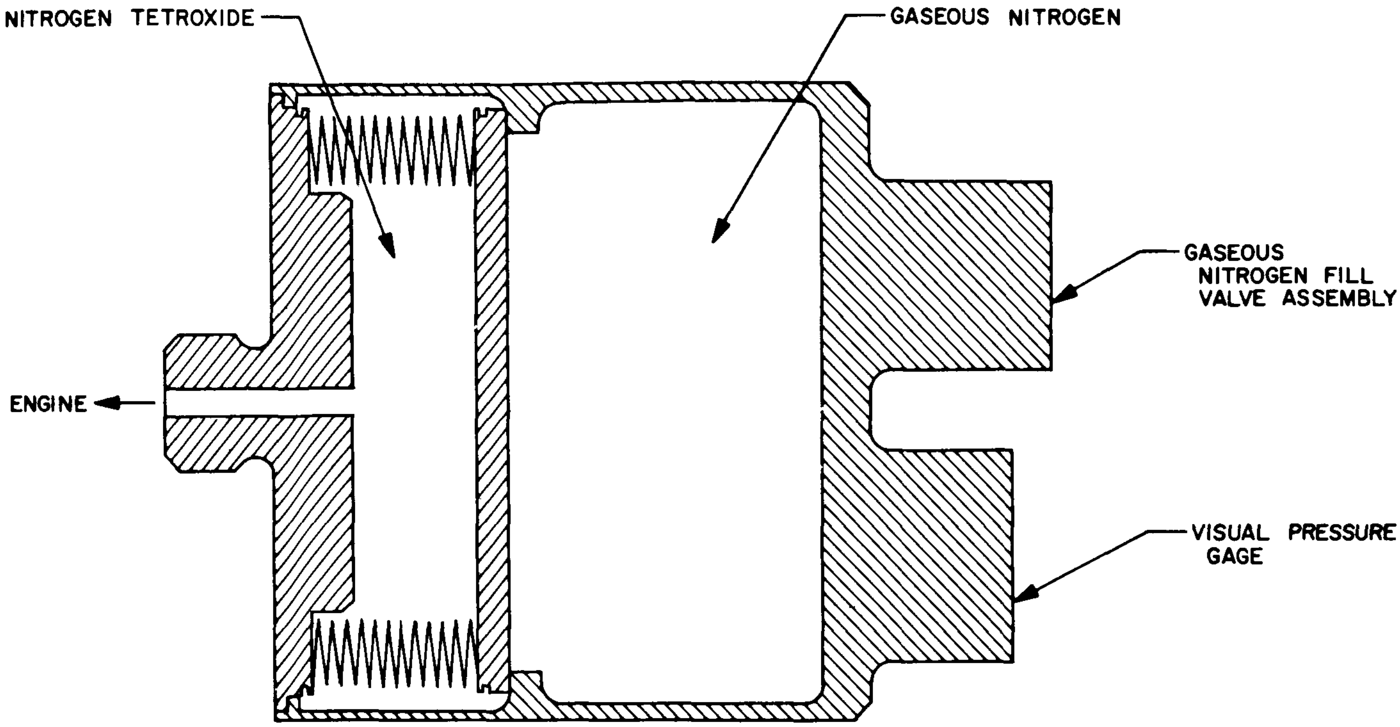


Fig. 8. Oxidizer start cartridge, sectioned view

into the internal volume of the bellows and the cartridge is pressurized with gaseous nitrogen. For engine ignition, an explosive valve is actuated, which allows the nitrogen pressure to collapse the bellows and inject the nitrogen tetroxide into the engine.

This unit was redesigned from the previous *Ranger* Block II midcourse propulsion system. The earlier start cartridge utilized a long, small-diameter tube to contain the oxidizer, and a burst diaphragm to seal off the oxidizer from the engine. It had been found difficult to obtain a good seal with the burst diaphragm. Also, it was necessary to expose the oxidizer to the atmosphere during filling operations. The oxidizer is very hygroscopic, and under conditions of high humidity such as that encountered at the Eastern Test Range (ETR), there was a tendency for the oxidizer to absorb water from the atmosphere, and to form a highly corrosive solution.

In the new design, the burst diaphragm has been eliminated and the explosive valve, which previously separated the oxidizer from the nitrogen gas, was used to seal the oxidizer from entering the engine. The new design allows the oxidizer to be filled without exposure to the atmosphere. Figure 9 is an external view of the Block III start cartridge showing the nitrogen gas fill valve assembly and the visual pressure gage.

The pressure gage is utilized to provide monitoring of the cartridge pressure prior to installation of the system into the spacecraft. It underwent a leak test and calibration prior to the environmental tests. The TA tests included static acceleration, transportation vibration, shock, temperature extremes, cycling, and an operational life-test. After completion of this series of tests, the gage was incorporated into the TA system. For flight acceptance testing visual pressure gages were subjected to calibration, boost phase vibration, and tenperature tests.

Several start cartridge assemblies were subjected to TA testing. One assembly was subjected to the mechanical and environmental tests of cycling, shock, static acceleration, vibration, and leak tests. Following these tests, the assembly was cycled and leak-tested again. Subsequently, the assembly performance was demonstrated during engine ignition tests. A second assembly was filled with N_2O_4 and the temperature was elevated to +165°F. It was determined from pressure and temperature measurements that a potential problem might occur at elevated temperatures due to the hydraulic expansion of the N_2O_4, forcing the bellows against the mechanical expansion stop and eventually causing failure of the bellows. The correction for this problem was to incorporate a 1.4-cm^3 fluid-removal step in the oxidizer fill procedure, thus allowing space for subsequent fluid

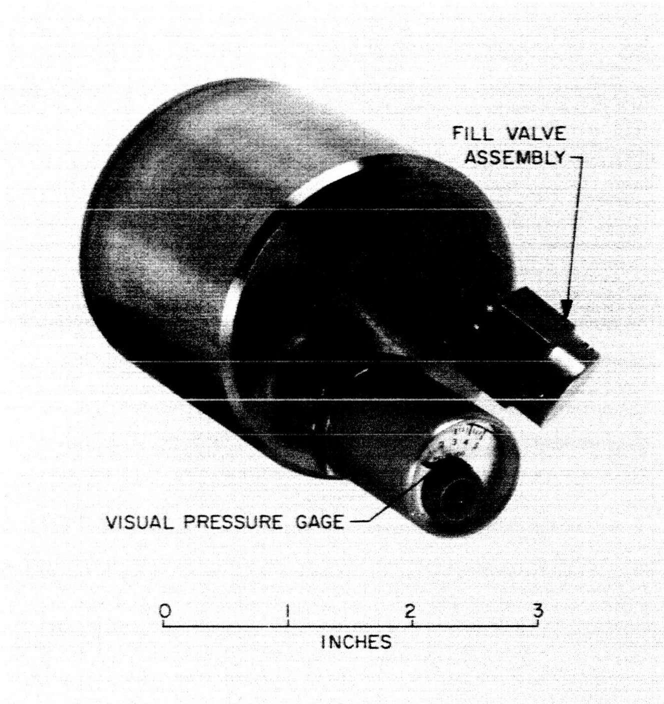


Fig. 9. Oxidizer start cartridge, external view

Fig. 10. Nitrogen tank, external view

expansion. A third assembly was subjected to FA tests of boost-phase vibration and a leak test before installation on the TA system.

D. Nitrogen Tank

The nitrogen tank (Fig. 10) was sized such that with maximum-duration propulsion system operation, the reservoir pressure of 3300 psia would decay to 700 psia. This resulted in a spherical tank with an internal diameter of 5.45 in. and a wall thickness of 0.083 in. Maximum design operating pressure was 3600 psia, occurring at a temperature of +125°F. The tank material is titanium alloy (Ti-6Al-4V). The tank design was verified when two tanks were burst at 9710 and 9850 psia. FA testing consisted of low-temperature liquid-nitrogen and ambient-temperature hydrostatic proof tests.

E. Propellant Tank

For the Block III propulsion system the propellant quantity was increased from 13.5 to 21.5 lb_m. This provided a maximum velocity increment of approximately 60 m/sec for a spacecraft weight of 800 lb. In order to attain a maximum quantity within the available space, an oblate-spheroid design was employed, as on Block II. The tank was fabricated of Ti-6Al-4V titanium alloy.

The tank has a major axis dimension of 13.16 in. and a minor axis dimension of 7.52 in., with a minimum wall thickness of 0.044 in. Figure 11 shows the external view of the fuel tank. The fuel tank nominal operating pressure is 305 psia. In addition to the propellant and bladder volumes, an allowance for 12% ullage volume was

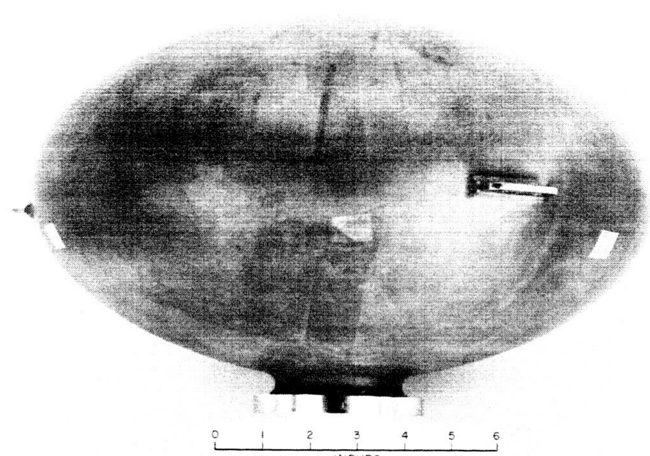


Fig. 11. Propellant tank, external view

incorporated in the tank design. In order to avoid electrical or mechanical sequencing, the tank is prepressurized with nitrogen at a nominal pressure of 275 psia so that engine ignition and regulated-nitrogen pressurization of the propellant tank can occur simultaneously. The maximum tank pressure that could be encountered under maximum-temperature conditions would be 455 psia. The design burst pressure of the tank was 1000 psia. One tank was pressurized to the burst point, which occurred at 1755 psia, resulting in a safety factor of 3.86. FA testing consisted of a low-temperature liquid-nitrogen proof test and an ambient-temperature hydrostatic proof test.

F. Bladder

A cell-type elastomeric bladder was used to contain the propellant in order to accomplish fuel positioning in a zero-*g* field. This bladder is located within the propellant tank and isolates the propellant from the nitrogen gas pressurant. During system operation, nitrogen gas under regulated pressure is admitted into the fuel tank where it collapses the bladder, forcing the propellant into the engine.

As a result of a bladder–fuel incompatibility problem, which resulted in a gradual increase in the fuel tank pressure during the *Mariner II* and *Ranger V* flights, numerous tests were conducted on a variety of elastomers to determine their suitability as bladder materials. These specimens were subjected to compatibility and permeability tests. As the result of these tests, the number of bladder material candidates was reduced to two possible elastomers: ethylene–propylene rubber (EPR) and butyl–rubber formulations. The EPR was found to be more compatible; however it was more than 30 times more permeable than the best butyl (FR 6-60-26).[5] The butyl also possessed much greater tensile strength and elongation capability. Since the FR 6-60-26 butyl caused only slight decomposition rates at expected *Ranger* spacecraft temperatures, this compound was selected for the propulsion system.

Since the compatibility test was performed with a small sample immersed in a glass vial of hydrazine, and is not directly comparable to the propellant tank case, an additional test was performed. In this test a fuel tank and bladder assembly was filled with hydrazine, pressurized to 275 psia, and stored at ambient conditions with the pressure of the tank being monitored on a daily basis. No pressure rise was noted after six months of storage.

[5]Fargo Rubber Company, Los Angeles, California.

The bladder shape is an oblate spheroid with a major axis dimension of 12.88 in. and a minor axis dimension of 7.36 in. The wall thickness of 0.035-in. is controlled by the molding techniques and is maintained as uniform as possible. The bladder is shown attached to the propellant tank manifold in Fig. 12. The bladder interior volume was designed to exceed the maximum propellant volume by 5% (at 125°F) to ensure against bladder stretching if the system should be slightly overfilled. There are several meridional ribs on the exterior surface of the bladder. These ribs ensure that the pressurizing gas will be able to flow to any portion of the bladder exterior. A mast protrudes from the tank outlet manifold into the interior of the bladder to prevent sealing of the outlet as the bladder collapses. The lip of the bladder outlet was molded into the shape of an O-ring. This provides a means of attaching the bladder to the tank outlet manifold and seals the pressurizing gas and hydrazine fuel interface.

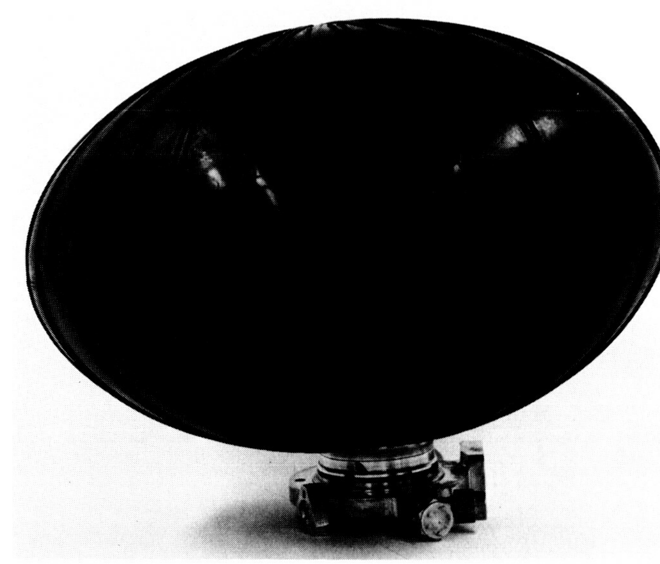


Fig. 12. Fuel-tank bladder and manifold assembly

TA testing of the bladder consisted of initial leak tests followed by 50 cycles of pressure–vacuum cycling. The bladder was then used in the TA-1 propulsion system and underwent tests described in Section IV-B. FA tests consisted of pressure–leak tests, vacuum and pressure cycling, and a final leak test.

G. Manifolds

Both nitrogen and fuel manifolds are fabricated of an aluminum alloy (6061-T6). The nitrogen manifold is bolted to the nitrogen tank and the explosively-actuated valve is mounted directly to the manifold to form a positive seal. A pressure transducer is installed on the manifold to monitor the nitrogen pressure level during the spacecraft prelaunch and flight operations.

The mast described in Section III-F above is mounted to the fuel manifold. The bladder is clamped to the manifold before the manifold is bolted to the propellant tank. The manifold has two integral fill-valve assemblies, one to admit anhydrous hydrazine into the bladder, and one to permit prepressurization of the propellant tank with gaseous nitrogen. As in the case of the nitrogen manifold, a pressure transducer is mounted to the manifold to monitor the tank pressure. Two apertures are provided within the manifold to permit regulated nitrogen gas flow into the tank and to allow fuel flow into the engine during system operation. Both manifolds were thoroughly tested in the propulsion system TA tests.

