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APOLLO VEHICLE PROPULSION SYSTEMS

by

L. F. BELEW, W. H. PATTERSON
and
J. W. THOMAS, JR.
NASA
Marshall Space Flight Center
Huntsville, Alabama

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APOLLO VEHICLE PROPULSION SYSTEMS

BY

Leland F. Belew; Manager, Engine Program Office, NASA, MSFC

Wayne H. Patterson; Senior Staff Engineer, Systems Engineering Office, NASA, MSFC

James W. Thomas Jr.; Senior Staff Engineer, Systems Engineering Office, NASA, MSFC

INTRODUCTION

The engines which provide the propulsion requirements for various stages of the Apollo vehicles are of a variety of different configurations, duty cycles and propellant combinations. Development of these engines reflects an extensive effort which has been highlighted with significant achievements; some of which have advanced the state-of-art in propulsion systems. Applications of the engines require that they achieve a higher degree of maturity prior to flight than engine development programs with which we have been familiar. This is, of course, due to the man-rating requirement and high costs for Apollo vehicles. During recent months, we have seen the H-1 and RL10 engines demonstrate flight capabilities. The other engines which we will be discussing are in the latter stages of development and are rapidly approaching flight dates. Apollo engines which will be discussed are the H-1, F-1, J-2, RL10, the Apollo service module propulsion engine, and the lunar excursion module descent and ascent engines. The salient features of these engines and some of the highlights of the development programs are reviewed.

Development of the Apollo engines has produced an extensive test bed which can and should prove fruitful to many areas of industry. Such problems as stress corrosion under certain environments, brazing and welding techniques, special sealing techniques, special test equipment, protective coatings for materials, and many other areas offer improvements and refinements which are applicable in many areas of our technological society. This information is being made available to industry through the NASA Technology Utilization Program.

H-1 ENGINE

In comparison to present-day rocket engine systems, the Thor and Jupiter engines are fairly rudimentary propulsion systems. However, these systems laid the groundwork for our present-day liquid rocket engines; in particular for the H-1 and F-1. The H-1 engine evolved from five different engine system designs (the Thor, Jupiter, X-1, S-4, and the Atlas). The H-1 program was initiated with Rocketdyne in September 1958. Originally designed for a sea-level thrust of 188,000 pounds, the first H-1 engine rating tests were conducted at 165,000-pounds thrust. The first four Saturn I flights were flown with engines at this rating. Beginning with the fifth flight, the engine has been flown at 188,000-pounds thrust. More recently, based on successful testing at a higher-thrust level, a decision was made to uprate the engine to 200,000-pounds thrust.

The H-1 rocket engine (Figure 1) is a fixed-thrust engine utilizing RP-1 and liquid oxygen as propellants. Eight engines are clustered to obtain the desired thrust level for the first stage of both the Saturn I and Saturn IB vehicles as shown in Figures 2 and 3. 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. Turbine exhaust of each inboard engine is ducted outside the vehicle envelope and vented overboard. 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. 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 which spins the turbine, producing propellant flow and pressure, and ignites the liquid bi-propellant gas generator. Ignition of the main chamber is accomplished by use of a hypergolic slug of triethylaluminum. An H-1 engine schematic is shown in Figure 4.

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 pre-determined pressures. A heat exchanger is located in the turbine exhaust and provides for pressurization of the vehicle oxidizer tank by passing LOX through the heat exchanger to produce gaseous oxygen. Gaseous nitrogen is contained in a separate container and is used for pressurization of the vehicle fuel tank.

Significant program developments within the H-1 program are shown in Figure 5. During the H-1 development, four types of problems were encountered: combustion

oscillations, pump gears and bearings, thrust chamber cracking, and LOX dome cracking. Combustion oscillations, occuring particularly at higher thrust, have been corrected by use of baffled injectors. The pump troubles were corrected by increasing volute and gear case strength, beefing-up gears and strengthening bearings. Thrust chamber tube cracking was attributed to sulphur embrittlement of the nickel tubes; the tubes have been changed to stainless steel and now have a useful life of about 3,600 seconds. Liquid oxygen dome cracks occurred on an engine installed in a flight vehicle. The problem was attributed to stress corrosion of the aluminum alloy when exposed to certain ambient weather conditions. A different aluminum alloy with an improved manufacturing sequence involving the forging process, heat treatment, machining and surface treatment alleviated the problem.

