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# DEVELOPMENT OF LOX/RP-1 ENGINES FOR SATURN/APOLLO LAUNCH VEHICLES

by

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# DEVELOPMENT OF LOX/RP-1 ENGINES FOR SATURN/APOLLO LAUNCH VEHICLES

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## Abstract

The development of liquid rocket engines follow similar patterns regardless of engine size. During the development of the H-1 and F-1 engines, many problems were encountered. Methods of solving the combustion instability problem are discussed. A description is given of the major components of each engine, outlining their unique features. The requirements for an insulation cocoon are discussed. Problems associated with materials substitution are provided; also highlighted is the fact that problems occur after engine deliveries and require continued development support. Safety features incorporated on the engines are mentioned. Solutions to problems encountered in flight are discussed. Upratings of both engines systems are presented graphically.

## Introduction

In the development of liquid rocket engines problems occur at several distinct intervals. The type of problem and the time phase can be predicted, but since the exact nature of the problem cannot be so readily defined, a five to seven year development program becomes a necessity.

The first problem phase occurs because of the inability to totally extrapolate and build on existing knowledge. Each liquid rocket engine design begins with knowledge gained from design and operation of earlier engines. The state-of-the-art is then pushed to improve performance, increase thrust, simplify operation or increase operational flexibility. The problems will occur in the combustion mechanics, propellant movement or in the propellant control systems. These problem areas show up in early development hardware; are often catastrophic, and require the utilization of all available resources to overcome. They occur usually as the full engine systems begin to be tested to rated requirements and duration. Extensive test programs are planned for just this eventuality. Examples of problems in this time phase on the H-1 engine (Figure 1.) were combustion instability, and thrust chamber tube splits. F-1 engine (Figure 2.) problems in this time phase were combustion instability, turbopump explosions and injector cracks.

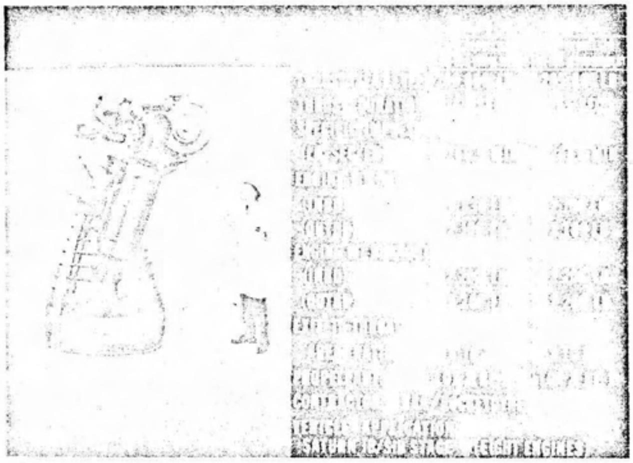


Figure 1. H-1 Engine

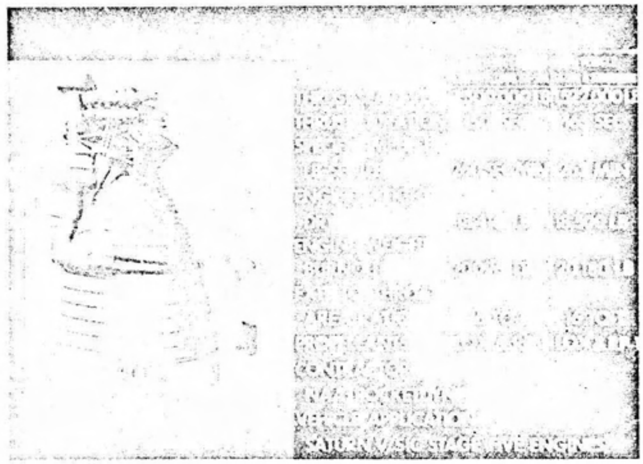


Figure 2. F-1 Engine

The second problem phase occurs after preflight rating or qualification of the basic engine design. These problems are usually caused by changes in the stage-engine interface requirements or the changes in the operational environment requirements. These problems are in part caused by the fact that engine development precedes stage development by two to three years. At the start of the engine development program, stage development or definition has not progressed to a point where all criteria can be provided. Extensive engine-stage ground test activity is therefore planned to ferret out and provide the data to overcome these problems. Early prototype or preflight rated engines are provided the stage contractors and the procuring activity to also conduct these systems tests. Examples of problems occurring in this time phase for H-1 engines were LOX dome cracking, turbine seal problems, thrust O.K. pressure switch vibrations, and turbopump gear failure; for F-1 engines gas generator cracks, turbine manifold failures, and fuel inlet duct cracks.

The third problem phase occurs as the engine begins to phase out of development into production and can be attributed to process definition, material selection, vendor selection, manufacturing tolerances, or quality control. A continuing development program is planned during this period to provide the trained personnel, facilities and hardware capabilities, to investigate these problems and prove out the required corrective effort. Examples of problems in this phase for H-1 were turbopump contamination, LOX pump seal leakage, and turbine blade material substitution. For F-1 engines the following problems occurred -- control valve failure, loose fuel pump inlet fairing bolt, loss of thrust chamber exhaust grates, and flight transducer failures.

The fourth problem phase occurs with vehicle flights due to the difficulty in duplicating completely the exact operational environment. These problems require quick response and therefore a fully staffed development capability at the engine contractor's engineering and test facilities must be maintained. Examples of these problems are -- H-1 engine turbopump gear failure. F-1 engine heat exchanger flow requirements and POGO effects investigations.

Even after early flight evaluations show the basic design concept to be correct, and the engine/stage system to be compatible, there is a need for a continuing development or sustaining engineering effort. When engine systems are tested to longer durations and more extreme limits, problems are uncovered that may have existed for a long time but were not evident until the more severe testing on a larger engine sample produced the failure mode. Subtle process changes, manufacturing improvements, or vendor changes may also result in quality or material substitution problems. These kinds of problems were encountered on H-1 in the following areas -- heat exchanger coil leaks, turbine exhaust hood cracks, stability sampling problems, and LOX seal failure. On F-1 problems occurred in the following areas -- hydraulic control line bellows cracks, various valves and turbopump leakage, and LOX seal wear ring cracks.

## Part I. H-1 Engine Development

### Description

The H-1 engine is an outgrowth of the combined knowledge gained on Redstone, Thor, Jupiter, and Atlas engines. The H-1 program was initiated with Rocketdyne Division, North American Rockwell in September 1958. (Figure 3.)

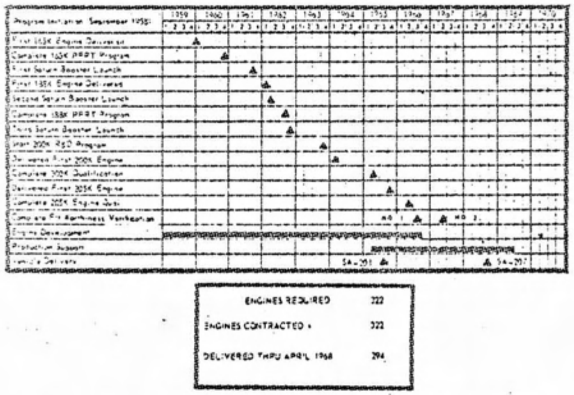



Figure 3. H-1 Engine Project Milestones

The H-1 engine is a fixed-thrust engine utilizing RP-1 and liquid oxygen as propellants. An eight-engine cluster was used for the first stage of the Saturn I vehicle which achieved ten successful flights. The uprated Saturn (IB) also uses a cluster of eight engines to obtain the desired thrust level for its first stage. (Figure 4.)

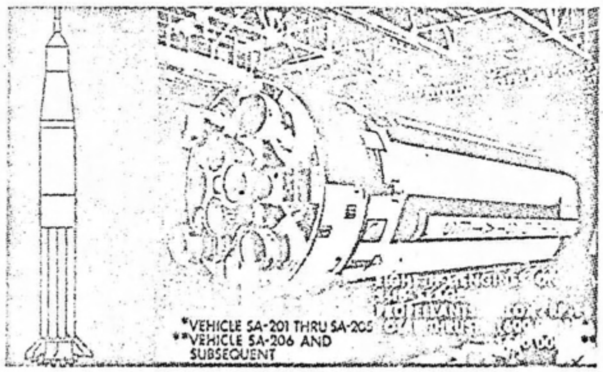
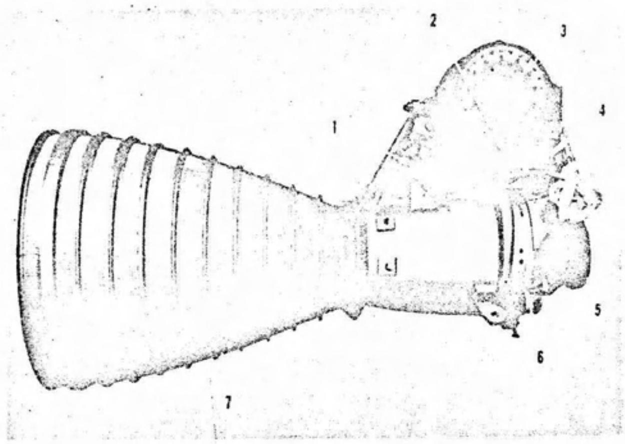



Figure 4. Uprated Saturn IB

The engine cluster is arranged with four inboard and four outboard engines, the four inboard engines are fixed and the four outboard engines are gimballed to provide first stage thrust vector control. The outboard engine turbine exhaust is ducted into collectors located at the nozzle exit plane and aspirated into the main flow of the outboard engine nozzle exhaust.