H. Pneumatic Regulator

The pneumatic regulator for the *Ranger* Block III propulsion system was designed and fabricated by JPL. The configuration developed provides consistent and repeatable results by incorporating features that minimize friction and hysteresis. A cutaway view of the component is shown in Fig. 13. This regulator maintains a constant pressure to the fuel tank of the propulsion system, nominally allowing 0.006 lb/sec of gaseous nitrogen to flow. During flow, the supply pressure may decay from 3600 psia to 80 psia above the nominal outlet pressure of 305 psia. Figure 14 shows typical regulator performances during steady-state flow conditions; these represent the final calibrations of flight units prior to installation into flight propulsion systems. The regulator must also be

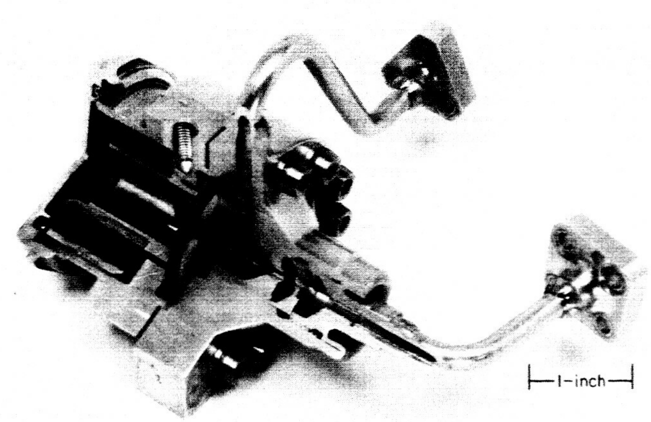


Fig. 13. Pneumatic regulator, cutaway view

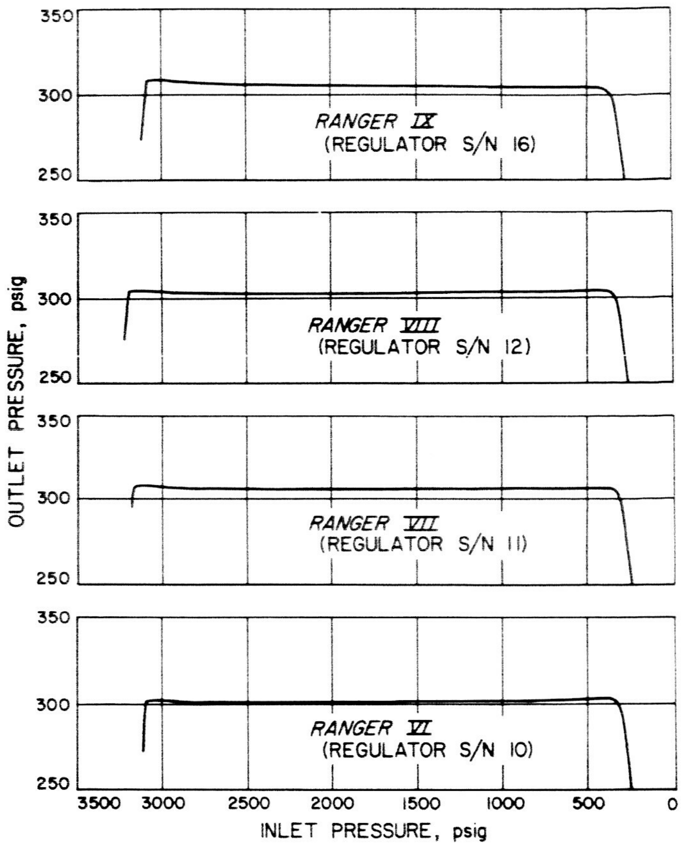


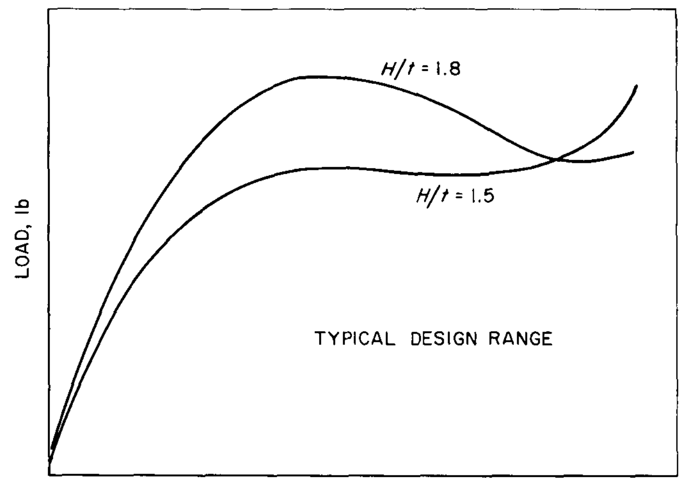
Fig. 14. Final performance calibrations of pressure regulator

capable of shutting off flow and maintaining a minimal leakage not to exceed 10 std. cm^3/hr of gaseous nitrogen (GN_2) during this locked-up condition.

The regulator is an all metal, unidirectional flow, normally-open type and operates in the following manner. The variable inlet pressure flows around the sapphire ball and seat into the low-pressure outlet chamber. Impression of this pressure on the diaphragm effective area of 1.631 $in.^2$ causes movement of the push-rod assembly and results in controlled outlet pressure. The total travel between internal stops is 0.012 in. Displacement during operation is accomplished by flexure of the diaphragm along with the mechanical back-up configuration. Predetermined outlet pressure is maintained by an equalizing force of 498 lb_f from the Belleville springs. Nonferrous structural materials are used primarily to minimize magnetic characteristics.

During the preliminary phases of the regulator development program, difficulty was experienced in complying with design performance requirements because of internal leakage and nonuniformity of springs. To meet the internal leakage rate specification, special fabrication and assembly techniques were developed, e.g., lapping the critical metallic seat area and making the final closure weld on the inlet side. Each unit was then assembled and completely tested under closely controlled conditions; the procedural methods minimized effects of contamination and resulted in compliance with the requirements.

The regulator design and performance is predicated upon the load–deflection characteristics of nonlinear type (Belleville) springs operating in the negative-rate range as shown in Fig. 15. The spring package combination, utilizing solid film lubrication, consists of an average of 18 precision springs that operate individually in a special guide to minimize hysteresis. This supporting guide concept and arrangement was developed by JPL for this particular purpose.



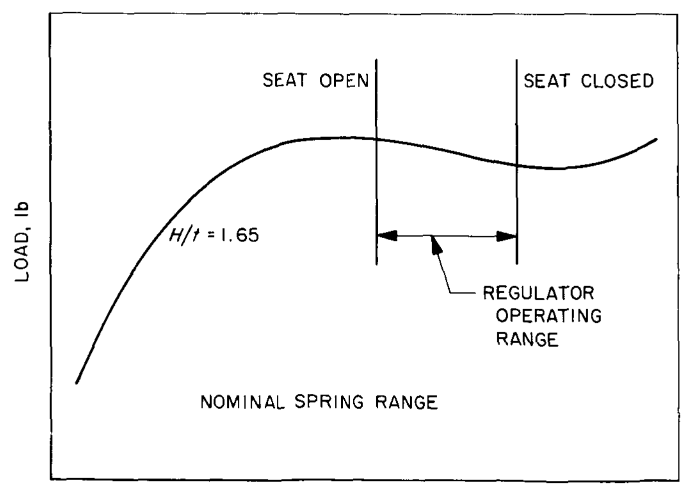


Fig. 15. Typical and nominal design characteristics of regulator spring

In order to provide uniform characteristics in the final composite spring package, the following steps were adhered to:

1. Each spring was calibrated to determine its characteristics, because of the lack of reproducibility in the same manufacturing lot. Figure 16 shows the typical scatter of maximum load for 150 springs.
2. The springs were then selectively matched based upon load, deflection, and slope determined from the individual calibrations.
3. The spring package combination was then calibrated as a complete unit. Figure 17 represents a typical final result.

Each regulator was subjected to complete TA or FA test programs (Table 8) to assure satisfactory performance prior to delivery.

Table 8. Regulator test program

Test sequence	Test performed	
	Type-approval	Flight-approval
A. Performance evaluation test		
1. Examination of product	Yes	Yes
2. Proof pressure	Yes	Yes
3. Leakage test	Yes	Yes
External leakage test	Yes	Yes
4. Operation	Yes	Yes
a. Flow rate test A (calibration)	Yes	Yes
b. Flow rate test A-1	No	Yes
5. Leakage	Yes	Yes
Internal leakage test A-1	Yes	Yes
B. Flight environmental-boost phase		
1. Shock	Yes	No
2. Static acceleration	Yes	No
3. Vibration	Yes	Yes
4. Flow rate test A (calibration)	Yes	Yes
C. Flight environment–space flight		
1. Low temperature test	Yes	Yes
Flow rate during test	Yes	Yes
2. Flow rate test A (calibration)	Yes	No
3. High temperature test	Yes	Yes
Flow rate during test	Yes	Yes
4. Flow rate test A (calibration)	Yes	No
5. Vibration:		
Three-planes flow during test	Yes	No
6. Flow rate test A (calibration)	Yes	No
7. Internal leakage test A	Yes	Yes
8. Flow rate test A (calibration)	Yes	Yes
9. Flow rate and lockup test	Yes	Yes

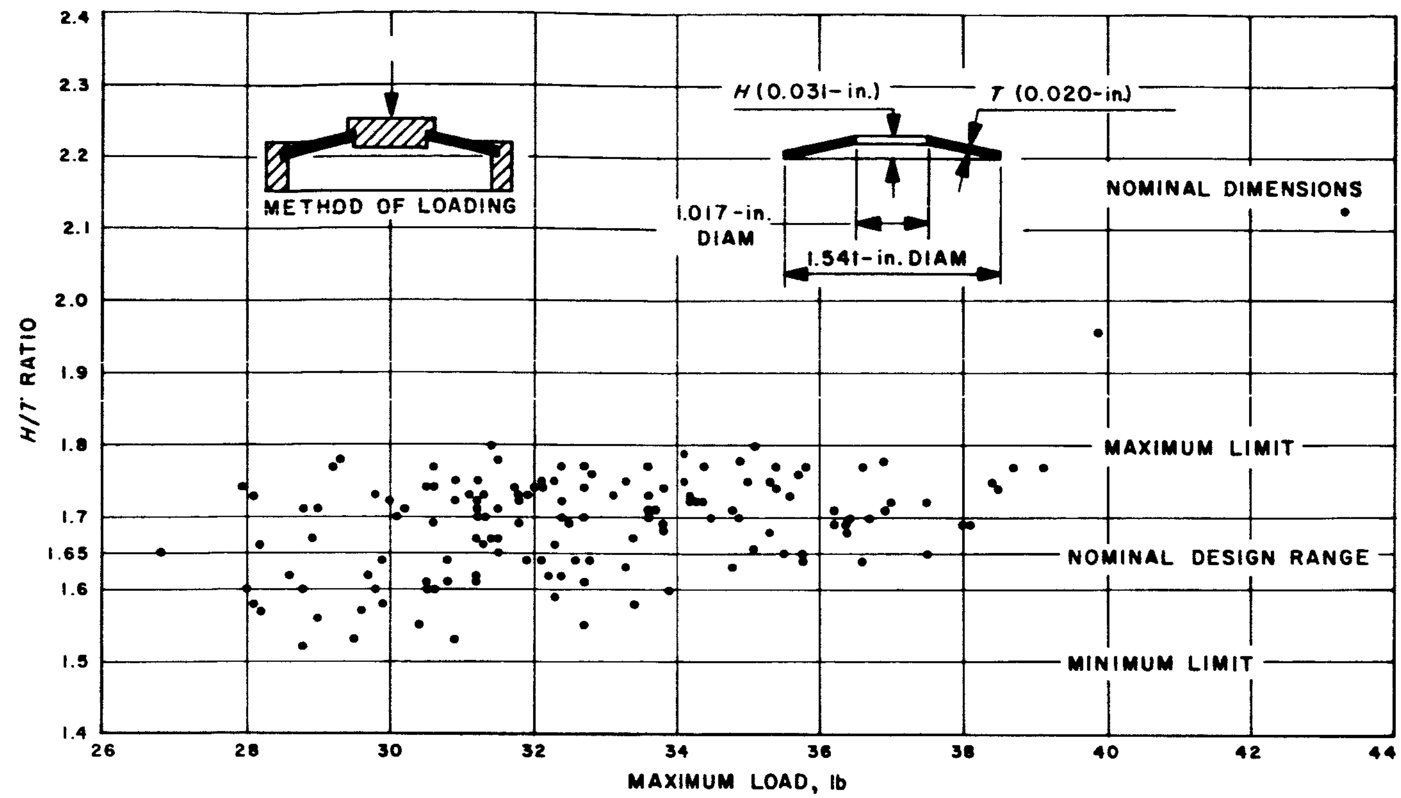


Fig. 16. Typical scatter of design parameters for a batch of 150 Belleville springs

I. Explosively-Actuated Valves

Commercially available explosively-actuated valves[6] were employed to provide a positive seal. These valves operate in the following manner (refer to Figs. 18 and 19). Application of power to the squib installed in the normally-closed port causes it to fire and supply energy to move the normally-closed ram which shears through the parent-metal seal, and seats itself upon the threaded plug, thus allowing flow through the valve. For valve closure, a similar operation occurs, causing the normally-open ram to deform the metal seat area, forming a positive seal. The valve body is fabricated of an aluminum alloy 6061-T6 and the operating parts are stainless steel. With the exception of the external sealing serrations and the ram bores, all external and internal surfaces of the valve body were anodized. Figures 18–21 show sectioned valves (one unfired fuel valve and one each of the expended valves).

Approximately 560 firings were conducted with the explosive valves. Several sets of fuel, nitrogen, and oxidizer valves were subjected to TA tests. The formal TA program for the valves and squibs is indicated in Table 9. During functional operation tests as part of this TA program, three nitrogen valves failed to close completely. In addition, numerous squib ventings were encountered and these are discussed subsequently. However, the valve malfunctions occurred when the squibs did not vent. The nitrogen valve failures resulted in leakages through the valves of from 180 to more than 2760 std. cm^3/hr after the valve closing operation. Fortunately, the nitrogen valve closing cycle is the least critical valve function since the downstream pressure regulator allows a maximum leakage of only 10 std. cm^3/hr. A combined failure of the nitrogen valve and the pressure regulator would have been required in order to cause fuel tank overpressurization and rupture.

An extensive investigation was conducted to determine the cause of the nitrogen valve failures. The failed valves were sectioned to examine the internal nitrogen flow passage surface. It was found that the ram bore and seat had been erroneously anodized, thus preventing positive sealing. A rework removed the anodization. Subsequent tests of the reworked nitrogen valves resulted in satisfactory operation and sealing. The reworked valves were subsequently used in the flight propulsion systems.

[6]Conax Corporation, Buffalo, New York.