There has been one failure of an engine in eight flights, resulting in a premature shutdown of that engine. The failure was traced to a weak gear in the turbopump gear case. This failure caused no problem in the flight of the Saturn vehicle since its control system is designed with an engine-out capability. This problem had been isolated and appropriate action had been initiated several months prior to the flight failure and resulted in a change which was implemented on the next flight vehicle.

The H-1 Engine Development Plan is directed toward the further development of an H-1 engine rated at 200,000-pounds thrust. Primary elements of the continuing development program are:

1. Thrust chamber and other combustion devices
2. Component qualification
3. Engine system qualification
4. Turbopump development
5. Controls development
6. Reliability
7. Product improvement

The 165,000-pounds thrust H-1 engine PFRT (preliminary flight rating test) Program was performed on three production engines and was completed November 1960, with an accumulated total of 34 tests for 2,507 seconds of mainstage duration. Engine assemblies were selected at random from the production line for PFRT and have satisfactorily completed the normal hot-firing acceptance testing. Test objectives were as follows:

1. Calibration/performance tests

2. Safety limits/malfunction tests
3. Drainage tests

PFRT program performance ratings were 165,000 pounds, nominal thrust; 540 psia, nominal chamber pressure, 2.27 O/F, engine mixture ratio, and 165 seconds, rated duration.

The 188,000-pounds thrust H-1 engine PFRT test program consisted of two engines and was completed December 1962. Test objectives were essentially the same as for the 165,000-pounds thrust engine. PFRT program performance ratings were 188,000-pounds, nominal thrust, 598 psia, nominal chamber pressure; 2.26 O/F, engine mixture ratio; and 165 seconds, rated duration.

Manned application requirements have placed very stringent reliability goals upon the H-1 engine system. These goals are demonstrated by defining a mature program and systematically carrying it through the R&D (Research & Development) Program, which includes PFRT, and qualification phases. Engine qualification program was completed April 7, 1965, with an accumulated total of 4,452 seconds of mainstage operation. Since the inception of the program in September 1958, a total of 5,299 single-engine tests have been conducted for a total testing of 384,231 seconds. Also, 63 cluster firings have been conducted for a total of 4,997 seconds.

NASA is presently undertaking various studies to increase the payload capability of Saturn vehicles. The H-1 engine system is currently under study to determine uprating limitations with present hardware. Limits tests are being conducted to determine strain and vibration levels of major components, thrust chamber pressure profiles and wall temperatures, and engine reliability. Preliminary data and information obtained from post-test inspections indicate the uprating can be accomplished with relatively minor changes.

F-1 ENGINE

The F-1 engine, which is capable of over 1,500,000-pounds thrust, is the largest liquid engine system in the free world. Development of the F-1 engine was begun in January 1959. 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 relfected 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 will be used on the first stage of the Saturn V, producing a total stage thrust of over 7,500,000 pounds (Figure 6). 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.

Original study work preceding the F-1 development, and dating back to 1955, was performed under Air Force cognizance. Responsibility for the development of a high-thrust engine was assigned to NASA in 1958, and required close liaison with the Air Force during the transition to assure full advantage of the previous study work. With the conclusion of the studies indicating that an F-1 class engine was feasible, a development contract was awarded to Rocketdyne.

When the enormous size of the engine is considered, the F-1 engine has achieved an impressive list of accomplishments. Figure 7 shows the F-1 engine and some of its features. A summary of the major milestones in the development of the F-1 is presented in Figure 8. One year after R&D contract initiation, full-scale components were undergoing tests and after 27 months, complete engine system testing had begun. The F-1 program has now developed through the flight-rating test phase, which was completed in December 1964. Engine qualification for manned application is scheduled for late in 1966. Component and engine system testing is extensive with a total of almost 1,500 engine system tests planned.

Component and engine development have been aimed at hardware simplification to insure achievement of reliability requirements. Component testing has proven 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 is proving a useful tool in establishing reliability and confidence by testing beyond the rated specification. Engine extended-limits testing is also being performed. Component testing is especially significant in the F-1 program because of engine size and cost, and the expense of engine system tests.

An F-1 engine schematic is shown in Figure 9. The gas generator which is used to drive the turbine is initiated by a pyrotechnic charge with a tank head start; i. e., no auxiliary turbine spinners, and 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, is controlled by two oxidizer valves and two fuel valves. A separate valve is used to admit propellants to the gas generator. 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. This, of course, 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 3 tons per second of liquid fuel and oxidizer into a high-velocity, high-temperature gas which provides the 1,500,000 pounds of reaction thrust. The nozzle provides an expansion ratio of 16 to 1. The thrust chamber and nozzle, out to an expansion ratio of 10 to 1, are regeneratively cooled by the fuel. The remaining portion of the nozzle extension, to an area ratio of 16, is cooled by the turbine exhaust which is ducted into the main exhaust stream through slots in the shingled liner. Valve activation and gimbal actuation is accomplished with the fuel acting as the hydraulic fluid. Thrust vector control for the S-IC stage is achieved by gimbaling the four outboard engines with the single inboard engine remaining fixed.