The H-1 engine features a tubular-wall, regeneratively fuel-cooled, bell-shaped thrust chamber with an expansion ratio of 8 to 1, and a single turbopump with dual pumping unit, consisting of an oxidizer pump, and fuel pump. To simplify the engine system high-pressure plumbing, the turbopump is mounted piggy-back on the side of the thrust chamber, with the main shaft at right angles to the thrust vector. This mounting provides a high pressure duct routing with minimum pressure drop, reducing the requirements for development of high pump outlet pressures. (Figure 5.)




1 FUEL ADDITIVE BLENDER UNIT
2 TURBOPUMP GEAR CASE
3 LOX PUMP
4 LOX HIGH PRESSURE DUCT
5 GIMBAL ASSEMBLY
6 LOX DOME
7 THRUST CHAMBER

Figure 5. H-1 Engine Major Components

A gas generator, using the same propellants as the thrust chamber, powers the turbopump. Ignition is accomplished by the use of a solid propellant gas generator cartridge (Figure 6) which spins the turbine, producing propellant flow and pressure, and ignites the liquid bipropellant gas generator.

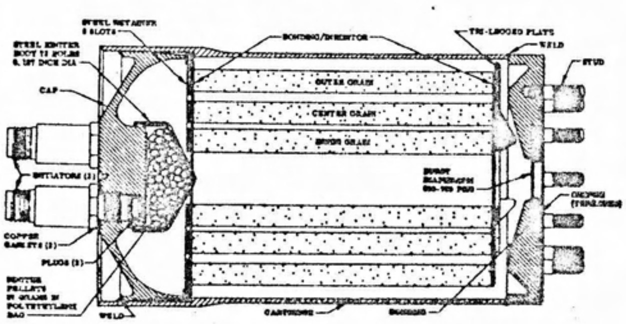



Figure 6. H-1 Solid Propellant Gas Generator

Ignition of the main chamber is accomplished by use of a hypergolic slug of triethyl-aluminum.

Control valves in the engine are actuated by fuel pressure from the main pump. Engine transients are governed by the "pressure ladder" sequence in which valves are actuated at predetermined pressures. (Figure 7.)

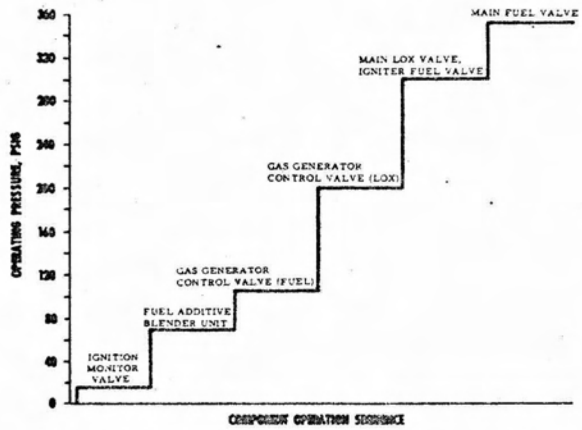



Figure 7. Pressure Ladder

A heat exchanger is located at the turbine exhaust and provides for pressurization of the vehicle oxidizer tank by passing LOX through the heat exchanger to produce gaseous oxygen.

Development Problems - H-1

The development problems for the H-1 engine are shown on Figure 8.

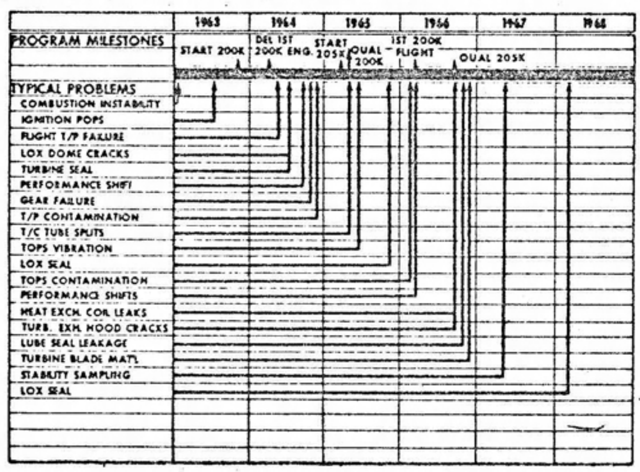



Figure 8. H-1 Development Problems

The grouping of the problems in the various time phases of the development and test programs is evident. A brief description of some of these problems and their solutions for the H-1 engine are listed below:

Combustion Instability. The technique of artificially inducing combustion instability during the starting transient on the H-1 engine was developed in late 1963. A small 50-gram bomb was installed in a special boss in the face of the injector. The bomb was constructed of a fuel-cooled cylindrical nylon case. After engine start the bomb was thermally ignited. The detonation of the bomb disturbs the flame front creating instability. The injector was designed to recover from the instability within 100 milliseconds. The Thor-Atlas injector uprated to 188K did not recover from 8 of 16 stability bomb tests. The injector was improved by addition of baffles (Figure 9.) and rearrangement of orifices until the stability was good up to thrust levels of 205K. The improved injector also provided an increase in performance of 4.2 seconds Isp.

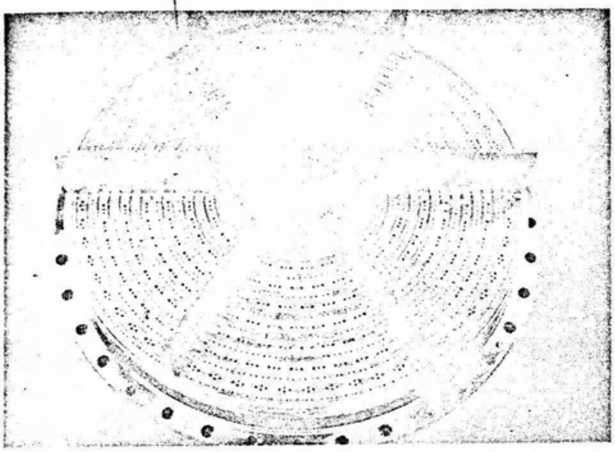


Figure 9. H-1 Injector Baffles

LOX Dome Cracks. During leak checks of engines on vehicle SA-7, at KSC, a crack was discovered in the LOX dome on one engine. (Figure 10) Stress corrosion of the aluminum alloy was determined to be the reason for failure and the domes on all eight engines had to be replaced.

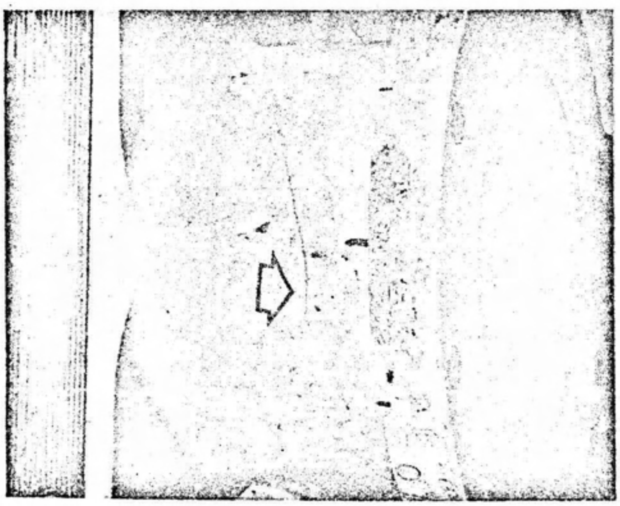


Figure 10. H-1 LOX Dome Cracks

A new dome with improved stress corrosion resistant properties was being developed under the production support program as a product improvement effort. The new dome manufacturing process was changed to allow an additional heat treat step and to shot peen the finished machined part before anodizing. (Figure 11.)

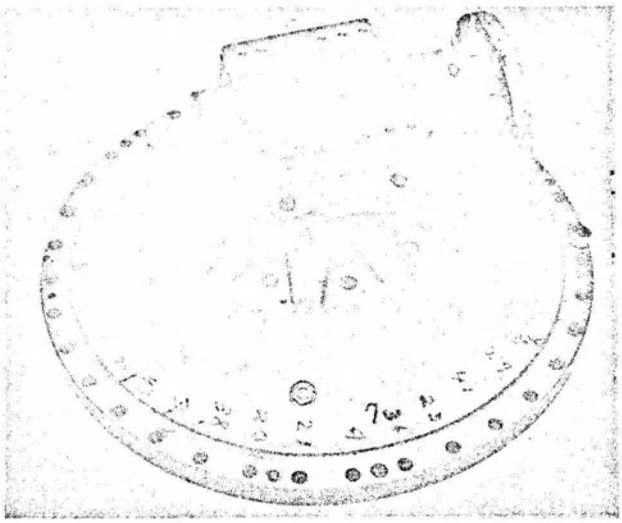


Figure 11. H-1 LOX Dome

Thrust Chamber Tube Splits. With the uprating in early 1962 from 165K to 188K, longitudinal tube splits (Figure 12.) had occurred much more frequently than had previously been experienced. Fuel spraying from tube splits near the injector face was suspected of contributing to combustion instability. Laboratory analysis of tubes from earlier incidents showed that sulphur embrittlement of the nickel alloy tubes was the cause of the ruptures. Sulphur in the kerosene fuel was found to be precipitating out and combining with the nickel alloy due to the higher temperature of the 188K thrust chamber. The problem was eliminated by changing thrust chamber material to stainless steel (347 alloy) which did not react with sulphur. No serious tube splitting problems had been encountered since this material change until the stage static testing of S-IB-1. After thorough investigation and testing of two instrumented thrust chambers, a determination was made that the problem was not similar to the earlier incident. The problem was eliminated by raising the thrust chamber prefill quantity.