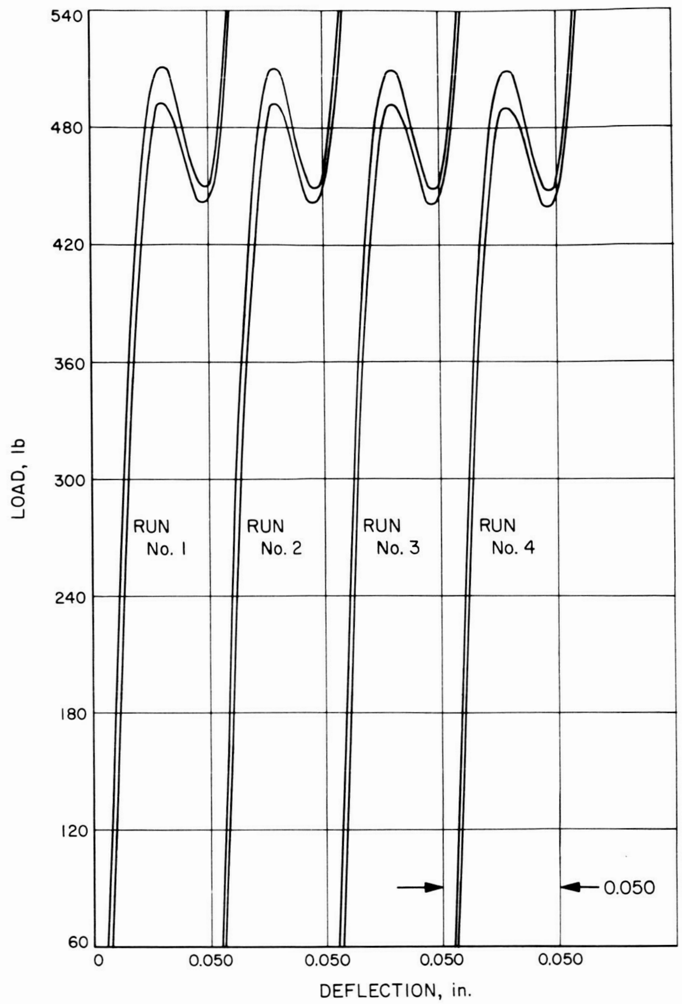


Fig. 17. Typical spring package calibration, total of 18 springs

During the test program a partial opening was observed in the normally-closed port of the fuel valve. However, full fuel flow was still achieved through the test system. This problem occurred only once during approximately 500 operations.

J. Squibs

The single-bridgewire squibs used on the early *Ranger* propulsion system were found to be unsuitable for sterilization, and the manufacturer was asked to supply another type of squib—one with two bridges, capable of withstanding higher temperatures. These new squibs were used for the first time in the successful *Mariner II* flight (August 1962) and again in the *Ranger V* flight (October 1962). During *Ranger* systems tests in mid-1963, however, it was found that the squibs had a tendency to vent or rupture. In two tests a small amount of

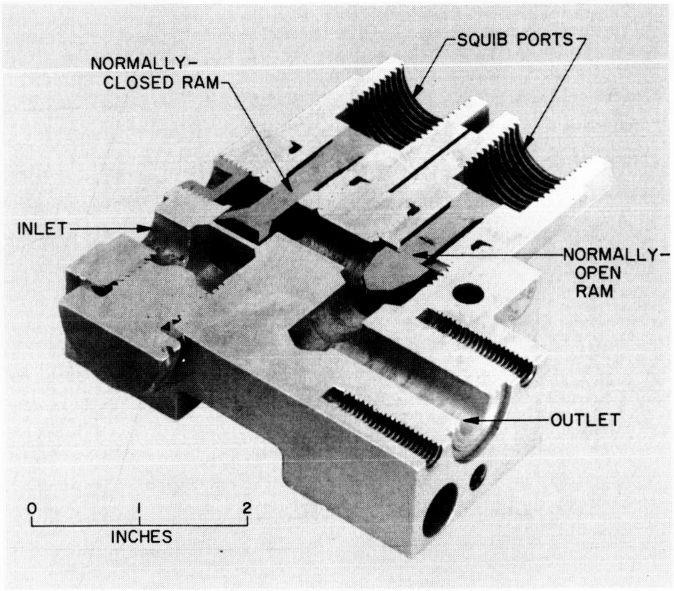


Fig. 18. Unfired, explosively actuated propellant valve, sectioned view

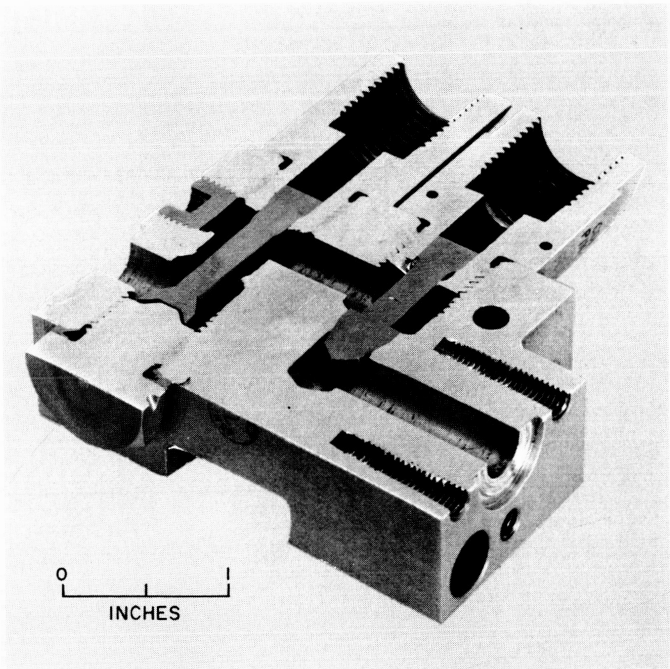


Fig. 19. Expended propellant valve, sectioned view

gas blew back through the squib connector, and in one later test the connector and squib cable were blown away from the valve. Although the observed venting rate was low (about 4%), the possibility of such an occurrence leading to either a valve failure, or damage to adjacent components, was a matter of serious concern—not only for *Ranger*, but also for future *Mariner* flights

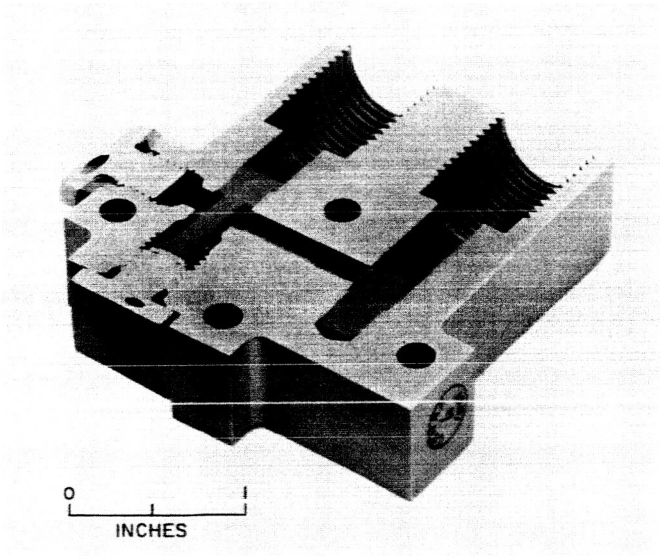


Fig. 20. Expended nitrogen valve, sectioned view

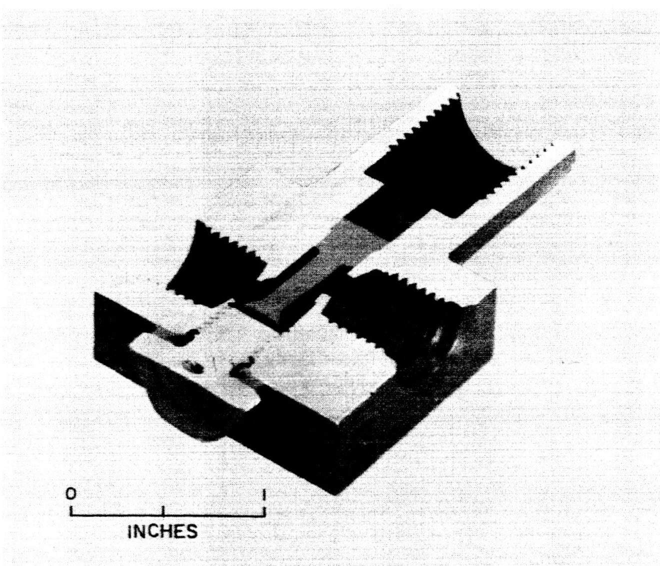


Fig. 21. Expended oxidizer valve, sectioned view

for which use of identical valves and squibs had been planned.

A simplified sketch of a squib is shown in Fig. 22. Application of a firing current through the connector causes the explosive charge to ignite and burst its rupture disc, and the pressure developed by the explosive charge is used to cause movement of the ram to operate the valve. Squib rupture occurred when the strength of the squib seals was inadequate to contain the pressure developed; in some instances the connectors separated violently

Table 9. Type-approval test program for squibs and valves

Tests (in sequential order)	Number of units		
	Individual squibs	Individual valves	Assemblies
Examination	168	8	8
Magnetic inspection	168	8	8
Radiographic examination	168	8	8
Electrical tests:			
Room temperature	168	—	8
Low temperature	6	—	—
High temperature	6	—	—
Proof pressure, valve bodies	—	8	8
Leak test, valve bodies	—	8	8
No-fire pulse test	98	—	7
Squib no-fire	78	—	—
50% firing curve	6	—	—
50% no-fire	6	—	—
50% all-fire:			
Room temperature	6	—	—
Low temperature	6	—	—
High temperature	6	—	—
Autoignition temperature	6	—	—
40-ft drop	5	—	—
Accelerated high-temperature deterioration	5	—	—
Temperature altitude test	5	3	—
Bench handling	—	3	4
42-in. drop test	56	3	4
Transportation vibration	56	3	4
Temperature shock	40	—	—
Salt fog	7	—	—
Temperature–humidity	9	—	—
Shock	56	4	4
Static acceleration	56	4	4
Vibration	56	4	4
Magnetic inspection	—	4	4
No-fire pulse test	31	—	4
Electrical tests	86	—	4
Sampling tests: squib all-fire	40	—	—
Squib operation:			
Low temperature	6	—	—
Room temperature	6	—	—
High temperature	6	—	—
Integrity tests:			
Low temperature	6	—	—
High temperature	6	—	—
Assembly operation:			
Low temperature	—	6	6
Room temperature	—	6	6
High temperature	—	6	6
Magnetic inspection	40	6	6
Temperature shock	—	7	7
Leak test valve bodies	—	7	7
Storage	40	1	1

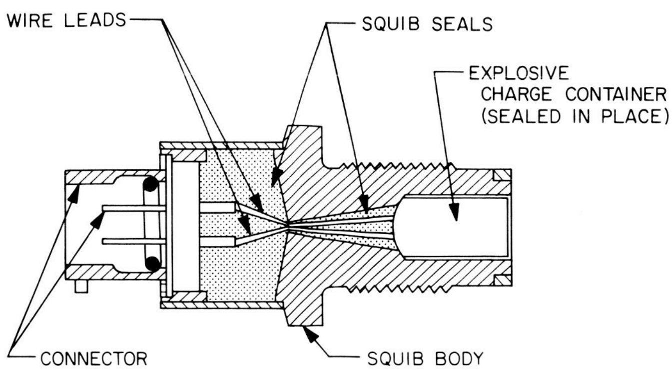


Fig. 22. Dual-bridgewire squib assembly, cutaway view

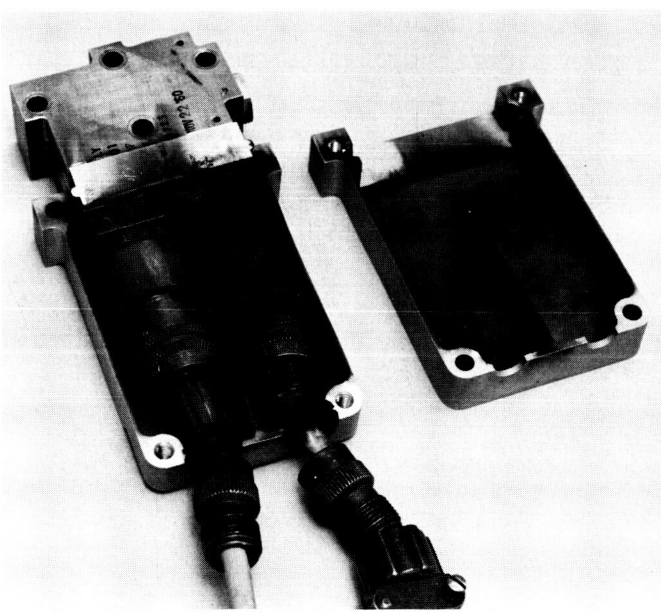


Fig. 23. Squib failure contained within the blast shield

from the squib bodies (Fig. 23). Examination of the squibs and valves (all supplied by one manufacturer[7] as proprietary items) showed that the operating pressures were much higher than could be reliably contained by state-of-the-art squib seals. Development of a new valve–squib combination was immediately undertaken for use on future *Mariner* flights, but schedules did not allow such an approach for *Ranger*. Efforts to obtain an improved seal on an urgent basis for the *Ranger* program were unsuccessful, and consequently the decision was made to continue use of the same type of squib, but to establish stringent quality control procedures and to provide "blast shields" on the valves so that ejected debris, if any, would be contained.

[7]Conax Corporation, Buffalo, New York.

A new lot of squibs was purchased and subjected to comprehensive TA and FA test programs, which were closely coordinated with similar TA and FA test programs for the valves. No significant problems were noted during the squib test program, during which many squibs were fired in blind chambers simulating tests in valves; but in the valve FA tests, a squib vent or rupture rate of about 16% was recorded.

The question of why the failure rate had increased from 4 to 16% in spite of improved quality control was answered after an intensive investigation was undertaken which revealed unsuspected weaknesses—not in the squib (whose weaknesses were known), but rather in the valves. The valves were found to be sensitive to (1) minor variations in the torque applied to mounting bolts, (2) the fit of tools used for assembly, (3) the quantity of squib thread lubricant, and (4) the fit of the blast shields. These sensitivities manifested themselves by an increase in pressure, thus increasing the tendency for the squib to rupture.

The manufacturer had concurrently developed a new process for "potting" the epoxy squib seals, and tests of a small lot of squibs manufactured by the new process showed that voids in the epoxy were smaller and less common than before. The number of samples available did not allow a firm conclusion as to whether the integrity of the squibs had or had not been improved, but it was decided to obtain a flight lot manufactured by the new process.

Because manufacture and testing of the new lot of squibs was not completed until June 1964, *Ranger VI* was flown with squibs from the first lot, extreme care being taken in handling and installation of both squibs and valves; the squibs apparently functioned satisfactorily. This new lot of squibs was subsequently used in the midcourse systems of *Rangers VII*, *VIII*, and *IX*. Extreme care was again exercised in the handling and installation of valves and squibs. If any of the squibs did rupture in service, they produced no apparent adverse effects on their spacecraft.

K. Blast Shields

During the system and valve assembly tests, an alarming number of squib failures were noted, as previously described. It was decided to provide some sort of shielding device around the squib assembly to prevent the connector and debris from damaging the adjacent squib assembly or other spacecraft components if a squib failure should occur during flight. These shields are aluminum box-like devices that are attached to the explosive valves. The blast shields for the three valves are shown in Figs. 24–26.