The F-1 engine has external insulation which is molded into segments and attached by brackets to the engine. When the attachments are completed, the engine is incased in a cocoon. A jet engine exhaust has been impinged onto this insulation to verify both its insulating and dynamic load capability.

Problem areas have not been unknown to the F-1. Pump problems and combustion instability problems were encountered during the early phases of development. Since these have been reported extensively in the past, we will only state here that these have been overcome through systematic investigation and the engine has demonstrated satisfactorily through the flight rating test phase of the program. Testing is now routine, with efforts directed toward meeting qualification requirements. Extensive test time has been accumulated in the last several months. Total test time accumulated from June 1964 to March 1965 was greater than the total test time for the 5 1/2-year program prior to June 1964. Engine verification testing presently being performed at the Marshall Space Flight Center in Huntsville is demonstrating the relatively trouble-free operation of the F-1 at present development maturity.

A limited analytical study has been performed to determine the uprating potential of the F-1 engine. This study encompassed the region from the present 1,500,000-pounds thrust level to an upper limit of 1,800,000. At the upper limit, there will be several areas of hardware modification necessary. Presently, there is no further work planned in this area since there is no requirement for an uprated F-1 at this time.

The F-1 production is presently on schedule. In spite of difficulties in the development program, engine production deliveries have remained consistant with Saturn V requirements. To date, no schedule impacts have resulted from engine deliveries and further problems are not anticipated.

RL10 ENGINE

The RL10 engine has evolved into a sophisticated and versatile engine system which is capable of a relatively long cyclic life, restart capability, and can be modified to be throttled in the range of 10 to 100-percent thrust. Original work on the RL10 was begun in October 1958 by Pratt & Whitney Aircraft Corporation. The engine was originally designed for Centaur stage use and later adapted for the Saturn I program. To date, a total of approximately 145 engines have been delivered. The engine has performed satisfactorily on two Atlas/Centaur flights and on four Saturn flights. Original development was undertaken by the Air Force, to develop a high-performance space engine capable of restart.

In the Atlas/Centaur application, two engines are utilized to produce a total stage thrust of 30,000 pounds. In the S-IV stage of the Saturn I vehicle, six engines are clustered to provide a total thrust of 90,000 pounds. The Centaur and the S-IV stages are shown in Figures 10 and 11, respectively. These two applications have proven in flight the feasibility of a liquid hydrogen/liquid oxygen stage and propulsion system. A vehicle design is now underway to adapt the Centaur stage for use as a third stage on the Saturn IB A brief summary of some of the RL10 characteristics is shown in Figure 12. Major chronological milestones of the development program are presented in Figure 13. The successful engine firing in August 1959 is noteworthy because of the short development time and state-of-the-art of liquid hydrogen knowledge at that time. In addition to the design and fabrication of the engine, it was necessary to design, fabricate and checkout the altitude ejector-diffuser system required to test the RL10. This type of facility was required because of the RL10 high-expansion ratio (40 to 1) and relatively low-chamber pressure.

The RL10 engine uses a regeneratively-cooled thrust chamber and a turbopump-fed propellant system. A schematic of the engine is shown in Figure 14. The turbopump assembly consists of: (1) a two-stage centrifugal liquid hydrogen pump that is driven by a two-stage hydrogen turbine mounted on a common shaft, and (2) a single stage centrifugal liquid oxygen pump that is mounted side-by-side with the hydrogen pump and is driven by a gear train from the main shaft. The fuel (hydrogen) flow path is from the

vehicle fuel tank through the fuel inlet shutoff valve, fuel pump, thrust chamber cooling tubes, turbine, main fuel shutoff valve, injector, and into the combustion chamber. The fuel is utilized to cool the thrust chamber tubes and the heat-energy picked up drives the turbine. This so-called "topping" cycle provides a performance gain of approximately 1/2 to 1 percent over that of a conventional gas generator type cycle. The oxidizer (oxygen) flow path is directly from the oxidizer tank through the oxidizer inlet shutoff valve, oxidizer pump, oxidizer flow control valve, injector and into the combustion chamber.