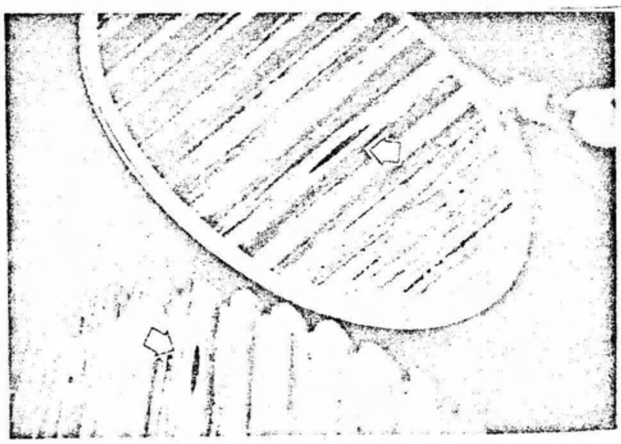


Figure 12. Thrust Chamber Tube Splits

Thrust O.K. Pressure Switch (TOPS) Vibration Problems. Although extensive testing of TOPS on normal development engine systems was accomplished, a problem with switch chattering was evidenced during an engine instability. A burst of instability during a flight would have chattered the switches and initiated a cutoff signal even though the engine itself could have self-damped and completed the mission successfully. Considerable production support effort was expended to arrive at a properly sized orifice sensing line to prevent recurrence of the problem.

LOX Pump Seal. Following two successive turbopump explosions during acceptance testing of production engines, leakage at the LOX pump seal (Figure 13.) was found to be the problem. A special pump was assembled incorporating high sensitivity instrumentation to measure pressure behind the LOX impeller, and to measure shaft axial movement and LOX seal carbon nose movement.

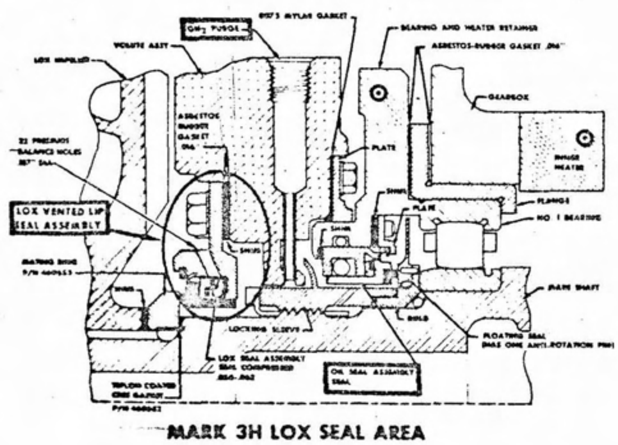



Figure 13. Mark 3H LOX Seal Area

The instrumentation showed that a momentary pressure differential occurred across the LOX seal nose piece as a result of a pressure surge occurring during the start transient. The pressure differential, occurring simultaneously with a shaft axial movement, was great enough to over balance the seal spring force and cause momentary separation of the carbon nose and mating ring. Temperature measurement of the LOX seal cavity showed a low temperature spike as the LOX seal nose moved away from the mating ring. To eliminate the pressure differential within the seal twenty-two 187-inch diameter holes were drilled in the seal housing. (Figure 14.)

The carbon ring was also redesigned to improve its strength characteristics.

Heat Exchanger Coil Leaks. After the tenth engine test of the 205K requalification program, a leak was discovered at the heat exchanger internal LOX coil support. Vibrations during engine operation caused a chaffing between the coil and support which produced a leak in approximately 1550 seconds of operation. A coil support bracket was qualified for follow-on procurements. Heat exchangers without the support brackets were limited to 1000 seconds operation prior to flight.

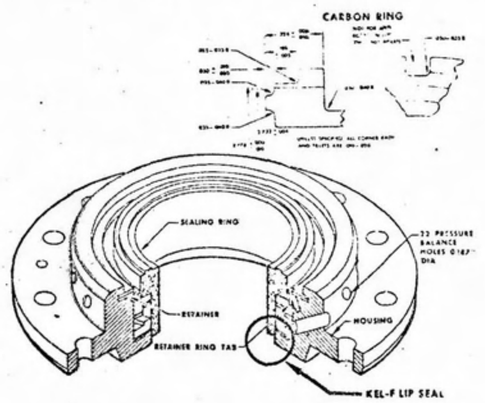



Figure 14. LOX Vented Lip Seal Assembly

Turbine Blades. Turbine blades on one H-1 engine failed during a "hot test" of the first stage of a Saturn IB. Subsequent analysis showed that the defective blades were cast from 316 stainless steel instead of Stellite 21 material. The defective blades were the result of an error in the vendor's plant. Although X-ray pictures of each blade were evaluated for flaws, weld penetration, and material differences, it could not reveal the error if all blades were of the wrong material. (Figure 15.)

Figure 15. X-Ray of Turbine Blade Wheel

After discovering the defective blades, additional efforts were made to locate other discrepant wheels. To alleviate the necessity for pulling the engines from stages and disassembling to check the wheels, an "eddy current" machine was developed by the vendor. With proper calibration, the machine could differentiate between the 316 stainless steel and the Stellite 21 material. The detailed investigation following the engine failure revealed that defective blades had been installed in ten H-1 engines. All engines known to have discrepant turbine wheel blades were removed from the stages for replacement of the turbine wheels and "hot fired" to verify performance and reliability. Inspection procedures were amplified by requiring alloy identification of each mold poured, by providing a reference standard to spot density variations in the x-ray review, providing hardness check of each blade, and expanding the metallurgical tests conducted on a percentage of each shipment received from the vendor. The damaged wheels are shown in Figure 16.

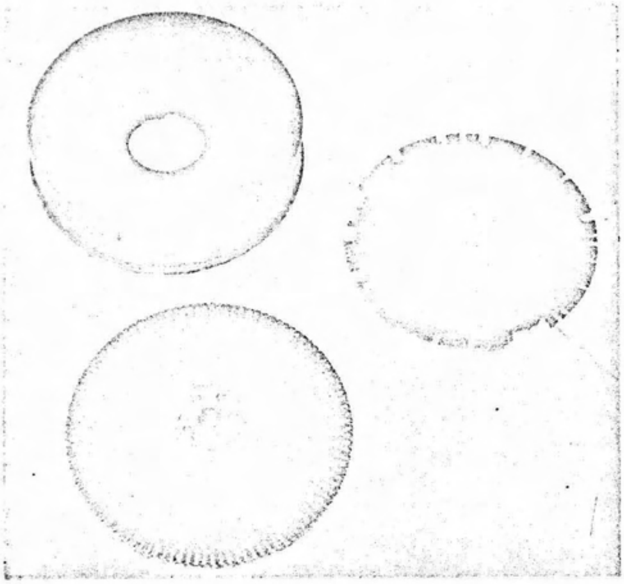


Figure 16. Damaged Turbine Wheels

Stability Sampling. An H-1 production support engine was subjected to bomb tests as a part of the continuing operational stability program. This engine had a new production configuration injector, but was otherwise built with used R&D hardware. Nine bomb tests were conducted, three of which were terminated by the rough combustion cutoff (RCC) monitor. The engine was disassembled for component inspection, and reassembled with an injector that had previously demonstrated stable operation. The reassembled engine also failed to recover from bomb induced instability. A third engine using the previously stable injector and a previously stable thrust chamber was then tested. It recovered from the first bomb test normally, but was terminated by an RCC on the next test before the bomb detonated. Tests of new production engines indicated a low frequency "chugging" that could be triggered by a 165 cps frequency in the facility fuel inlet system. The test of these new engines at the Neosho test facility showed that the rough combustion (chugging) was influenced by a resonance with the test stand supporting structure. A test program was established to determine if the engine-vehicle system exhibited sensitivity in this range or could be bombed into instability. Stability tests were conducted. One engine was bombed on the first test, and two engines were bombed on the last three tests. All tests were completely successful with damp time ranging from 3 to 27 milliseconds. No engine-vehicle resonance in the suspected range existed.