In order to provide design verification, a blast shield attached to the nitrogen valve was subjected to a squib failure by increasing the severity of operating conditions such that the squib would vent. The squib failed in the most severe manner; the connector was blown off the body with considerable force. The blast shield design proved capable of containing the connector and protecting the adjacent squib (Fig. 23).

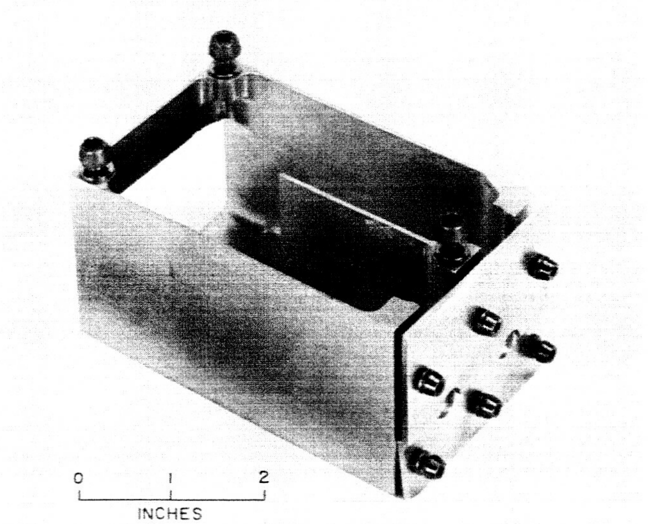


Fig. 24. Blast shield for propellant valve

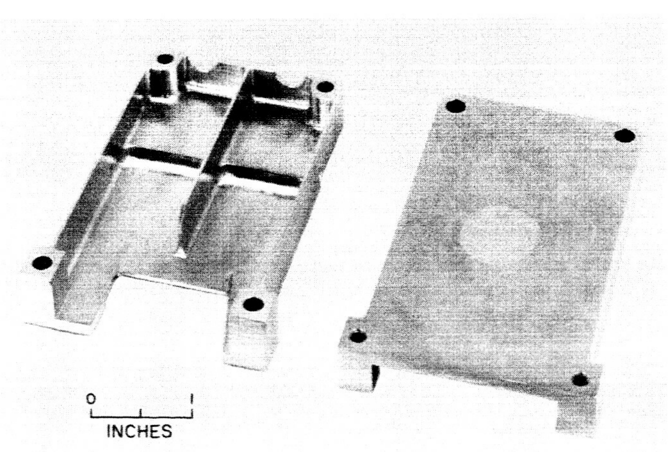


Fig. 25. Blast shield for nitrogen valve

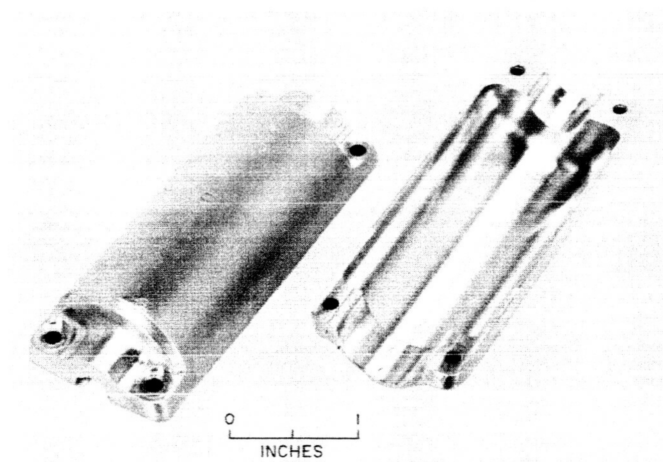


Fig. 26. Blast shield for oxidizer valve

L. Miscellaneous Components

In addition to the major components described above, the following components were installed on the propulsion system:

1. Fill Valve Assembly

Five manually operated fill valves were incorporated onto the propulsion system. (A sectioned view of a typical fill valve is shown in Fig. 27.) They were used to fill the propellant tank with anhydrous hydrazine, to prepressurize the propellant tank with nitrogen, to fill the start cartridge with nitrogen tetroxide, to pressurize the start cartridge with nitrogen, and to fill the pressurant

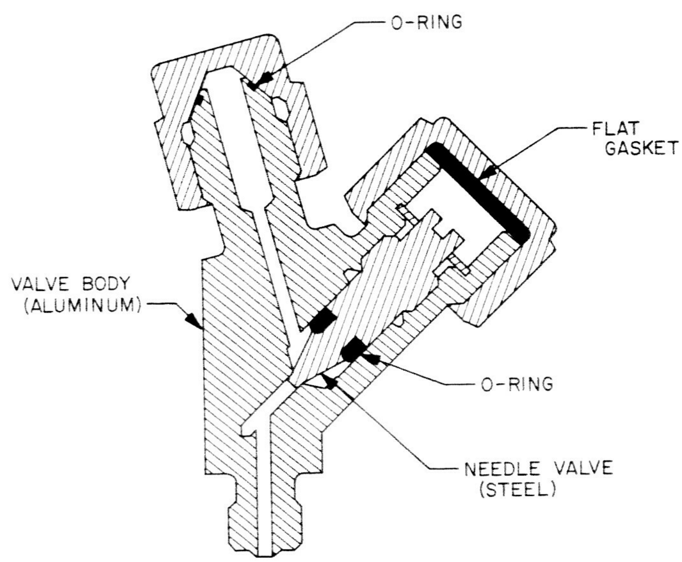


Fig. 27. Fill valve assembly, cutaway view

tank with nitrogen. The valve body is made of 6061-T6 aluminum alloy and the needle valve is made of 17-4-PH stainless steel. The softer valve body material provides an excellent sealing surface for the stainless steel needle valve. A silastic O-ring attached to the needle valve provides a peripheral seal during the time when the valve is open. A cap assembly provides a redundant seal.

2. Filters (Screens)

Filters were utilized in order to keep possible debris generated by the explosively-actuated valves from entering critical components. Two Poroloy[8] 10-μ filters were installed upstream of the pressure regulator assembly. A 10-mesh and a 100-mesh screen were located in the fuel line upstream of the injector, while a 20-mesh and a 100-mesh screen were installed in the oxidizer tube leading to the injector. All screens were made of corrosion-resistant stainless steel AISI 347.

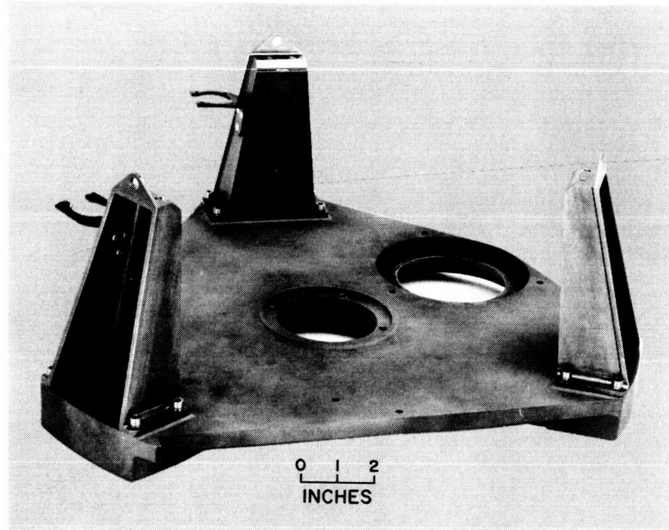


Fig. 28. Thrust plate and fuel-tank bracket assembly

3. Metallic Crush Gaskets

Metallic crush gaskets were used to seal the joining flange surfaces of the components. They were made of 1100-H14 aluminum alloy, and placed on recessed, serrated flange surfaces. The serrations consist of sharp-edge dual concentric rings which deform the aluminum seal when the components are bolted together.

4. Thrust Plate

The thrust plate was made from AZ31B magnesium alloy. In addition to providing a mounting structure for the rocket engine, this rigid structure supports the fuel tank brackets, jet vane servo support, start cartridge, pressure regulator, and nitrogen tank. It is also utilized to align the propulsion system within the spacecraft. The thrust plate with three fuel tank brackets is shown in Fig. 28.

5. Flexible Fuel Line

A flexible, Teflon-lined, wire-jacketed fuel line is used between the explosively-actuated fuel valve assembly and the rocket engine. During the Block II propulsion system tests, the fuel inlet tubing to the engine ruptured as a result of an instantaneous overpressure created by the rapid functioning of the explosively-actuated valve. Subsequent tests performed with a flexible fuel line installed in the system resolved this problem. This permitted the line to absorb the energy of overpressurization.

6. Nitrogen Tank Strap

A spider strap unit is used to hold the nitrogen tank in place upon the thrust plate. It consists of three metallic strips, located 120 deg apart. Each strip is of 0.50-in. width and 0.031-in. thickness, and is fabricated of stainless steel AISI 347, with a stud at the end.

7. Belleville-Spring Assembly

This assembly is utilized to spring-load the nitrogen tank strap to the thrust structure. With the proper setting on the spring, the assembly is maintained at a nearly constant tension irrespective of any pressure differential or temperature change. Four Belleville springs are used per strap.

[8]Model 12380, Bendix Filter Division, Madison Heights, Michigan.

IV. SYSTEM TESTING

As discussed previously, the basic philosophy governing the testing performed on the *Ranger* propulsion system was one of selective assembly and minimum handling prior to fueling operations. But during the design review period following *Ranger* Block II, it was concluded that a more exhaustive TA test program would demonstrate an adequate design margin. This revised test program consisted of two major parts—component TA and FA testing, and system TA and FA testing. Test levels are summarized in Table 10. Component tests were described in Section III; this section discusses some of the key tests performed on the system level.

A. System Type-Approval and Flight-Acceptance Testing

Two propulsion systems were subjected to TA testing. One system, designated TA-1, was assembled of components which had previously passed either TA or FA tests. This system was then subjected to environmental tests, which are detailed in Section 1 below. A second system, designated TA-2, was assembled of components which had previously passed FA testing. Complete system firings over the design range of initial tank pressurization levels were conducted with this system. Figure 29 is a schematic diagram of the system test configuration.

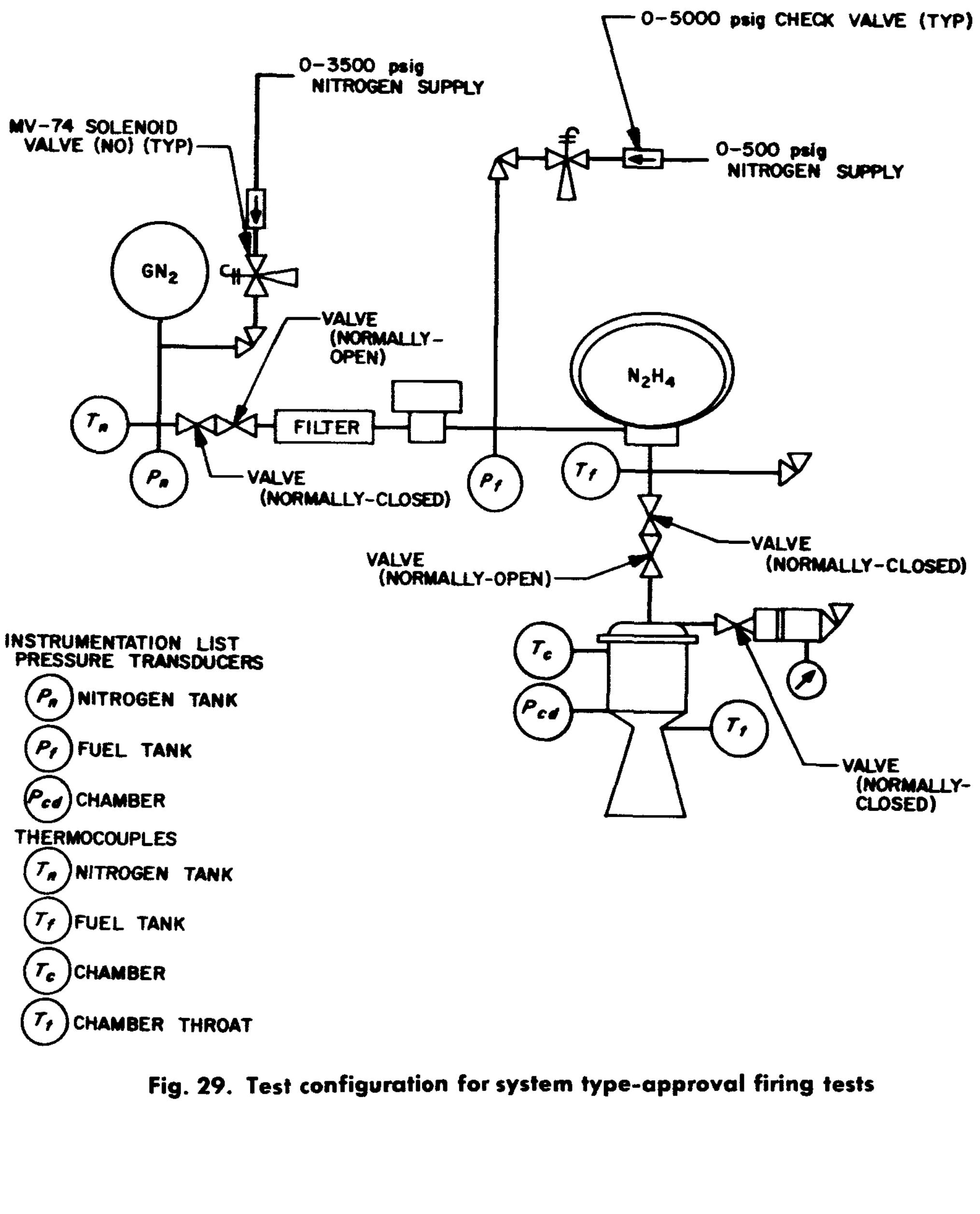


Fig. 29. Test configuration for system type-approval firing tests

Table 10. Type-approval system test levels

Environmental test	Description
Transportation shock	30-in. drop, six times (unserviced system)
Transportation vibration	1.3-g peak from 2 to 35 cps 3.0-g peak from 35 to 48 cps 5.0-g peak from 48 to 500 cps Sinusoidal, 1 hr in each of three directions (unserviced condition)
Static acceleration	14 g in three orthogonal directions, for 5 min (fuel tank and start cartridge filled with water at flight levels)
Shock	100-g, 0.5 to 1.5 msec, five times in each of three orthogonal directions (fuel tank and start cartridge filled with water at flight levels)
Vibration	X-axis: 1.35-g-RMS sinusoid 20–50 cps 3.0-g-RMS sinusoid 50–80–50 cps 1.35-g-RMS sinusoid 50–20 cps } 7 min Y-axis: 1.35-g-RMS sinusoid 20–50 cps 5.0-g-RMS sinusoid 50–80–50 cps 1.35-g-RMS sinusoid 50–20 cps } 7 min Z-axis: 2.70-g-RMS sinusoid 20–50 cps 6.0-g-RMS sinusoid 50–80–50 cps 2.70-g-RMS sinusoid 50–20 cps } 7 min (Filled with propellants at high levels)