Engine refinement has been through development and continuing product improvement effort to its present state of maturity. The improved performance RL10 has an increased area ratio, increased chamber pressure, modified turbopump, and minor injector and thrust control changes. The engine is designed to fit into the same envelope as the present RL10 but with an area ratio of 57 to 1. The increase in chamber pressure, required to get the increased area ratio into the same envelope as the present RL10, is obtained by improving the turbopump efficiency and by reducing pressure losses. Major development plan milestones for the improved RL10 program are shown in Figure 15.

Present status and maturity of the RL10 can best be shown by reviewing the testing to date. Figure 16 presents a brief summary of some of the engine statistics as of March 1965. The number of firings and total time accumulated on the RL10 program exemplifies the maturity of the program. The maximum duration of 1,680 seconds represents over 3.5 times the required thrust duration of 470 seconds. This philosophy of "limits" testing has proven successful in developing an engine with a high reliability and a high degree of confidence. Engines have operated over two hours and for 50 to 70 firings without maintenance or parts replacement. This is equivalent to 10 round trips to the moon. The demonstration of 30-percent excess thrust, a high degree of throttling, low idle characteristics, instant start capability, and the adaptability of the engine to utilize fluorine has shown the RL10 to be a highly flexible, as well as a highly reliable engine.

Additional technology efforts are being pursued on the RL10 Engine Program. The use of trioxygen difluoride (O_3F_2) as an additive to liquid oxygen to provide hypergolic ignition has been demonstrated. The use of fluorine as an oxidizer has also been demonstrated successfully. This was accomplished with a minimum of engine changes. The changes required were primarily in the area of oxidizer seals with minor gold plating required for material compatibility. Uprating of the RL10 is feasible and demonstration firings have been made with the 15,000-pound thrust engine up to a thrust level of 20,000 pounds.

Test results have demonstrated that a modified RL10 with a fixed-area injector is capable of operating over a 10 to 1 thrust range. Figure 17 shows a simplified schematic of the variable-thrust engine. As can be seen from the schematic, the throttling principle is relatively simple. The oxidizer flow and the turbine bypass flow are controlled through a linkage system. This provides the necessary pump speed and mass-flow control to produce the 10 to 1 throttling capability. It would be too much to expect such a system to be demonstrated without any difficulties. Problems such as feed system instability were encountered and reasonable solutions obtained. Further work in this area is required as well as in the areas of controls development and performance predictions.

The use of the main propulsion system to provide low-thrust levels for propellant settling, mid-course corrections, and burning of vehicle tank residuals offers a potential improvement in the overall vehicle performance. This use of the main propulsion system would require low-idle thrust capability. The RL10 idle feasibility program demonstrated the capability for extended low-idle operation as well as the ability to accelerate from low-idle to rated thrust. The feasibility of operation with gaseous and mixed phase as well as liquid propellants has been demonstrated. Engines have been tested with engine inlet pressures as low as 25 psia on the fuel side and 30 psia on the oxidizer side with satisfactory results. The engine was operated as a pressure-fed system under these conditions. This was accomplished by by-passing the turbopump and operating the engine with tank pressurization only. Figure 18 presents a comparison of the total impulse possible as a function of run duration and thrust level.

Instant engine start has also been demonstrated. In normal operation, a 20-second period is required to cool the pumps to a point where there is assurance that liquid hydrogen will be available at the pump impeller. Instant start has been achieved in feasibility tests by coating the internal wetted area of the hydrogen pump with Kel-F insulation and by increasing the oxidizer pump inlet pressure and flow rate. The problem of RL10 pump cooldown centers around eliminating pump cavitation caused by localized gassing. Schemes such as recirculation systems, wet pump systems, and submerged systems offer promise. Further work on the insulated turbopump for instant start will include evaluation at lower values of engine inlet pressures.

In summary, the RL10 has developed into one of our most reliable and flexible engine systems. Engine development has advanced the state-of-the-art in hydrogen engine technology and has provided an operational system as well as a functional test-bed for experimental R&D investigations.

J-2 ENGINE

Prior to October 1958, research by government agencies and the rocket engine industry had proven the feasibility of liquid hydrogen as a rocket engine fuel. Vehicle studies indicated an engine utilizing liquid oxygen and liquid hydrogen as the propellants with a thrust level of 200,000 pounds would be required for advanced space vehicles. Subsequently, the development program for the J-2 engine was initiated with Rocketdyne on September 1960.