LOX Pump Shaft Seal. During the fourth of a series of special stability tests on S-IB-11, the test was prematurely terminated at three seconds by the automatic fire detection system. Evaluation of hardware and data indicated that at about 193 milliseconds after first pump rotation, a LOX leak occurred which caused the lube seal to unseat, allowing liquid or gaseous oxygen to enter the gearcase. At about 1.72 seconds after first pump rotation, detonation occurred within the gearcase which separated the lube drain manifold from the gearcase and fractured the accessory drive pad. The uncontrolled spillage and burning of combustibles initiated the cutoff at 3.27 seconds after engine ignition start signal. Damage to the vehicle was minor. This H-1 engine incorporated a vented lip type LOX seal (see Figure 14). The first stage static test was conducted for 34 seconds. Three additional static tests were conducted prior to the failure for a total of 44 seconds during which instabilities were bomb-induced on two development engines installed in the stage for that purpose. A total of seven engine tests for 322 seconds was accumulated on this engine prior to the test during which the failure occurred. Two qualified seals are available, the vented lip seal design that failed during the eighth test of the engine, and the vented bellows type seal developed in the production improvement program. (Figure 17.)

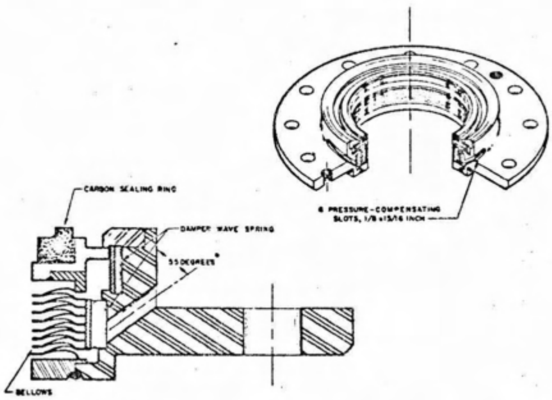



Figure 17. LOX Vented Bellows Seal

After further analysis, the best of these two designs will be utilized for the remaining flights.

The H-1 engine has achieved maturity and has accumulated an outstanding test and flight history. (Figure 18.)

Flight Problems - H-1

The H-1 engine has experienced only one problem during fourteen flights. One of the eight H-1 engines shut down prematurely in the flight of SA-6 in May 1964. Data indicated that the shutdown was caused by a power train failure somewhere between the turbine shaft and the C-pinion in the turbopump. An improved power train had been developed under the product improvement portion of production support and was already installed in H-1 engines in vehicles SA-7 and subsequent. In spite of the premature shutdown of one engine, the mission for SA-6 was successfully completed proving the design of one engine-out capability.

| NUMBER OF TESTS | | |
|---|---|---|
| SINGLE ENGINE | APRIL 1968 | TOTAL (TO DATE) |
| ROCKETDYNE | 18 | 5,795 |
| MSFC | 1 | 460 |
| TOTAL | 19 | 6,255 |
| CLUSTER (8 ENGINES) | - | |
| FLIGHT (SA-1 thru SA-10, AS-201, 203), 202, 204) | 0(0) | 14(112) |
| MSFC | 2 | 97(648) |
| TOTAL | 2 | 111(762) |
| TOTAL NO. OF TESTS | 21 | 7,017 |

| NUMBER OF SECONDS | | |
|---|---|---|
| SINGLE ENGINE | APRIL 1968 | TOTAL (TO DATE) |
| ROCKETDYNE | 1,662 | 421,026 |
| MSFC | 40 | 32,445 |
| TOTAL | 1,702 | 453,471 |
| CLUSTER (8 ENGINES) | | |
| FLIGHT (SA-1 thru SA-10, AS-201, 203) 202, 204) | 0(0) | 1,986 (15,595) |
| MSFC | 180(1440) | 5,942 (46,949) |
| TOTAL | 180(1440) | 7,928 (62,544) |
| TOTAL NO. OF SECONDS | 3,142 | 516,015 |

NOTE:
NUMBER IN PARENTHESIS IS EQUIVALENT SINGLE ENGINE TESTS

Figure 18. H-1 Engine System Test Summary

By providing a well balanced program of flight support, production support, and product improvement the NASA has been able to avail itself of the payload improvements inherent in uprating of the engine as it matures. A graphical presentation of the payload gain with uprating is shown in Figure 19.

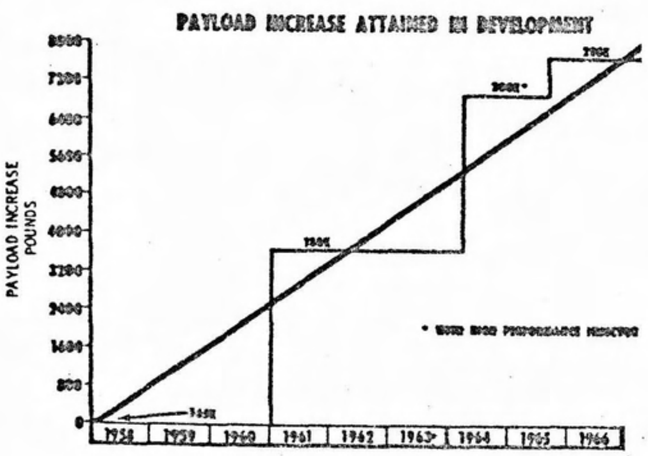
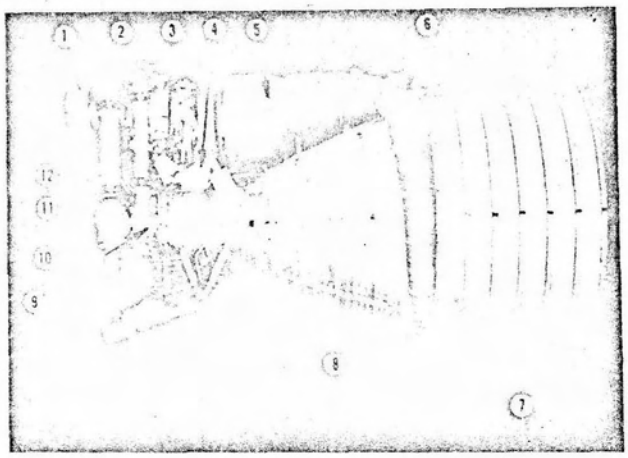



Figure 19. H-1 Payload Gain with Uprating

## Part II. F-1 Engine Development

### Description

The F-1 engine, which is capable of over 1,522,000-pounds thrust is the largest liquid engine system in the free world. Major components of the F-1 engine are shown in Figure 20.



1 INTERFACE PANEL
2 TURBOPUMP
3 GAS GENERATOR BALL VALVE
4 GAS GENERATOR
5 HEAT EXCHANGER
6 TURBINE EXHAUST MANIFOLD
7 THRUST CHAMBER EXTENSION
8 THRUST CHAMBER
9 NO. 1 MAIN FUEL VALVE
10 NO. 1 MAIN LOX VALVE
11 NO. 1 HIGH PRESSURE FUEL DUCT
12 NO. 1 HIGH PRESSURE LOX DUCT

Figure 20. F-1 Engine Major Components

Development of the F-1 engine was begun in January 1959. (Figure 21.)

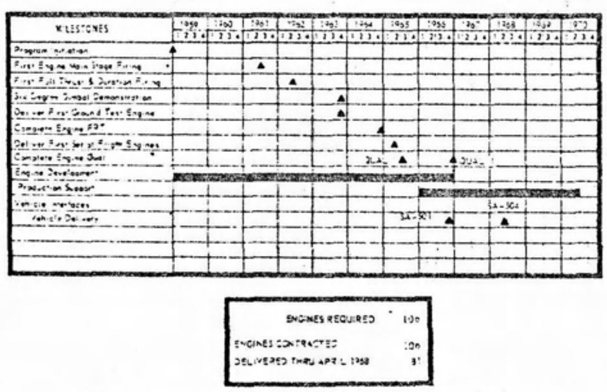



Figure 21. F-1 Engine Program Milestones

At that time, no definite vehicle application was evident and the engine development was pursued initially without the advantages of a known use. This is not unusual in engine development programs. Initial engine design reflected requirements for the engine-vehicle interface and resulted in a minimum of redesign once the vehicle was defined. A cluster of five F-1 engines is used on the first stage of the Saturn V, producing a total stage thrust of over 7,610,000 pounds. The center engine in the cluster is fixed. The four outer engines are gimballed for six degrees in each of two directions, to provide thrust vector control. (Figure 22.)

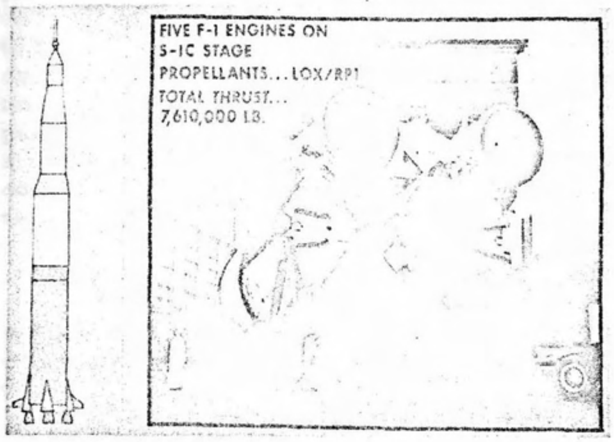



Figure 22. Five F-1 Engines on S-IC Stage

The F-1 engine is a single-start, fixed thrust gimballed system using liquid oxygen as the oxidizer. RP-1 is used as the fuel, the turbopump bearing lubricant, and the control system fluid. The engine has a regeneratively fuel-cooled thrust chamber with a turbine exhaust gas-cooled extension. Propellants are provided to the thrust chamber by a direct drive turbopump driven by exhaust gases from a gas generator also using LOX-RP-1.