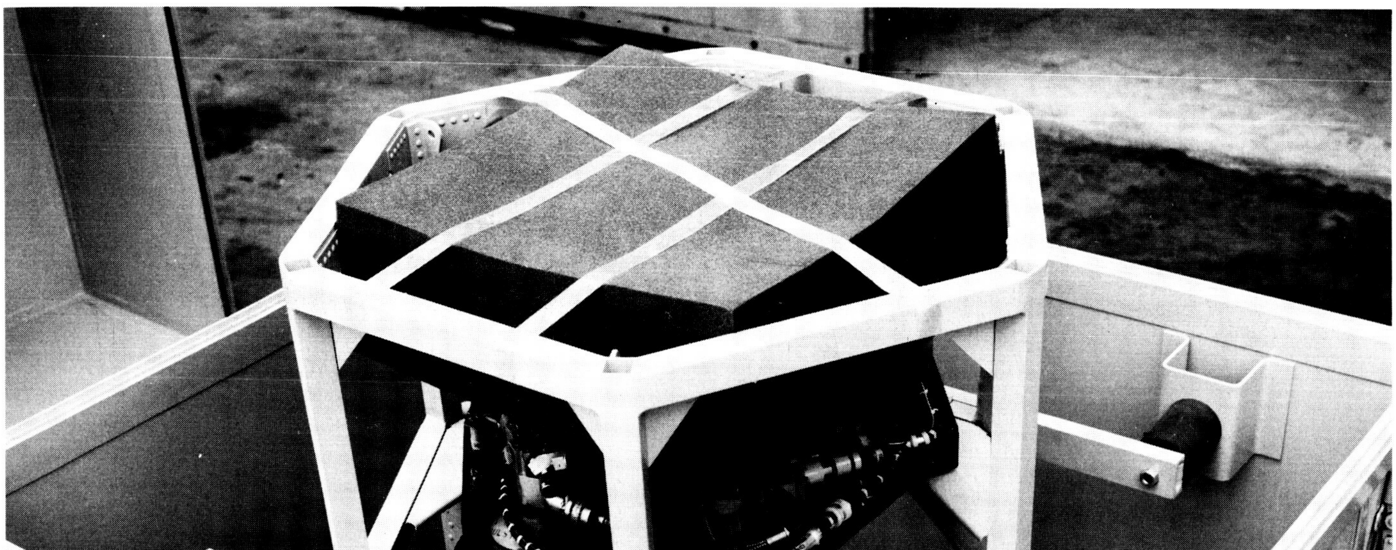


Fig. 30. System installed in shipping container

Upon completion of the TA-2 system firings, the system was used in a storage test.

1. TA-1 System

Upon completion of the assembly, this system was placed into a shipping container similar to that shown in Fig. 30 and subjected to transportation and handling tests. Figures 31–33 show the system configuration for the shock, static acceleration, and low- and high-frequency vibration tests. After each of these tests, a thorough leak check was performed. Only one anomaly occurred: a small leak in the nitrogen circuit after the low-frequency vibration test. The seal was replaced and the test was successfully rerun. No structural damage was noted during this series of tests[9] on the TA-1 system. After completion of the vibration tests, the system was subjected to temperature tests and then fired at high and low temperatures. Firings were made at +40, +70, and +167°F. Performance was satisfactory in each case.

2. TA-2 System

The TA-2 system was fired initially three times—once at nominal pressure levels, again at expected maximum

[9]The system was subjected to two vibration cycles—one with propellants and with GN_2 pressurization, and one without propellants and without GN_2 pressurization.

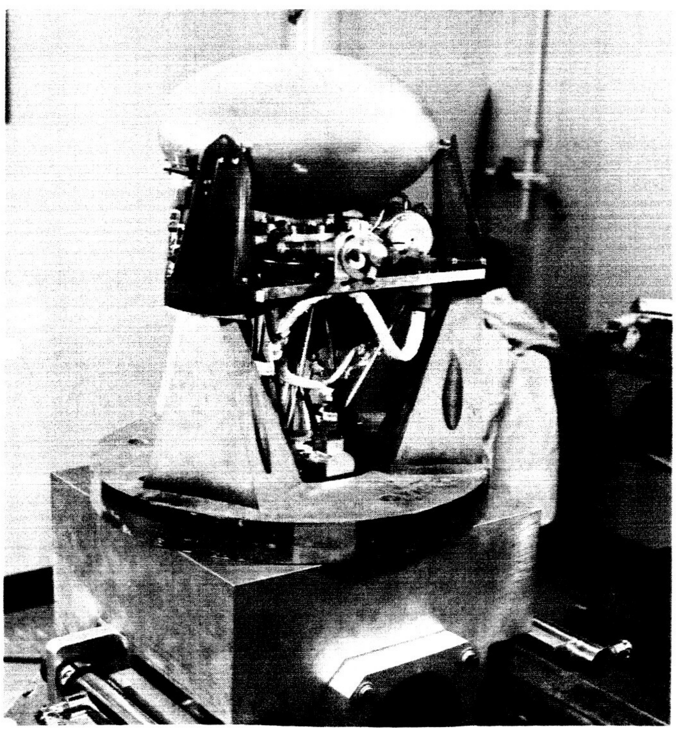


Fig. 31. System shock-test configuration

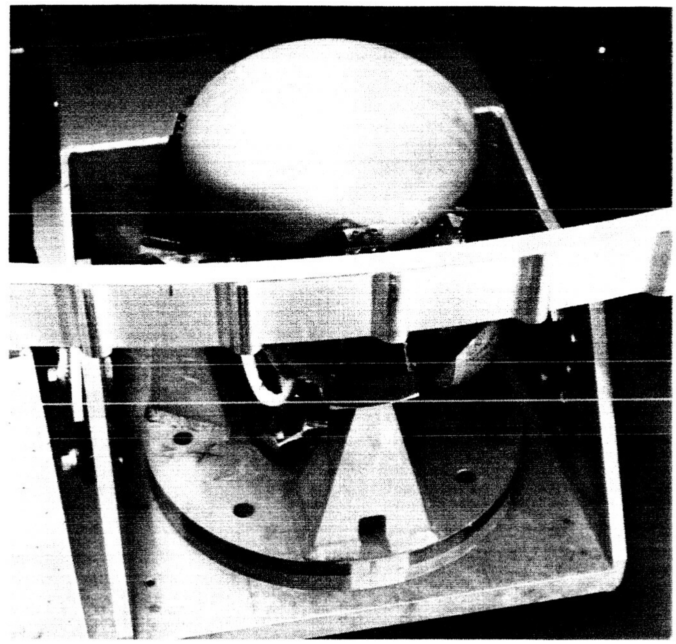


Fig. 32. System static-acceleration-test configuration

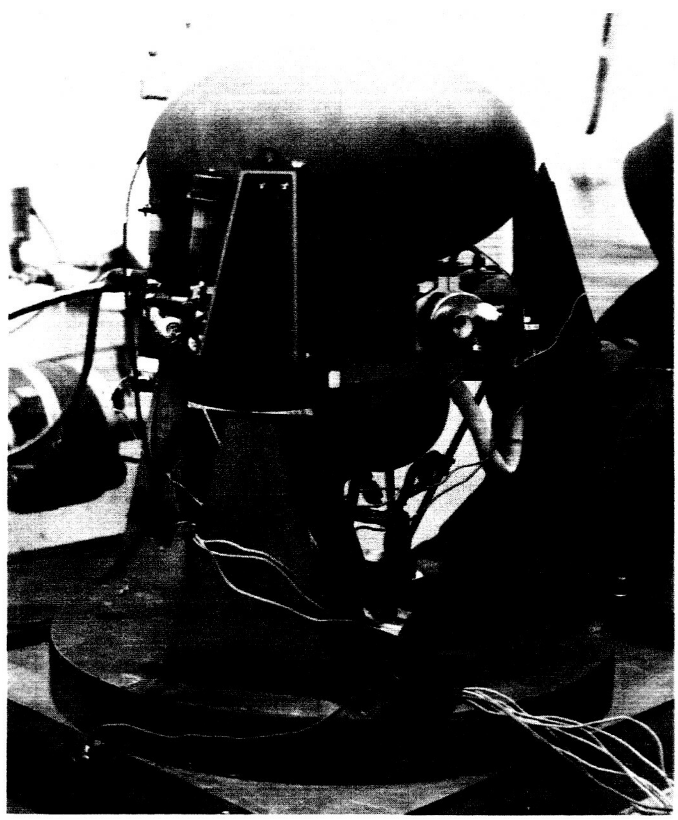


Fig. 33. System vibration-test configuration

pressure levels, and the third time at expected minimum pressure levels. The system underwent these firings with no problems. Subsequent to the above-mentioned firings, the system was fueled and pressurized to the nominal levels, and then stored at JPL Edwards Test Station at ambient temperature and pressure for 30 days. On removal of the system from storage, an attempt was made to conduct a firing; however no fuel flow was obtained. This problem was traced to a clogged fuel screen. It was determined that the clogging was caused by trapped explosive-valve particulate matter, resulting from the previous three firings. A new screen and fuel line were installed and the system performed successfully. This problem was not considered to be a flight hardware problem since one start would not produce enough material to clog the fuel screen significantly.

The TA-2 system was also used to demonstrate the compatibility of the explosive-valve squib blast shields with their respective squib–valve combination. A frequency sweep through the TA low- and high-frequency vibration levels was made with the system loaded to its nominal value. After the vibration test, a hot firing was made. The performance of the system was normal.

B. Vacuum Performance Testing

1. System Vacuum Firings

Complete system tests were made at a vacuum chamber facility[10] with a *Ranger* TA propulsion system. These

[10]Sunstrand Corporation, Pacoima, California.

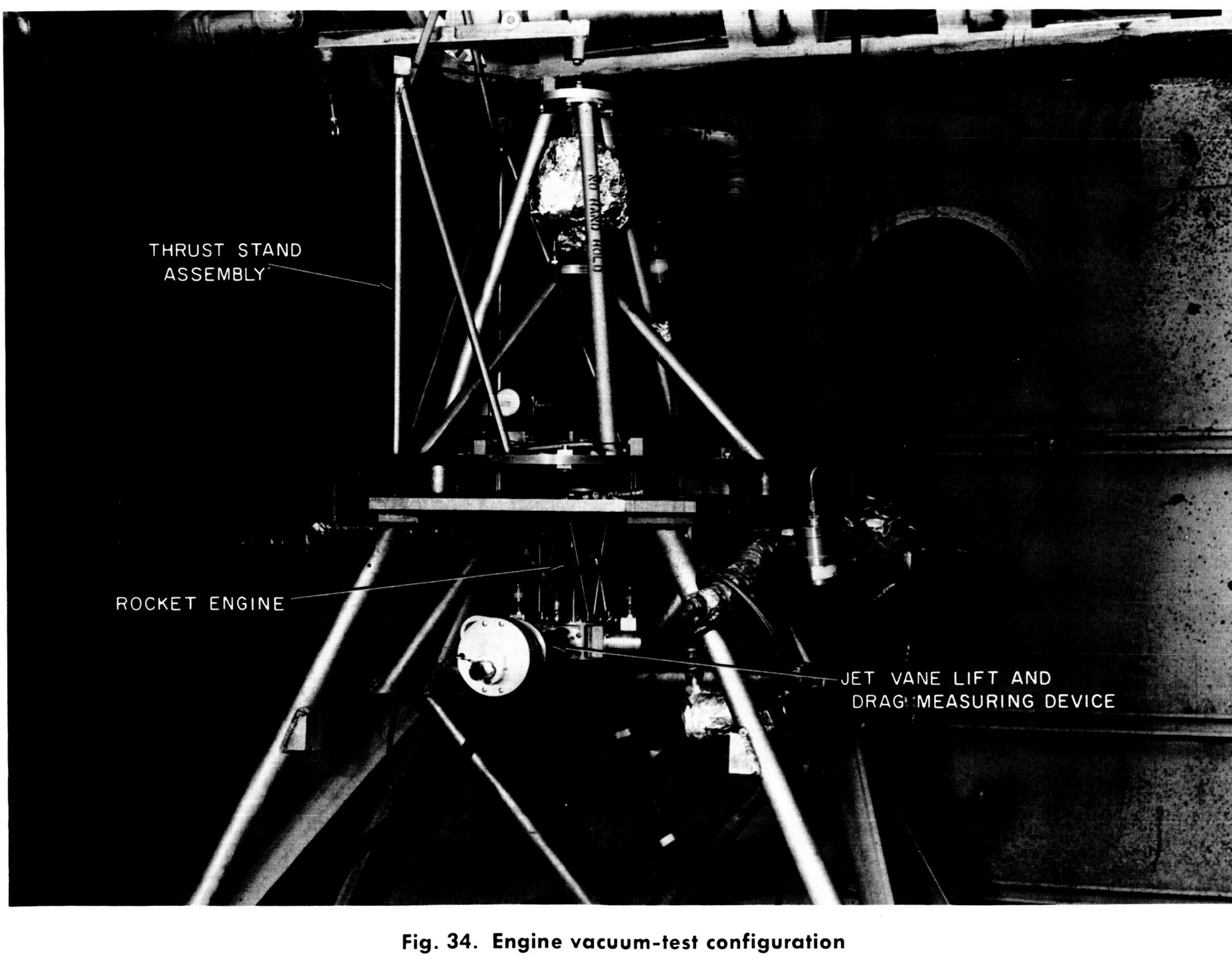


Fig. 34. Engine vacuum-test configuration

Table 11. Summary of ETS vacuum engine test data

Test No.	Time sec	P_{cd} psia	A_{t_1} in.^{2} [a]	T_t °F	A_{t_2} in.^{2} [b]	$P_{cd}A_{t_2}$ lb	Q_{fl} cfs	T_{fl} °F	w_{fl} lb$_m$/sec	P_a psia	A_e in.^{2} [a]	P_aA_e lb$_f$	F_{meas} lb$_f$	F_{vac} lb$_f$	c^*_1 ft/sec [c]	c^*_2 ft/sec [d]	$C_{F_{vac}}$	$I_{s_{vac}}$ lb$_f$-sec/lb$_m$ [c]	$I_{s_{vac}}$ lb$_f$-sec/lb$_m$ [d]	Record
Dd 28	40	192.2	0.14999	1580	0.15294	29.40	0.0034754	58.	0.2200	0.045	6.62	0.30	51.10	51.40	4299	4311	1.748	233.6	234.3	Chart
		192.44		1579	0.15294	29.431	0.0034738	57.5	0.2200	0.068		0.45	51.09	51.54	4304	4316	1.751	234.3	235.0	Digital
Dd 29	40	192.2	0.14999	1558	0.15288	29.38	0.0034573	61.	0.2185	0.060	6.62	0.40	50.60	51.00	4326	4335	1.736	233.4	233.9	Chart
		192.01		1563	0.15289	29.357	0.0034582	60.3	0.2187	0.074		0.49	50.63	51.12	4318	4328	1.741	233.7	234.3	Digital
Dd 30	22.5	191.2	0.14999	1556	0.15287	29.23	0.0034418	53.	0.2184	0.055	6.62	0.36	50.77	51.13	4305	4322	1.749	234.1	235.0	Chart
		191.17		1562	0.15289	29.228	0.0034400	52.0	0.2184	0.075		0.50	50.53	51.03	4305	4323	1.746	233.6	234.6	Digital
Dd 31	40	190.8	0.14999	1571	0.15291	29.18	0.0034311	61.	0.2169	0.045	6.62	0.30	50.80	51.10	4328	4337	1.751	235.6	236.0	Chart
		190.42		1576	0.15293	29.120	0.0034295	61.5	0.2167	0.061		0.40	50.69	51.09	4323	4332	1.754	235.8	236.2	Digital
Dd 39	70	192.3	0.14999	1547	0.15285	29.39	0.0034614	70.	0.2179	0.0612	6.62	0.405	51.46	51.87	4339	4339	1.764	238.0	238.0	Chart
		192.37		1554	0.15287	29.407	0.0034607	69.5	0.2179	0.062		0.41	51.22	51.63	4342	4342	1.742	237.0	237.0	Digital
Dd 40	70	192.3	0.14999	1556	0.15287	29.40	0.0034599	68.	0.2180	0.0640	6.62	0.424	51.66	52.08	4339	4340	1.772	238.9	239.1	Chart
		192.31		1559	0.15288	29.401	0.0034599	68.2	0.2179	0.063		0.42	51.31	51.73	4341	4344	1.759	237.4	237.6	Digital
Dd 41	70	193.3	0.14999	1554	0.15287	29.55	0.0034787	68.	0.2191	0.0640	6.62	0.424	52.15	52.57	4339	4341	1.779	239.9	240.0	Chart
		193.46		1556	0.15287	29.575	0.0034779	68.4	0.2191	0.063		0.42	51.97	52.39	4342	4344	1.771	239.1	239.2	Digital
Dd 42	70	192.1	0.14999	1562	0.15289	29.37	0.0034458	62.	0.2177	0.0637	6.62	0.422	52.39	52.81	4340	4348	1.798	242.6	243.0	Chart
		191.84		1566	0.15290	29.332	0.0034458	60.6	0.2175	0.064		0.42	52.39	52.81	4338	4349	1.800	242.8	243.3	Digital

[a] Geometric measurement at ambient temperature.
[b] Calculated based on [a] and actual temperature.
[c] Raw
[d] Normalized to 70°F.

tests were divided into three categories: (1) rocket engine performance tests (all flight-type components were employed with the exception of an externally located fuel tank); (2) the same configuration as above, but with the addition of a jet vane force-measuring device; and (3) tests of the complete flight-type propulsion system. The entire system operated normally under simulated vacuum conditions (equivalent to ~100,000 ft altitude). No excessive heat transfer was noted to any of the close-coupled components (regulator, fuel tank, or GN_2 tank). The ignition transients and system operation were identical to those tests at atmospheric pressure.

2. Engine Vacuum Firings

Table 11 illustrates the results of a series of vacuum-environment engine-firing tests that were conducted at JPL Edwards Test Station. The test setup, including the thrust stand, is shown within the vacuum tank in Fig. 34. The testing was divided into three different phases: Phase I consisted of three rocket engine ignition tests. These tests determined the vacuum ignition characteristics of the *Ranger* engine with the fuel explosive valve connected to the rocket engine by a flexline (see Section III). This mounting simulated the *Ranger* propulsion system installation. Phase II consisted of engine firings to determine the performance parameters of the *Ranger* 44:1 expansion area ratio engine (Table 11—Runs Dd 28 through Dd 31). In Phase III, four engine firings were conducted to determine the jet vane lift and drag characteristics. During each firing, the jet vane angle-of-attack was varied from −25 to +25 deg (tests Dd 39 through Dd 42). Figure 35 is a plot of the lift and drag characteristics of the jet vanes.

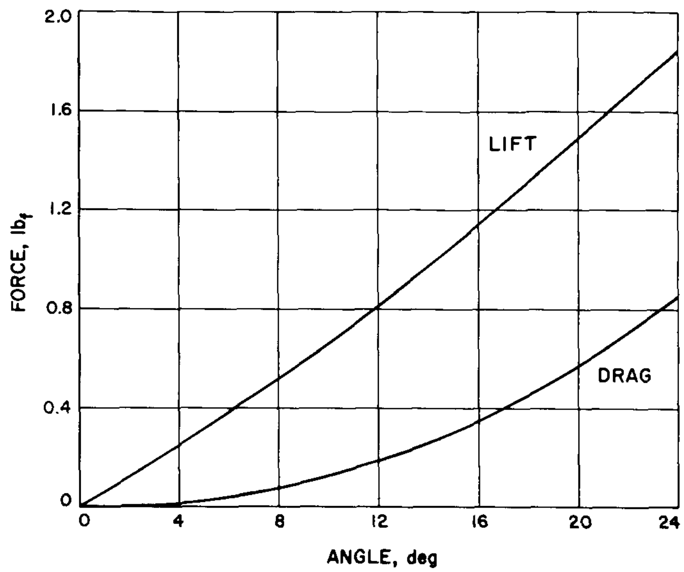


Fig. 35. Force on each jet vane as a function of angle of attack

V. FLIGHT OPERATIONS

A. Preflight Preparation

Prior to shipment of the propulsion system to the Eastern Test Range (ETR, Cape Kennedy, Florida) a preshipment leak test was performed. This was the first leak test performed subsequent to the final assembly leak test. In general, two techniques are utilized to discover leaks. In the first case, a soap solution[11] is applied to external fittings, caps, etc., and these areas are observed carefully for 3 min. Any leaks of sufficient magnitude to endanger the mission are evident by a frothing or bubbling of the solution. The second technique utilizes a small transparent tube assembly which is installed on the various tank ports with the outlet end of the tube submerged in a liquid, usually isopropyl alcohol. Any leakage through the fitting is evident by movement of the meniscus at the end of the tube. Normally a 3-min observation period is used. A typical leak check is divided into three main areas: the fuel tank, the oxidizer start system, and the nitrogen tank. A detailed discussion of the leak check procedure is provided in Section I of the Appendix.

After the preshipment leak check on the propulsion system has been completed, the unit is installed in the spacecraft for shipment by van to ETR. (When the spacecraft arrives at ETR, the propulsion system is

[11]SNOOP, Nuclear Products Corporation, Cleveland, Ohio.

removed and another leak check is performed, identical to the preshipment leak check discussed in Section I of the Appendix.)

Next the propulsion system is prepared for spacecraft system testing in the following manner: the nitrogen tank is pressurized to 200 psia with nitrogen, and the fuel tank is pressurized on both sides of the bladder to 100 psia with nitrogen by using the ground support equipment (GSE) leak check assembly (described in Section I of the Appendix, and shown in Fig. A-1). The above pressure levels are minimum levels which allow pressure transducer evaluation during spacecraft system testing.

After the spacecraft has undergone the prescribed system tests in the checkout facility, the propulsion system is removed and delivered to the ETR Explosive Safe Facility propulsion laboratory for flight preparation. If no loss in pressure has been detected from the spacecraft telemetry, the unit is prepared for explosive valve squib installation. The nitrogen and fuel tanks are vented to zero psig, the GSE leak check assembly is removed, the needle valves are closed, and the associated ports are capped. A verification is made that the squib harness has performed properly during the spacecraft system testing and that a resistance check of the explosive valve squibs has been performed by the pyrotechnic group. Simulated-squibs are then removed, squib gaskets and threads are lubricated with Fluorolube[12], and flight squibs are installed using 27 ± 2 ft-lb of torque. The squibs and triggers are then lockwired in place, squib connectors are mated, and bridgewire measurements are made. After the nitrogen valve and propellant valve blast shields are installed, a final leak check is made; again it is identical to the postshipment leak check. If no leaks are detected, the unit is ready for propellant filling. The three main propellant filling operations are oxidizer start cartridge filling, fuel tank bladder passivation, and fuel tank filling. A detailed discussion of the propellant filling operations is provided in Section II of the Appendix.

The nitrogen pressurization operation consists of pressurizing (1) the oxidizer start system nitrogen reservoir, (2) the fuel tank, and (3) the nitrogen tank. The oxidizer cartridge is pressurized at a rate not exceeding 180 psi/min, until a pressure level of 375 psia at 70°F is attained. Next, the fuel tank is pressurized at a rate not exceeding 20 psi/min, until a pressure level of 275 psia at 70°F is reached. Finally, the nitrogen tank is pressurized at a rate not exceeding 60 psi/min, until a pressure level of 3300 psia at 70°F is achieved.

The final propulsion system operations are then performed. All needle valve caps are lockwired, and a final inspection is accomplished. At this time, the final weight of the propulsion system is determined. After the transducer and visual pressure gage readings are recorded, the system is stored in the propulsion lab and monitored for a minimum of 24 hr before it is delivered to the spacecraft area.

Several spacecraft tests are performed for several days prior to launch, both in the explosive-safe area and on the pad. During these tests, spacecraft power is usually turned on, and the propulsion parameters are available. In this manner, it can be determined whether any leaks have developed in the fuel or nitrogen tank, and whether any corrective action need be taken. During the prelaunch countdown, the propulsion parameters are monitored both at the spacecraft checkout facility at ETR and at the Space Flight Operations Facility (SFOF) at JPL.

B. Space Flight Operations

After launch, control of the spacecraft is assumed by SFOF, located at JPL in Pasadena, California.

During the flight operations, it is possible for SFOF to monitor the nitrogen tank pressure, the fuel tank pressure, and the fuel tank temperature by way of telemetry data presented on a teletype (TTY) machine. During the midcourse phase of the mission, a high-speed data rate output of fuel tank pressure, nitrogen tank pressure, and nitrogen tank temperature is available on a Stromberg–Carlson 3070. The midcourse pyrotechnic start and stop commands and events are viewed on an analog recorder.

By knowing the weight of fuel on board the propulsion system, the spacecraft weight, the estimation of jet vane drag, and the system temperatures and pressures, and by utilizing the propulsion in-flight procedure, one can compute the maximum velocity increment capability of the system as well as the expected tailoff velocity. Since the spacecraft utilizes an integrating accelerometer system, the tailoff velocity must be subtracted from the required velocity to determine the programmed velocity increment. The tailoff velocity includes the thrust chamber tailoff and the time delay of the command from the central computer and sequencer (CC&S) to the pyrotechnic control, and from the pyrotechnic control to the explosive squibs along with the squib delay. After the velocity

[12]Hooker Chemical Company, New York.

which is required for the mission is known, prediction of the following parameters may be made: engine burn time, propellant consumption, final nitrogen tank pressure immediately after burn, and regulated fuel tank pressure. If any changes occur in the velocity requirement for the mission, iterations of the propulsion parameters must be made. Thus, when the motor burn occurs it is possible to determine immediately, by comparison of actual and predicted data, whether the operation was normal.

C. Flight Performance

The flight performance of the *Ranger* Block III propulsion systems is summarized in Table 12. It can be seen that all the systems performed very satisfactorily, and the accuracy of the velocity increments imparted to the spacecraft were well within the design tolerance. When looking at the comparison of actual burn time with predicted burn time, it should be noted that the predicted

Table 12. Propulsion-system flight performance summary

Spacecraft	Velocity required ΔV_{ideal} m/sec	Actual velocity ΔV_{act} m/sec	Predicted burn time $t_{b\ pred}$ sec	Actual burn time $t_{b\ act}$ sec	Predicted regulated pressure $P_{reg\ pred}$ psia	Actual regulated pressure $P_{reg\ act}$ psia	Predicted final N_2 tank pressure $P_{N_2\ pred}$ psia	Actual final N_2 tank pressure $P_{N_2\ act}$ psia
Ranger VI	41.27	41.23	67.0	69.0	301	303	1550	1305
Ranger VII	29.89	29.82	48.6	50.0	306	301	1815	1700
Ranger VIII	36.44	36.48	59.5	61.0	302	–	1620	–
Ranger IX	18.15	18.15	30.6	30.0	305	299	2230	2200

Notes:
1. Actual nitrogen tank pressure and regulated fuel tank pressure data are unavailable for *Ranger VIII*.
2. Resolution of the nitrogen tank transducer is 50 psi.
3. Resolution of the fuel tank transducer is 6 psi.
4. Actual burn time is calculated as the difference between squib-firing event blips.

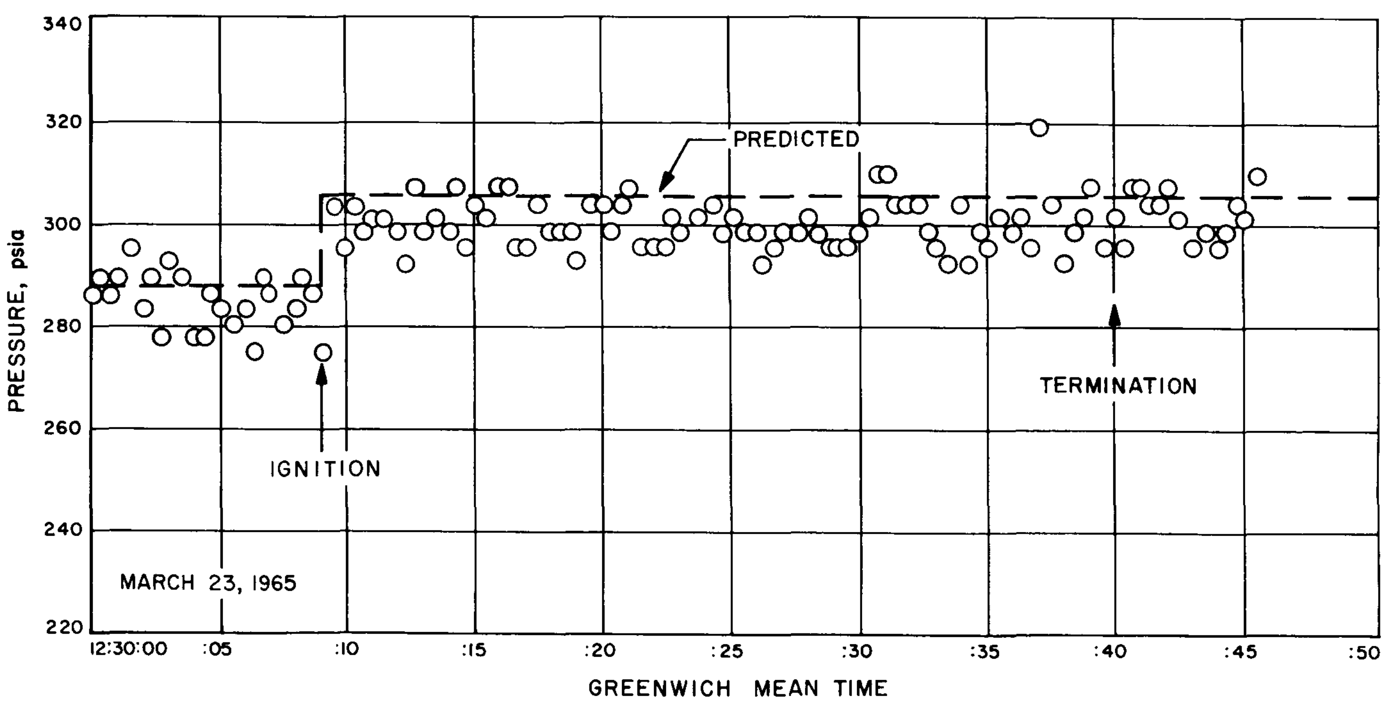


Fig. 36. *Ranger IX* propulsion-system fuel-tank pressure during engine firing

burn time depends not only upon the regulated gas pressure but also upon the estimated average jet vane drag. During the course of the program, refinements were made in the estimation of the burn time. In the earlier flights, the estimate of the specific impulse (I_{sp}) was too optimistic, but an unintentional compensation was made by making a very conservative estimate of the jet vane drag. For *Ranger IX*, enough information had been determined from previous flights to enable a much better prediction of burn time. Typical fuel tank pressure and nitrogen tank pressure plot profiles during motor burn are shown in Figs. 36 and 37, which employ actual flight data from *Ranger IX*.