The gas generator which is used to drive the turbine is initiated by a pyrotechnic charge with a tank head start; that is, no auxiliary turbine spinners, and it burns propellants at a rate equivalent to a 40,000-pound thrust engine. Main chamber ignition is achieved by the use of a hypergolic fluid contained in a canister which is pierced to ignite the main chamber for engine start. Oxidizer and fuel flow to the combustion chamber are controlled by two oxidizer valves and two fuel valves.

The primary consideration in the selection of the turbopump design was to attain reliability by using a minimum number of parts and proven design concepts. One of the most important requirements considered in the turbopump design was pump operation with a low-inlet pressure which permits the design of low pressure vehicle propellant tanks. Propellant tank pressurization is provided by a heat exchanger which utilizes the turbine exhaust to superheat the helium and liquid oxygen. The turbopump consists of back-to-back fuel and oxidizer pumps on a common turbine shaft. The thrust chamber assembly is designed to convert approximately three tons per second of liquid fuel and oxidizer into a high-velocity, high-temperature gas which provides the 1,522,000 pounds of reaction thrust. The nozzle provides an expansion ratio of 10 to 1. Approximately 30 percent of the fuel is routed directly to the injector while 70 percent of the fuel flows through the body tubes for cooling. A bolted thrust chamber extension increases the expansion ratio to 16 to 1, and is cooled by the turbine exhaust which is ducted into the main exhaust stream through slots in the shingled liner. (Figure 23.)

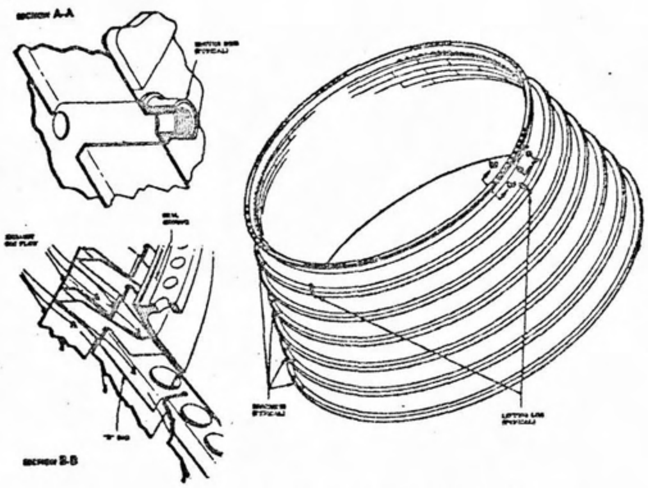


Figure 23. F-1 Nozzle Extension

The development of this bolted-on extension provides a means of testing the entire system together, yet allows the extension to be removed for ease in handling and shipping of the engine system or stages. Valve activation and gimbal actuation is accomplished with the fuel acting as the hydraulic fluid.

The development of the F-1 engine, while attempting to stay within the state-of-the-art, did, by size alone, require major facilities, test equipment, and other accomplishments which had not been attempted prior to F-1 development. (Figure 24.)

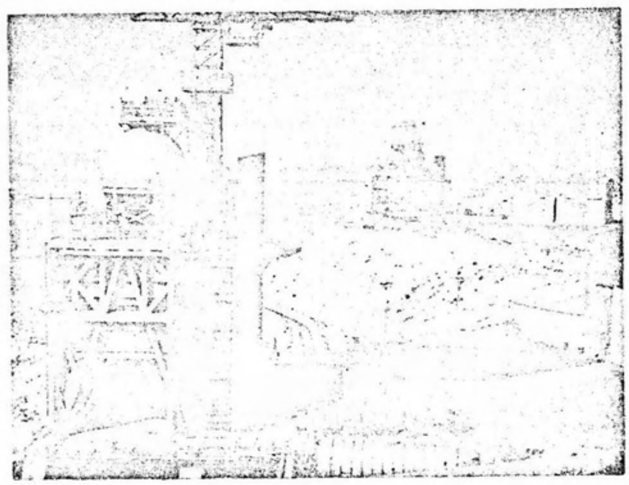


Figure 24. F-1 Engine Test Stands

Component and engine development were aimed at hardware simplification to ensure achievement of reliability requirements. Component testing proved to be an efficient and low-cost method of working out a major portion of the problems without the complexity of a full engine test. Component extended limits testing proved to be a useful tool in establishing reliability and confidence by testing beyond the rated specification. Engine extended-limits testing was also performed. Component testing was especially significant in the F-1 program because of engine cost and size, and the expense of engine system tests.

The thrust chamber stand at the Rocket Engine Test Site on Edwards Air Force Base was designed to permit repeated thrust chamber tests under varying flow rates and pressures. (Figure 25.)

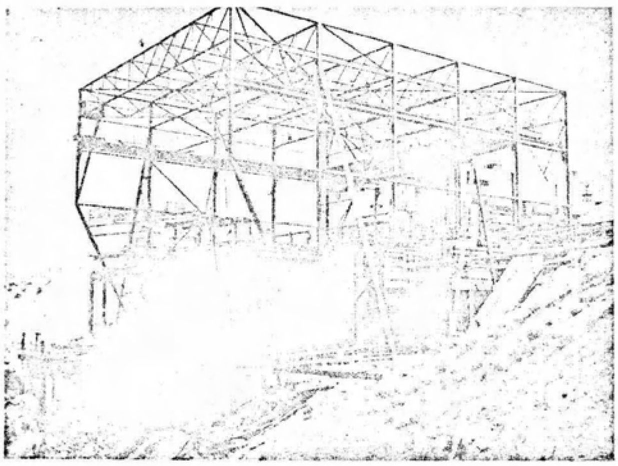


Figure 25. F-1 Thrust Chamber Stand

Injector configurations can be evaluated without risking other engine hardware because of the unique pressure-fed propellant system incorporated in the design of the stand. All resonant modes in the facility which affect the stability of the chamber and injector have been systematically eliminated.

## Development Problems F-1

The development problems for the F-1 engine program are shown in Figure 26. The grouping of the problems in the various time phases of the development and test program is evident.

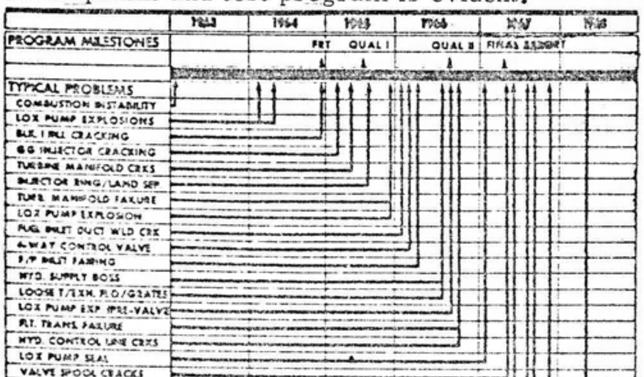



Figure 26. F-1 Development Problems

A brief description of these problems and their solutions for the F-1 engine is listed below.

Combustion Instability. During early development testing following inception of the F-1 engine, three basic injector designs evolved. These injectors were similar in design and construction to earlier H-1 engine development injectors. However, stability characteristics were notably poorer. None of the F-1 injectors exhibited dynamic stability. Where spontaneous instabilities occurred, the unstable combustion persisted until cutoff of propellant flow. Spontaneous instability occurred with flat-faced injectors and also with baffled injectors. Hardware damage was generally greater with flat-faced injectors. Major differences between the F-1 and H-1 engines with stable combustion were higher operating chamber pressures, lower contraction ratio, greater injector density and greater thrust chamber diameters. These inhibited direct scaling of the H-1 injector for F-1 application. A program was initiated to develop a dynamically stable F-1 injector and to determine the design and operational parameters fundamental to dynamically stable liquid propellant rocket engines. During the development program, several radial injector designs were tested in low-pressure, two-dimensional transparent thrust chambers, allowing high speed framing and streak movies. A few tests were made with these injector designs scaled up to full size. Combustion instability was induced by bomb tests. Many tests were also conducted in which the fuel system was pulsed with an explosively driven piston.

The preflight rating injector evolved from results of component tests (June 1963). Subsequent engine tests with the preflight rating injector revealed self-triggered or spontaneous instability had been eliminated. However, the preflight injector did not exhibit self-dampening when subjected to the bomb tests in a few instances and the basic instability frequencies were about the same as in the original injectors. The flight rating injector design resulted from additional component and engine test data by January 1965. Engine test results showed this injector was dynamically stable and with improved performance. Improvements in the present injectors were accomplished primarily by changing the LOX and fuel orifices and flow passages to optimize propellant flow distribution and injection patterns and by a slight rearrangement of the injector baffles.

LOX Pump Explosions. A total of eleven failures have occurred on the F-1 engine turbopumps. The damage resulting from a turbopump explosion is shown in Figure 27.

Two failures were non-explosive type structural failures of the LOX pump impeller. Five failures were explosive type failures during engine testing and four failures were explosions occurring on the turbopump during component testing. The two non-explosive failures occurred very early in the turbopump development program and were structural failures of the LOX pump impeller.