D. Problems Encountered

During any operation, certain problems occur that are not anticipated, and often there is sufficient time to only make an immediate repair, rather than accomplish redesign and requalification. The most severe problems of this class encountered during *Ranger* Block III propulsion activities involved oxidizer leakage through joints that utilized K-seals[13]. Examples of such problems discovered on *Ranger VIII* follow. (*Ranger IX* encountered similar problems.)

When the postshipment leak check was performed on *Ranger VIII*, two leaks were noted: (1) at the K-seal between the oxidizer start cartridge body and the explosive valve, and (2) at the nitrogen tank fill valve dynamic O-ring seal. Both were repaired and satisfactorily leak-checked.

However, during the course of the actual propellant filling operations on *Ranger VIII*, four more leaks developed: (1) An oxidizer leak at the K-seal between the start cartridge and explosive valve was detected with methyl orange indicator paper. The oxidizer was removed and the K-seal replaced. (2) A helium–soap-bubble test detected a leak at the same location. The start cartridge was removed and the explosive-valve seat was polished. After

[13]Harrison Manufacturing Company, Burbank, California.

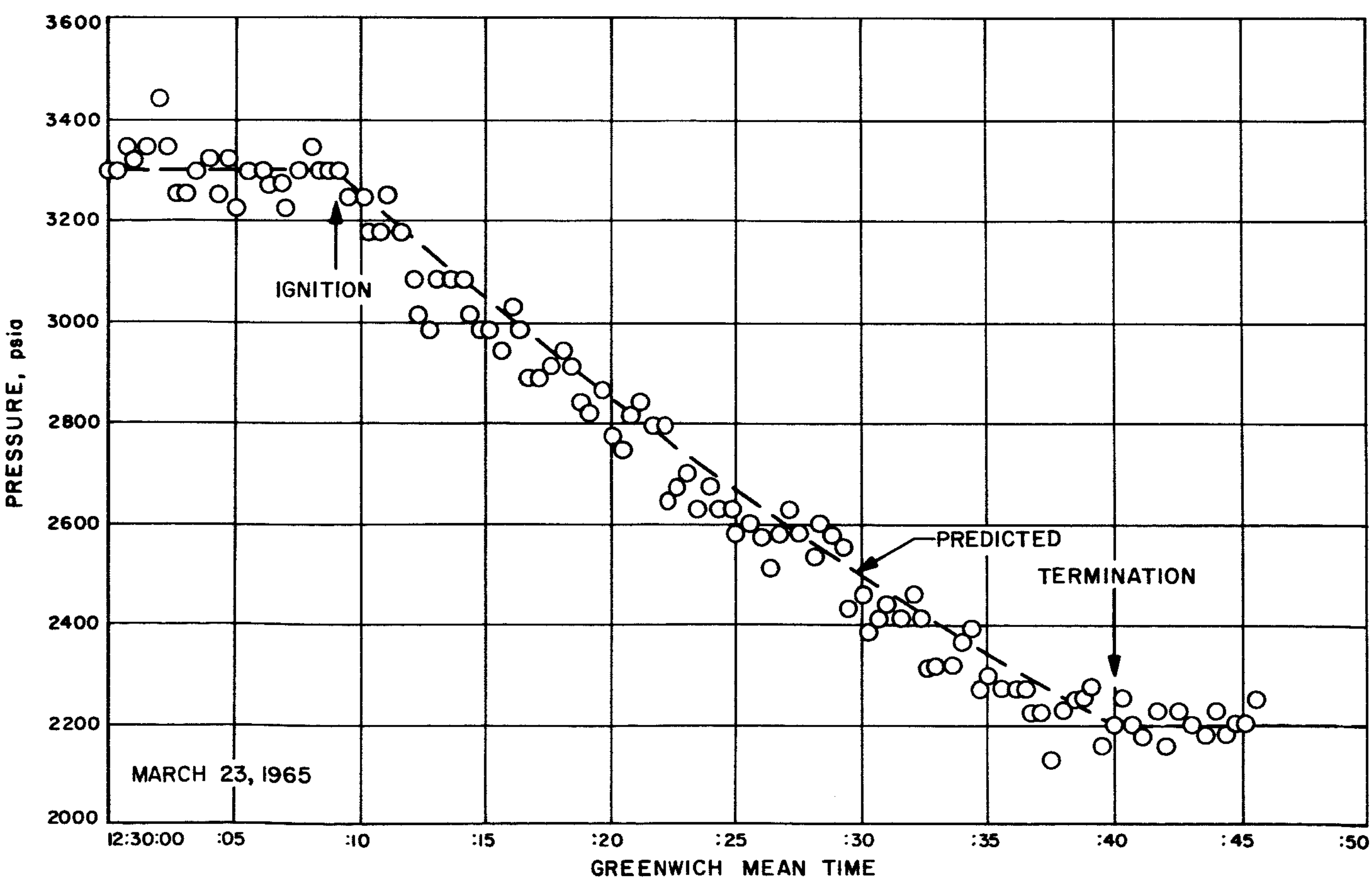


Fig. 37. *Ranger IX* propulsion-system nitrogen-tank pressure decay during engine firing

reassembly, a helium–soap-bubble leak check indicated that all fittings were tight. Oxidizer was refilled. (3) A leak was detected at the fill port assembly K-seal with methyl orange paper. The oxidizer was removed, and the seat on the explosive valve was polished. The fill port was reinstalled with a new K-seal, and a helium–soap-bubble leak check showed that all seals were tight. Oxidizer was again refilled, and no leaks were detected with the methyl orange paper. (4) A nitrogen leak was discovered at the nitrogen tank fill valve K-seal. The pressure was vented, the K-seal removed, the seat polished, and a new K-seal installed. The tank was repressurized to 3300 psig with nitrogen, and no leaks were found.

As an outcome of the discovery that an oxidizer leak could be detected with methyl orange indicator paper but not predicted by the helium–soap-bubble technique, a study was conducted to determine the sensitivity of the two methods of leak detection.

A specialist was consulted about test sensitivities regarding the mass spectrometer evacuated probe, commercial soap-bubble solutions, and methyl orange test paper (Ref. 7). It was determined that (1) the mass spectrometer is sensitive to 10^{-5}–10^{-7} std. cm^3/sec of helium, depending on operator techniques and test methods, (2) the commercial soap solution bubble test is sensitive to 10^{-4} std. cm^3/sec of helium, and (3) the methyl orange paper is sensitive to 10^{-5}–10^{-6} std. cm^3/sec of N_2O_4. It is evident, then, that the helium–soap-bubble test could not be expected to detect very minute oxidizer leaks which would cause a color change on the indicator paper. Section 3 of the Appendix describes tests performed on actual *Ranger* hardware, which verified these predictions.

APPENDIX

I. LEAK CHECK PROCEDURE

A. Fuel Tank

The fuel tank leak check is facilitated by using a special Ground Support Equipment (GSE) assembly on the fuel tank manifold (Fig. A-1). This allows pressurization to occur on both sides of the bladder and eliminates any pressure differential, as may be seen in Fig. A-2. The fuel tank is slowly pressurized to 200 psig from a 0- to 500-psig nitrogen source. A soap solution is then applied to all external fittings in the fuel tank circuit including the fuel tank pressure regulator. If no leaks are observed, the fuel fill and prepressurization needle valves are closed.

The nitrogen supply is vented and the supply line removed. A tube is installed on the GSE assembly valve and a leak check is made to verify that no leakage exists across the seats of the needle valves. Then the GSE valve is closed, the tube is removed from the assembly, and the fuel fill and prepressurization needle valves are opened. The tank and bladder are vented to 0 psig by slowly opening the GSE valve.

The leak check assembly is then removed from the fuel tank manifold so that a leak check of the bladder

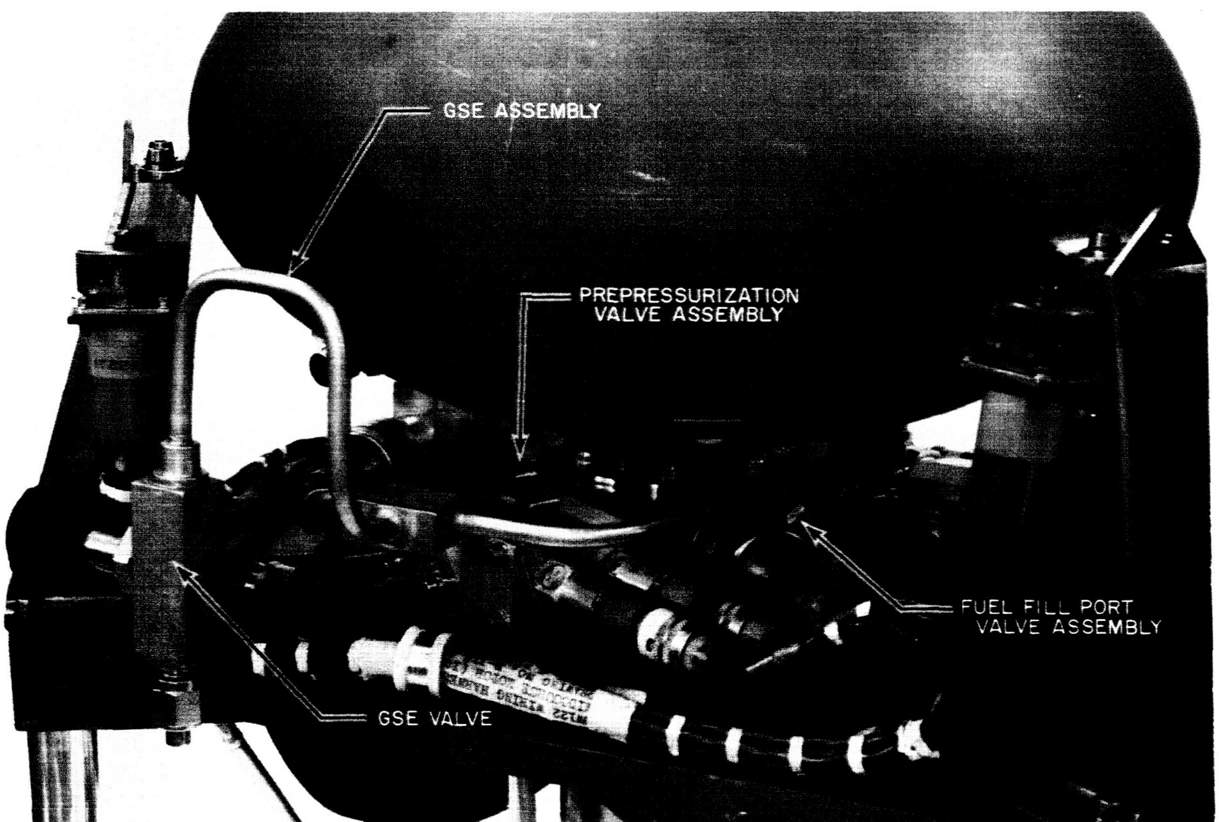


Fig. A-1. Fuel-tank leak-check assembly

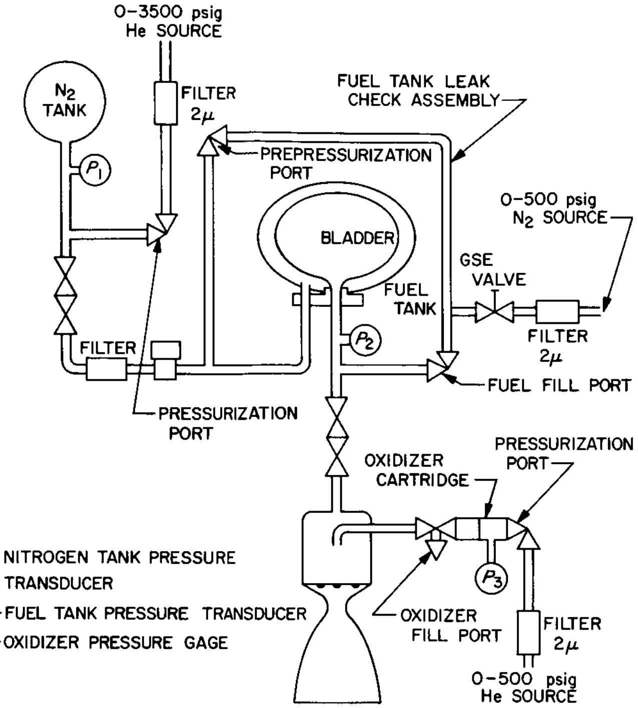


Fig. A-2. Propulsion-system leak-check schematic

may be accomplished. A 0- to 3-psig nitrogen source is connected to the fuel fill port, and a tube is attached to the prepressurization port. The bladder is pressurized to 1/4 psig and the fuel fill needle valve is closed. If no leakage is detected, the lines are removed, and the needle valves are closed and capped.

B. Oxidizer Start Cartridge

The leak check operation of the oxidizer start cartridge is initiated by removing the caps from the pressurization port and needle valve, and the oxidizer fill port and needle valve. With both needle valves open, a 0- to 500-psig helium supply is connected to the pressurization port and the nitrogen reservoir is slowly pressurized to 375 psig. A soap solution is applied to all external fittings in the oxidizer nitrogen system to verify that no leaks exist.

The pressurization needle valve is then closed, and the helium supply line is vented and removed from the port. A tube is attached to the pressurization port and a leak check is made to verify that no leakage exists across the needle valve seat. The tube is removed from the pressurization port and installed on the oxidizer fill port.

A 3-min leak check is made to verify that no leakage exists through the oxidizer bellows. The tube is removed, and a 0- to 500-psig helium supply is connected to the oxidizer fill port. The oxidizer reservoir is slowly pressurized to 375 psig. A soap solution is applied to all external fittings in the oxidizer reservoir system to verify that no leaks exist. The oxidizer fill needle valve is then closed, and the helium supply line is vented and removed.

A tube is installed on the oxidizer fill port and a leak check is made to determine that no leakage exists across the needle valve seat. The oxidizer fill needle valve is then opened and the oxidizer reservoir is vented to zero psig. The pressurization needle valve is opened and the nitrogen reservoir is also vented to zero psig. A 0- to 500-psig helium supply is connected to the oxidizer fill port and the oxidizer reservoir is slowly pressurized to 100 psig. The oxidizer fill needle valve is closed, and the helium supply line is vented and removed.