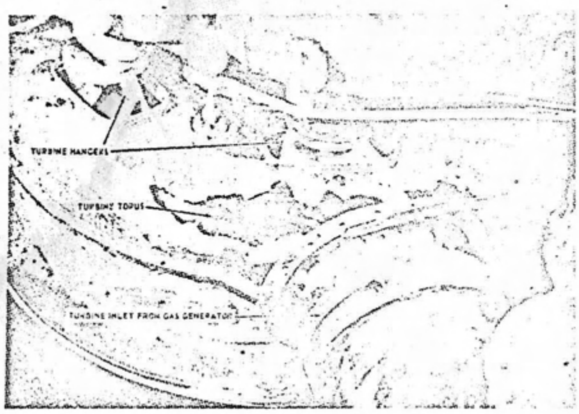


Figure 27. F-1 Turbopump Explosion

The impeller was redesigned to increase strength. The four turbopump explosive failures which occurred in component testing can be classified in two areas. Three of the failures occurred within five months and were attributed to shock loading resulting from high turbopump shaft acceleration prior to LOX loading. The fourth failure occurred during transient start resulting from the primary LOX seal rubbing a moving part. (Figure 28.)

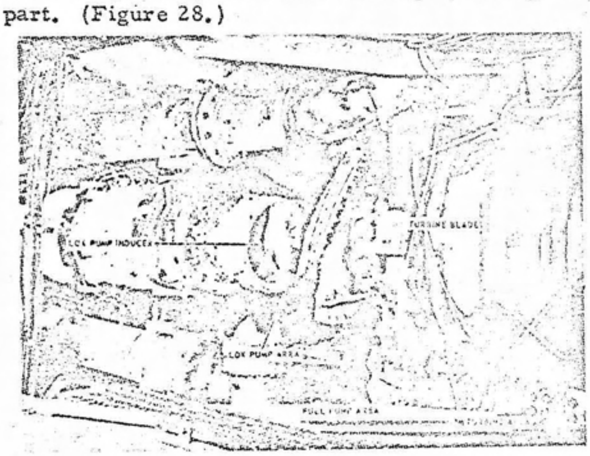


Figure 28. F-1 Turbopump Explosion

The five turbopump failures during engine testing can be classified in three areas. The first failure was similar to the last mentioned component test failure in that the failure occurred during transient start and resulted when the LOX seal rubbed a moving part. Redesign of clearances eliminated this problem. The next three failures were considered to have resulted from fatigue and fretting. This was corrected by shot peening the impeller blade edges, tightening of tolerances and moly-coating the metal surfaces at mating surfaces and by limiting the test life of the impellers. The fifth failure occurred during the first attempt of a LOX prevalve shutdown. The test was conducted to determine the feasibility of using the stage prevalve as a backup method of engine shutdown. This type of planned redundant shutdown was eliminated and a three-way electrically operated valve was developed using the override port of the hydraulic (four-way) control valve to provide the required redundant shutdown.

Turbine Manifold Cracks. The Rene' 41 material used for the F-1 engine turbine manifold was very new to industry during the early engine development. During manufacture cracks were occurring in the heat affected zone of the weld. Extensive development effort and training was required to determine the proper welding requirements. The use of automatic welding, once requirements were identified, has greatly improved this situation.

Injector Ring/Land Separation. This problem occurred after a number of engines had been delivered. It was discovered as a result of the increased test time accumulated on R&D engines. Thorough investigation of all delivered engines revealed additional cases of separation. The problem was caused by an incomplete bond between the braze material and the steel lands. The lands were gold plated to provide a better bonding surface for the braze material which solved the problem. Extensive testing was necessary both to uncover the problem and to prove the change was effective.

Flight Transducer Failures. The first 42 F-1 engines delivered contained 17 pressure transducers each, but elimination of some of the measuring requirements reduced the number to eight for each of the remaining engines to be delivered. These transducers are flush mounted by flange connection with the signal conditioning built into the same housing with the sensing element. The goal is a highly sensitive and accurate instrument with the highest reliability latest technology can provide.

Reliability under the severe environment was the most difficult problem. The most critical measurements are in the most severe environments and hence, have the highest failure rates. Two examples of severe environment are thrust chamber pressure and turbine outlet pressure. The first engines delivered experienced approximately 25% pressure transducer failure during engine testing. Failure mode was usually a capacitor or resistor and about half of the failures resulted in total data loss.

A design development program was initiated to eliminate the failures by design of a more reliable transducer. The desired result was achieved by repackaging and higher selectivity of electronic components. Figures 29 and 30 show the complexity of the transducer design.

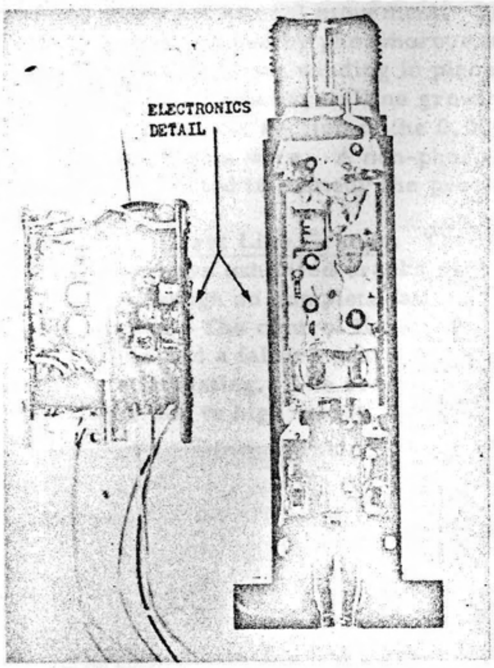



Figure 29. F-1 Pressure Transducers

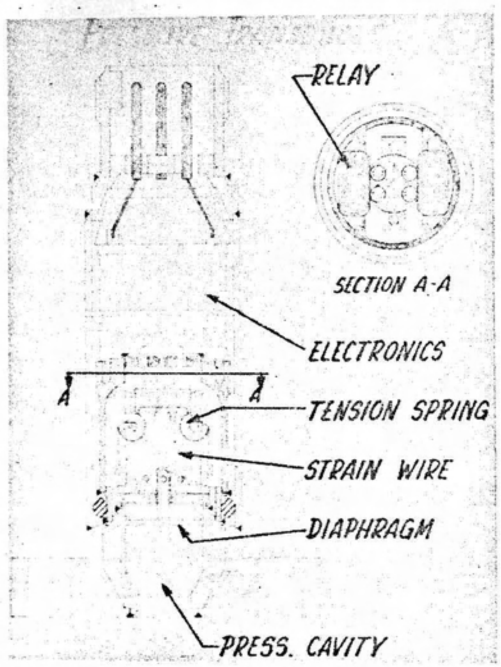



Figure 30. F-1 Pressure Transducers

There are nine temperature transducers on the present F-1 engine which are platinum wire resistance bulk type sensors. Their most frequent failure mode was sensor wire breakage.

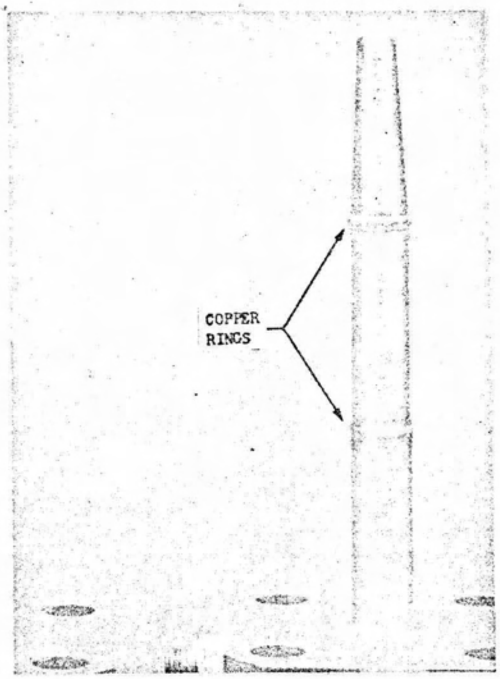



Figure 31. F-1 Temperature Transducers

Investigation revealed two reasons for failure; vibration and phosphorous contamination. The vibration problem was solved by putting two copper rings around the probe and pushing it into a stainless steel sleeve. Figure 32 shows details of the problem area.

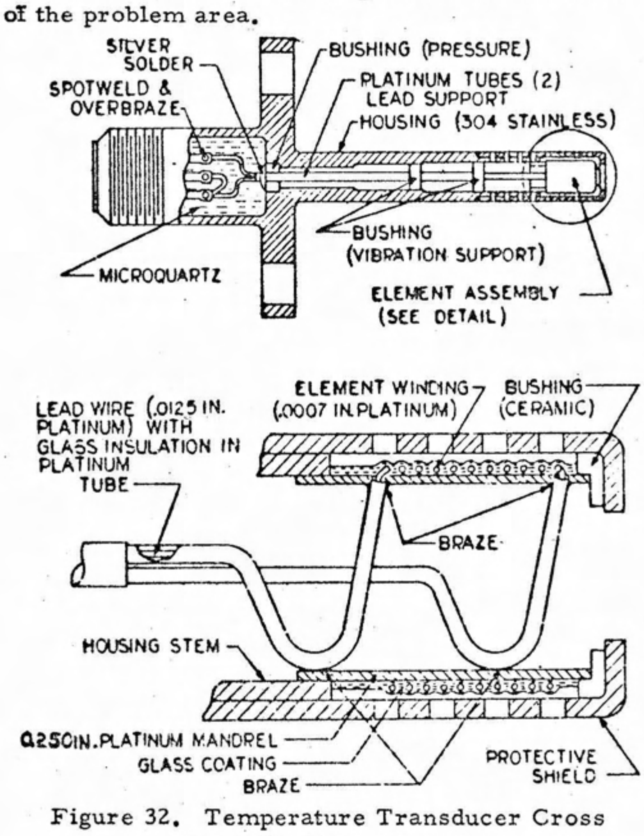



Figure 32. Temperature Transducer Cross Section

This left no room for lateral movement. The wire breakage was caused by phosphorous in the cement holding the element winding in place forming buffer layers between crystalline growths through the entire cross section of the 0.0007 inch diameter platinum wire. A non-phosphorous cement was substituted to remedy the problem.

Hydraulic Control Line Cracks. The hydraulic control lines have exhibited cracks in the weld affected area although no complete failure of a line has occurred. The control lines carry RP-1 at high pressure and a failure could result in a fire during static testing. The cracks were attributed to fatigue due to high vibration loads.

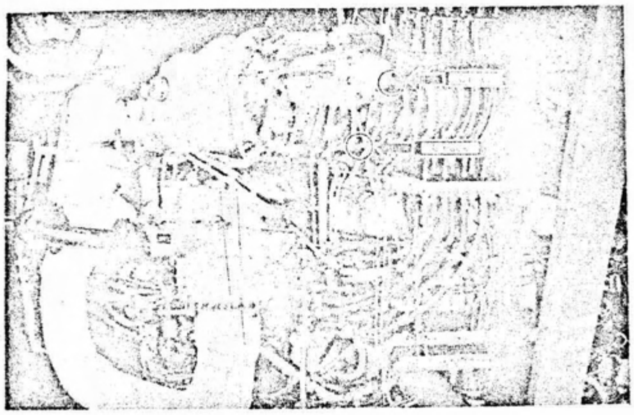


Figure 33. Hydraulic Control Lines

Support brackets with higher damping capabilities were developed and tested together with tighter line alignment tolerances. These improvements have eliminated the hydraulic control line cracking problem.

Turbopump LOX Primary Seal Wear Ring Cracking. During production support testing, the wear ring in the LOX primary seal was found cracked on two engines after 1700 and 700 seconds test time, respectively. The evidence of a cracked ring was the appearance of a sharp spike in the LOX seal cavity pressure at engine shutdown. Suspected cause of failure was overheating of the wear ring due to out of specification properties of the carbon. Comparison of photomicrographs of normal (within specification) carbon grain structure with that of the lot used to manufacture the cracked rings revealed distinct differences in grain structure. The photomicrographic examination of the cracked rings showed changes in the heat affected zones due to overheating. Through this investigation, the cracking cause was traced to only one lot of "out of spec" carbon that got into the system. The bad carbon was purged from the stock and the wear rings were removed from suspected pumps. Improved traceability, stock control, and quality controls were incorporated at the vendors and Rocketdyne plants to prevent recurrence. The crack is clearly shown in Figure 34.

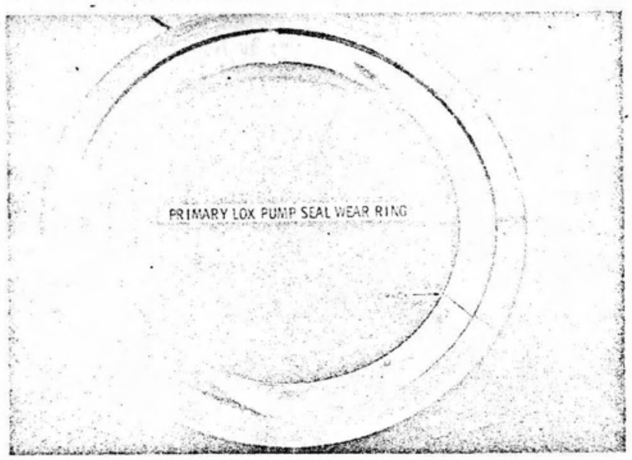



Figure 34. LOX Pump Primary Wear Ring

F-1 Thermal Insulation. During the flight of the S-IC stage, exhaust plumes from the F-1 engines expand as a function of decreasing ambient pressure resulting from increasing altitude. As these plumes expand (Figure 35.), they eventually join together forming definite pressure gradients and at some point in altitude encompass a complete plane across the plume pattern. At this point, and for the rest of the flight, there exists a differential pressure between ambient atmosphere and the joined plumes. The slower moving gases in the boundary areas of the exhaust breakoff from the mainstream and seek the lower ambient pressure of the vehicle base area. This results in a reverse flow of the hot gases over the exterior surface of the engine. Insulation is required to protect the engine from this in-flight thermal condition and from the high radiation at liftoff.

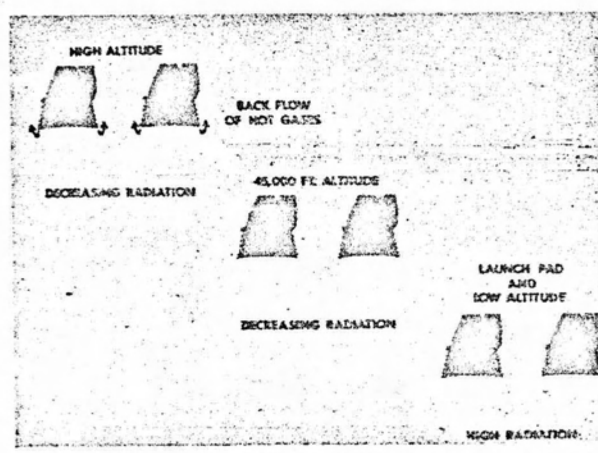



Figure 35. Exhaust Plumes

The F-1 engine has external insulation which is molded into segments and attached by brackets to the engine. When the attachments are complete, the engine is encased in a cocoon as shown in Figure 36.

Figure 36. F-1 Thermal Insulation

The development of this insulation cocoon was a challenge. The thermal insulation design was originally a stage responsibility. For each engine the weight allowed was only 150 pounds. However, because of the complexity of fit and potential interference with engine functions the engine contractor was asked to complete the development. The maximum design requirements for the thermal insulation were based on the flight environment expected for the F-1 engine. The radiation flux expected at lift off was 120 BTU/sq.ft./sec., the gas recovery temperature was 2560°F, the dynamic pressure 0.95 psi, the combined flux at altitude (radiation and gas flow) 105 BTU/sq.ft./sec., and the acoustical energy 160 db.

The original concept was a clam shell design to insulate each component valve and line individually. At this time Inconel foil .004 and .006 inch thick was established as inner and outer skins, respectively, with 1/4 inch refrasil batting as filler. The weight of the thermal insulation had grown to over 800 pounds and was increasing as engine development progressed.

A cocoon concept was adopted in order to stabilize the weight increase at just under 1200 pounds which included several hundred pounds for the cocoon bracketry and for modifications resulting from the environmental testing.

Jet engine exhaust was directed on the cocoon in a controlled test chamber to simulate flight temperatures and pressures for the development and qualification tests. Modifications, e.g., increasing inner skin from .004 to .006 inch thicknesses, addition of quilting, and adding doublers in stressed areas resulted from these tests and were verified in the qualification tests. The jet engine tests required the running of more extensive acoustic load tests. These were successfully completed at the Marshall Space Flight Center.

The temperatures measured under the cocoon in flight were well within the design limits.

Even after the successful design was completed, problems occurred at the launch site. During a driving rainstorm, the thermal insulation was soaked. While the vehicle waited on the pad, exhaustive tests were run in decompression chambers at the contractor's plant using representative water-soaked panels under expected flight heat flux and pressure conditions. These tests provided the final modifications, additional venting to accommodate any steam build-up in the thermal insulation panels which might occur during flight if moisture were present in the refrasil. The refrasil acted like a sponge in the damp Florida atmosphere. The quick response to this unexpected problem was provided by the availability of production support engineering and test effort.

The F-1 engine has accumulated considerable test time but only two flights. Test time through April 1968 is shown in Figure 37.

| NUMBER OF TESTS | APR 1968 | TOTAL (TO DATE) |
|---|---|---|
| SINGLE ENGINE | | |
| ROCKETDYNE | 24 | 2,345 |
| MSFC | 0 | 153 |
| TOTAL | 24 | 2,493 |
| CLUSTER (5 ENGINES) | | |
| MSFC | 0 | 22 (*) |
| MTF | 0 | 4 (20) |
| FLIGHT (AS-501/AS-502) | 1(5) | 2(10) |
| TOTAL | 1(5) | 28 (128) |
| TOTAL NO. OF TESTS | 29 | 2,626 |

| NUMBER OF SECONDS | APR 1968 | TOTAL (TO DATE) |
|---|---|---|
| SINGLE ENGINE | | |
| ROCKETDYNE | 2,650 | 196,728 |
| MSFC | 0 | 9,497 |
| TOTAL | 2,650 | 206,225 |
| CLUSTER (5 ENGINES) | | |
| MSFC | 0 | 1,286 (5,340) |
| MTF | 0 | 333 (1,663) |
| FLIGHT (AS-501/AS-502) | 147 (737) | 294 (1,472) |
| TOTAL | 147 (737) | 1,913 (9,475) |
| TOTAL NO. OF SECONDS | 3,387 | 215,700 |

NOTE:
NUMBER IN PARENTHESIS IS EQUIVALENT SINGLE ENGINE TESTS /TIME
*THREE OF THE 22 CLUSTER TESTS AT MSFC WHERE SINGLE TESTS FOR A TOTAL OF 22 SECONDS.