A tube is installed on the pressurization port and a leak check is made to verify that no leakage exits through the bellows. Since this leak check is performed while the bellows are slightly expanded, it may uncover any leaks that were undetected when the previous leak check was made across the bellows while they were in a compressed condition. If no leaks are detected, the oxidizer reservoir is vented to zero, and the needle valves are closed and capped.

C. Nitrogen Tank

The leak check on the nitrogen tank is begun by removing the caps from the pressurization port and needle valve. A 0- to 3500-psig helium supply is connected to the pressurization port. With the needle valve open, the nitrogen tank is slowly pressurized to 3300 psig. A soap solution is applied to all external fittings to verify that no leaks exist in the nitrogen tank system.

The needle valve is closed and the high-pressure helium line is vented and removed. A tube is installed on the pressurization port and a leak check is performed to verify that no leakage exists across the needle valve seat. If no leaks are detected, the needle valve is opened and the nitrogen tank is vented slowly to zero psig. The needle valve is then closed and the caps are replaced on the pressurization port and needle valve.

II. PROPELLANT FILLING PROCEDURE

A. Oxidizer Filling

During the oxidizer filling operation the propulsion system is weighed on a scale in order to verify that a nominal amount of oxidizer is transferred. An oxidizer vacuum fill assembly is installed on the oxidizer fill port as shown in Fig. A-3, with one line leading to the nitrogen tetroxide (N_2O_4) cylinder and the other line going to a vacuum pump. Before the N_2O_4 cylinder is connected into the system, it is pressurized with nitrogen to 100 psia and a sample is removed from the liquid port to assure that a bubble-free column of liquid is present. After a vacuum of 29 in. Hg is attained in the oxidizer reservoir, the appropriate valving is manipulated to allow N_2O_4 to flow into the cartridge under 115-psia pressure. After the cartridge is full, the N_2O_4 source is closed off and the start cartridge nitrogen reservoir is pressurized to 75 psig.

The ullage indicator latch (Fig. A-3) is released; this allows a piston to travel down a cylinder, and a calibrated amount of N_2O_4 is thus removed from the bellows. Then the oxidizer fill needle valve is closed. By removing a small portion of oxidizer from the start cartridge bellows, freedom of movement is provided for temperature expansion of the N_2O_4. If no oxidizer were removed, the end of the bellows would rest against the stop which is used for filling at ambient temperature, and any great temperature rise could cause a leak by deforming the bellows circumferentially, since they would not be free to expand in a longitudinal direction.

The vacuum fill assembly and lines are removed from the propulsion unit and strips of methyl orange indicator paper are taped to all oxidizer reservoir fittings to monitor for oxidizer leaks.

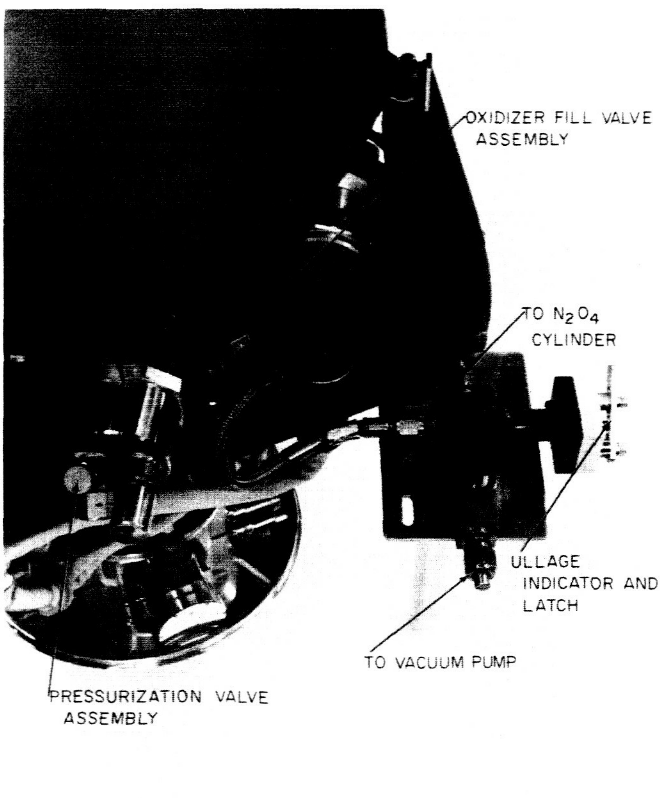


Fig. A-3. Oxidizer vacuum-fill assembly

B. Fuel Tank Bladder Passivation

Passivation is initiated by installing a standpipe in a drum of anhydrous hydrazine. Closure of the fuel drum outlet valve D, shown in Fig. A-4, is verified. A flex hose is attached to the outlet valve, and a 0- to 60-psig source of nitrogen is connected to the ullage pressurization check valve of the fuel drum standpipe assembly. The propulsion unit is weighed.

The vacuum-fuel fill assembly is installed on the fuel port. The unit is again weighed, and closure of all valves of the vacuum-fuel fill assembly is verified. After the fuel hose is attached to the vacuum fill assembly, the fuel drum is pressurized to approximately 6 psig. The fuel drum outlet valve is opened, and the fuel bleed valve C on the vacuum-fill assembly is opened until a steady stream is

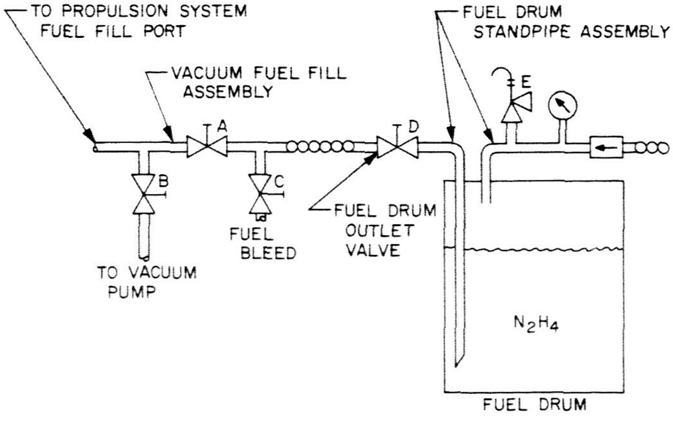


Fig. A-4. Fuel-fill schematic

obtained, at which time the bleed valve is closed. The propulsion unit is weighed.

The vacuum hose from the vacuum pump is attached to the vacuum port of the fuel fill assembly. After the fuel tank bladder has been evacuated and the vacuum has stabilized at 29 in. Hg, the vacuum is locked off and the vacuum hose is removed. The appropriate valving on the vacuum–fuel fill assembly is manipulated and 8.00 lb of fuel are allowed to flow into the bladder. After all valves have been closed, the fuel hose is removed from the vacuum–fuel fill assembly and capped.

The fuel is sloshed around gently in the fuel tank bladder at 10-min intervals for ½ hr. A water aspirator is connected to the vacuum port of the vacuum fill assembly to remove the hydrazine from the bladder. After the hydrazine has been depleted, the aspirator is removed and the line and port are capped.

Passivation is performed on the bladder as a precautionary measure to remove any catalytic agents that may be present.

C. Fuel Filling

Fueling operations are started by reconnecting the fuel hose to the vacuum–fuel fill assembly. The fuel drum outlet valve D, shown in Fig. A-4, is opened, and the fuel bleed valve C on the vacuum fill assembly is opened until a steady stream of N_2H_4 is obtained, at which time the bleed valve is closed. The propulsion unit is weighed. This weight is compared with the weight which was obtained just prior to the passivation fill, to determine the amount of residual fuel in the tank bladder. The weight of fuel which should be placed into the tank is then determined.

The hose from the vacuum pump is attached to the vacuum port of the fuel fill assembly. From this point, the actual fill operation proceeds exactly like the passivation fill. The propulsion unit is weighed after the fuel hose is removed. The vacuum–fuel fill assembly is removed from the propulsion unit and the propulsion system is again weighed. Caps are replaced on the ports and needle valves. A postfueling calibration is performed on the scale to verify its accuracy. The nitrogen source for the fuel drum is vented and removed. After lockwiring operations, the oxidizer valve blast shield is installed.

III. LEAKAGE CORRELATION

As a result of the oxidizer leak problems described in Section V-D, it was decided to investigate three commonly used leak detection methods for oxidizer start cartridge assemblies. The following leak tests were performed on these two assemblies: (1) helium mass spectrometer test with an evacuated probe, (2) helium mass spectrometer test in an evacuated bell jar, (3) commercial soap solution bubble test, and (4) methyl orange paper test after nitrogen tetroxide fill. The cartridge pressure was 375 psig for all tests. The probe test was employed only to determine the approximate order of magnitude of a leak.

The first test specimen registered a leak of approximately 2×10^{-5} std. cm^3/sec while in the evacuated bell jar of the helium mass spectrometer. The soap bubble test showed no signs of the leak for that range. However, the methyl orange paper did show traces of the oxidizer leak after the N_2O_4 was loaded into the start cartridge and pressurized. Titration of the test paper showed that approximately 10^{-6} to 10^{-7} std. cm^3/sec of oxidizer had come in contact with the paper. On the basis of this series of tests, it was also determined that the soap solution tests with a leak detection capability of only 10^{-4} std. cm^3/sec are not sensitive enough to locate very small leak paths.

The second test specimen showed a leak that measured approximately 1×10^{-7} std. cm^3/sec of helium in the evacuated bell jar. The soap solution test was negative. Lastly, the methyl orange paper displayed no color change, thus indicating that within the limits of this test there was no detectable oxidizer leak. The test results are summarized in Table A-1.

As shown above, the methyl orange paper test is more sensitive than the helium–soap bubble test. However, any leakage small enough to be undetected by the helium–soap bubble test would not result in the leakage of a sufficient amount of oxidizer to cause a propulsion system failure. If a similar midcourse system were to be built, the K-seal interfaces should be eliminated to increase the leak-tightness confidence.

Table A-1. Comparison of test methods to determine leakage

Leak test	Estimate of maximum test sensitivity std. cm^3/sec	Test results	
		Test specimen No. 1	Test specimen No. 2
1. Helium mass spectrometer (evacuated chamber)	10^{-9}	2.3×10^{-5} std. cm^3/sec	1.4×10^{-7} std. cm^3/sec
2. "Snoop" soap bubble	$\approx 1 \times 10^{-4}$	No bubble	No bubble
3. Methyl-orange detection of N_2O_4 vapors	10^{-5} to 10^{-6} (visual determination only)	Moderate color change	No color change
4. Titration of methyl-orange litmus by JPL Analytical Chemistry Laboratory		$\approx 10^{-6}$ to 10^{-7} std. cm^3/sec	No test

IV. SPECIFICATIONS AND PROCEDURES[14]

1. JPL Spec. FR 3-4-610, Functional Specification, *Ranger* Block III Flight Equipment, Midcourse Propulsion Subsystem, April 15, 1963.
2. JPL Spec. 30277, Environmental Specification, *Ranger* Block III Flight Equipment, Assembly Level Type Approval Test Procedures, February 18, 1963.
3. JPL Spec. 30278, Environmental Specification, *Ranger* Block III Flight Equipment, Assembly Level Flight Acceptance Test Procedures, September 19, 1963.
4. JPL Spec. 31260, Test Specification, *Ranger* Block III Flight Equipment, Midcourse Propulsion System Test Requirements, April 8, 1963.
5. JPL Spec. 31221, Test Specification, *Ranger* Block III Midcourse Propulsion System, Rocket Engine, February 28, 1963.
6. JPL Spec. 31255, Test Specification, *Ranger* Block III Flight Equipment Midcourse Propulsion System, Oxidizer Start Cartridge, April 2, 1963.
7. JPL Spec. 31222, Design Specification, *Ranger* Block III Midcourse Propulsion System, Oxidizer Start Cartridge Visual Pressure Gauge, March 30, 1963.
8. JPL Spec. 31268, Detail Specification, *Ranger* Block III Flight Equipment, Explosively-Actuated Valve Assembly for Hydrazine Fuel, May 26, 1964.
9. JPL Spec. 31508, Test Specification, *Ranger* Block III Flight Equipment, Explosively-Actuated Valve Assembly for Hydrazine Fuel, July 17, 1963.
10. JPL Spec. RCE-31519-TST, Test Specification, *Ranger* Block III Flight Equipment, Midcourse Propulsion System, Explosively-Actuated Valve Assembly for Gaseous Nitrogen, July 7, 1964.
11. JPL Spec. RCE-31520-TST, Environmental Test Specification, *Ranger* Block III Flight Equipment, Midcourse Propulsion System, Explosively-Actuated Valve Assembly for Nitrogen Tetroxide, June 15, 1964.

[14]These internal documents are not generally available on request without proper authorization.

12. JPL Spec. 31262, Test Specification, *Ranger* Block III, Midcourse Propulsion System Pneumatic Regulator, April 19, 1963.
13. JPL Spec. GPM-20068-GEN-A, General Specification Midcourse Propulsion Systems, Cleaning Process, December 23, 1963.
14. JPL Procedure No. 3R 108.01, *Ranger* Spacecraft Block III Midcourse Propulsion Unit Detailed Assembly, August 13, 1963.
15. JPL Procedure No. 3R 105.03, *Ranger* Spacecraft Block III Midcourse Propulsion System Leak Checking, Propellant Filling, and Pressurization Operation, January 14, 1965.
16. JPL Procedure No. 3R 103.00, *Ranger* Spacecraft Block III Belleville Spring Installation and Adjustment (Attitude Control and Propulsion Bottle Straps), June 6, 1963.
17. JPL Procedure No. 3R 107.01, *Ranger* Spacecraft Block III Alignment of Midcourse Propulsion Unit, October 22, 1963.
18. JPL Procedure No. 3R 112.00, *Ranger* Spacecraft Block III Midcourse Propulsion System Prelaunch Monitoring Operations, October 17, 1963.
19. JPL Procedure 3R 217.01, *Ranger* Spacecraft Block III Pyrotechnic Subsystem ETR Final Pre-Flight Evaluation.

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2. Grant, A. F., Jr., *Basic Factors Involved in the Design and Operation of Catalytic Monopropellant-Hydrazine Reaction Chambers*, Report No. 20-77 (CONFIDENTIAL), Jet Propulsion Laboratory, Pasadena, California, December 31, 1954.
3. Lee, D. H., Martinez, J. S. and Grant, A. F., Jr., *A Monopropellant-Hydrazine Thrust Unit for Velocity Control of Extraterrestrial Vehicles*, Memorandum No. 20-161 (CONFIDENTIAL), Jet Propulsion Laboratory, Pasadena, California, February 3, 1958.
4. Lee, D. H. and Foster, C. R., *A Monopropellant-Hydrazine Turboalternator Auxiliary Power Unit*, Publication No. 81 (CONFIDENTIAL), Jet Propulsion Laboratory, Pasadena, California, January 2, 1957.
5. Lee, D. H. and Evans, D. D., *The Development of a Heated-Hybrid Generated Gas Pressurization System for Propellant Tanks*, Technical Report No. 32-375, Jet Propulsion Laboratory, Pasadena, California, February 15, 1963.
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