Figure 37. F-1 Engine System Test Summary

Flight Problems. The F-1 engine has performed well on these two flights to date. However, data from these flights have revealed engine/stage system anomalies that require extensive test support at the engine contractor's facilities.

On the first Saturn V flight the flow rates required by the stage for the gaseous oxygen and gaseous helium used in tank pressurization exceeded the model specification limits for the engine heat exchanger design (See Figures 38 and 39). A test program has been initiated to determine if the model specifications limits can be expanded without detrimental long term effects on heat exchanger performance.

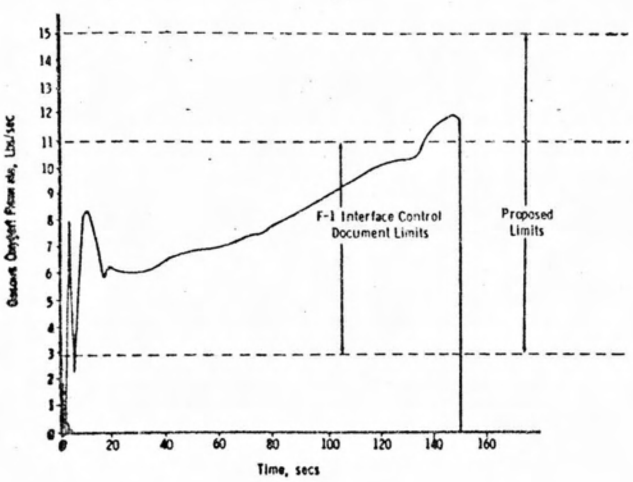



Figure 38. F-1 Engine Heat Exchanger Gaseous Oxygen Flowrate

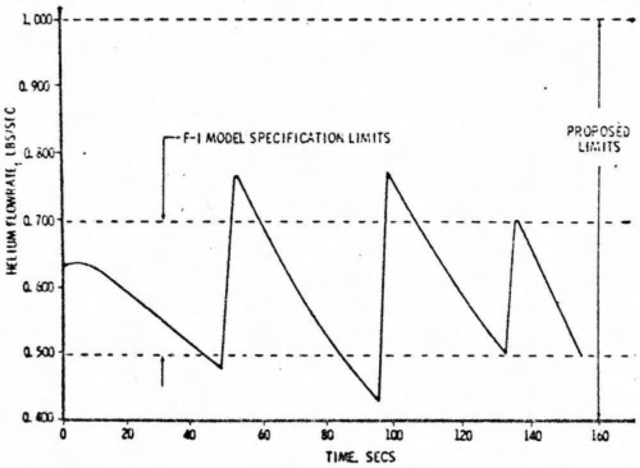



Figure 39. F-1 Engine Heat Exchanger Helium Flowrate (Typical)

POGO Investigation

Analyses of the data from the second Saturn V flight concluded that the vehicle first structural mode coupled with the engine response to the LOX suction system resonance within the 110 to 140 second flight period, resulting in a vehicle longitudinal oscillation which is termed "POGO". POGO is a self-induced longitudinal oscillation which involves the major vehicle components -- structure, fluid feedlines, turbopumps, and engines.

The description of the basic "POGO" loop can begin with consideration of the vehicle first longitudinal structural mode frequency. When this structural mode frequency approximates the vehicle suction system resonant frequency, a tuning can occur which with sufficient gain can combine to produce flow disturbances which can result in turbopump suction pressure oscillations. The suction system pressure oscillations at the engine turbopump inlets can result in perturbations, in turn, excite the vehicle structural resonances which cause further flow disturbances thus closing the loop. It is to be emphasized that the POGO phenomenon is a closed loop system interaction of vehicle structural, vehicle propellant feed system, and engine dynamics and that no single element such as the engine, the vehicle structure, the suction system or their individual components is, in itself, responsible for the occurrence of a POGO condition. (Figure 40.)

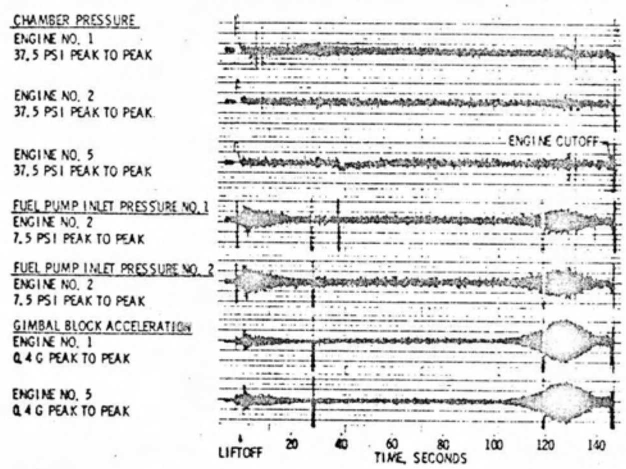



Figure 40. Flight Parameters Showing POGO Effects

The possibility of a POGO interaction was known prior to the launch of AS-501, and methods for combating the problem on the S-IC stage had been studied in 1965 through 1966. The feed system resonance is the most difficult to predict, and test data is required. A facility had been provided at Marshall Space Flight Center's Test Laboratory utilizing an F-1 turbopump and S-IC flight-type hardware from the suction duct inlets to the engine main valves. F-1 turbopump POGO testing was performed by MSFC to obtain dynamic data which could be used to identify cavitation compliance associated with the LOX and fuel pumps and to verify dynamic descriptions of these pumps.

The AS-502 flight produced coupled longitudinal oscillations of a classical POGO nature as shown on the 0-9 cps filtered data. The test program at MSFC in 1965 and 1966 included a demonstration that the feedline frequency could be shifted away from resonance. For example, by injecting small quantities of gaseous helium the oxidizer feed system natural frequency was lowered about 1.28 cps.

As a part of the engineering effort in support of the Saturn V launches, methods of eliminating POGO on future flights is being pursued. The effect on engine performance of injection of helium into the F-1 engine LOX system is being evaluated with engine tests at EFL. (Edwards Field Laboratory). Additional methods of suppressing POGO are being analyzed such as -- GOX injection at the turbopump inlet, an accumulator (prevalve) using helium or GOX, an actuator spring rate change, a mechanical damper for the LOX feed system, and other approaches. Rocketdyne has previously developed analytical models simulating the POGO phenomenon in THOR/AGENA and ATLAS/CENTAUR vehicles. This capability will be utilized to evaluate and verify the change required in the Saturn V vehicle to eliminate POGO.

The F-1 engine has been very successful and studies have been accomplished under the engine simplification portion of the flight and production support contract that indicate uprating to values as high as 1,800,000 pounds of thrust are quite feasible. The payload gains from such uprating is graphically shown in Figure 41.

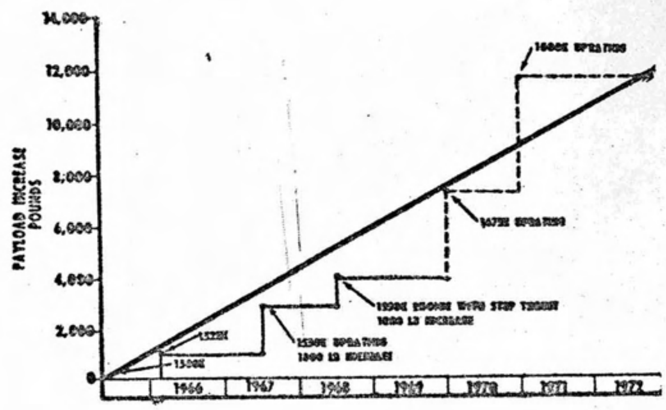



Figure 41. Payload Increase With Uprating

The need for the uprating is not evident at present, but historically the vehicle missions have required higher payloads as time progressed. The long lead required for successful engine development of a reliable and qualified engine dictate that some effort be continuously available in product improvement and that uprating be a consideration of the long range goal.

## Safety Features

Many safety features are designed into engines planned for manned flight applications. Safety features on the F-1 engine include provisions for dual gas generator igniters and dual turbine exhaust igniters. In addition an oversupply of hypergolic fluid is contained in the hypergol cartridge to insure mainstage ignition. Mechanical logic circuits are provided for valve operation to assure completion of each operation in the desired sequence. The main fuel valves are designed to remain open until commanded to close or in the event hydraulic pressure is lost.

During Qualification of the engine, malfunction tests are conducted to assure that safe shutdown will occur in the event of a malfunction prior to main chamber ignition. A redundant mode of initiating shutdown was developed for safety on the launch pad.


## Acknowledgements

The assistance of the many people who contributed to the preparation of this paper is appreciated. The author would particularly like to thank the many Saturn contractors and personnel from the laboratories at the Marshall Space Flight Center who contributed preliminary data and analyses for the flight problem portion of the paper.