The J-2 engine (Figure 19) is a 200,000-pound vacuum thrust, high-performance, multiple-restart engine utilizing liquid hydrogen and liquid oxygen as propellants and is designed to be used singularly or clustered. Engine application is on the S-IVB stage of the Saturn IB and Saturn V vehicles, which is a single engine installation as shown in Figure 20. The engine is also used in the S-II stage of the Saturn V vehicle, which is a five-engine cluster installation as shown in Figure 21.

The J-2 features a single tubular-wall, regeneratively-cooled, bell-shaped thrust chamber with an expansion ratio of 27.5 to 1 and two independently driven turbopumps; the fuel pump is a seven stage axial-flow machine and the oxygen pump is a more conventional centrifugal machine. Both turbopumps are powered in series by a single gas generator, which utilizes the same propellants as the thrust chamber. Multiple restart capability is provided by rechargeable, engine-mounted, hydrogen-start tank and a spark-ignition system. Welded joints are used to minimize leaks and to improve reliability. Dual seals incorporating an intermediate bleed from the low-pressure side are utilized at all separable hot gas and propellant connections. A heat exchanger, located in the oxidizer turbopump turbine exhaust duct, provides for pressurization of the vehicle oxidizer tank. For stage oxidizer tank pressurization, liquid oxygen flows through the heat exchanger and is expanded for the S-II stage and helium gas is expanded for the S-IVB stage. Gaseous hydrogen from the thrust chamber fuel manifold is utilized for fuel tank pressurization. Propellant utilization is accomplished by by-passing liquid oxygen from the discharge side of the oxidizer turbopump to the inlet side through a servomoter-driven valve. A J-2 engine schematic is shown in Figure 22.

Significant program developments within the J-2 Engine Program are shown in Figure 23. The first engine system test (2.57-seconds ignition test) was accomplished 18 months after the contract award. This was an uncooled thrust chamber with turbopumps, driven from a gaseous hydrogen facility rather than the gas generator. The first 250-second duration test was accomplished in October 1962 with the first 550-second test being accomplished in November 1963. The course of the J-2 development has not

been without incidents. The first engine tests exhibited side loads when operating at sea level conditions. Since the gimbal actuators were designed to take the normal thrust load, some means had to be found to contain these side loads. This was accomplished by modifying the test stands with the addition of side load restraining arm to physically hold the engine during testing.

Fuel-pump stall was a development problem early in the program. The fuel pump flow enters the regenerative cooled jacket before entering the injector. As the fuel pump delivers the first fuel to the (relatively) warm chamber, a considerable volume of hydrogen gas is created. This gas cannot pass through the injector at a rate sufficient to keep up with the flow. The answer has been to prechill the pump and chamber and to limit the temperature conditions under which a start will be attempted.

Another problem area has been with the start sequence involving the sequencing of the main LOX valve opening, allowing an excess amount of LOX to be passed on to the gas generator. This caused excessive temperature and pressure spikes in the gas generator. The problem was corrected by changing the sequencing of the main LOX valve to aid in regulating the flow of LOX to the gas generator. This has also helped to alleviate the previously discussed fuel-pump stall problem.

At the present time, the R&D program has progressed through the testing of a flight rating test version of the J-2 engine and fabrication and testing of the first flight-type R&D systems. The primary elements of the continuing development program are:

1. Engine system development
2. Thrust chamber and injector development
3. Gas generator development
4. Ignition system development
5. Gas and fuel turbopump development
6. Controls development
7. Inter-connect components development
8. Ground support equipment development

The Pre-Flight Rating Test Program consisted of one engine and was completed November 1964, with an accumulated total of 16 tests for 2,350 seonds of mainstage duration. The engine assembly selected had satisfactorily completed acceptance testing prior to use for the pre-flight rating test. The test objectives constituted a program which included:

1. Safety limits tests
2. Malfunctions tests

3. Performance tests
4. Oxidizer and fuel pump inlet-condition tests

The Pre-Flight Rating Test Program performance ratings are 200,000 pounds, nominal thrust; 632 psia, nominal chamber pressure; and engine mixture ratio of 5 to 1 (oxidizer to fuel ratio); and 500 seconds, rated duration.

The Flight Rating Test Engine Program will consist of one engine and is scheduled to begin May 1965 and be completed June 1965, with an accumulated total of 30 tests for 2,750 seconds of mainstage duration. The engine assembly selected will have satisfactorily completed acceptance testing prior to use for the flight rating test. Flight rating test objectives are as follows:

1. Safety limits tests
2. Malfunctions tests
3. Performance tests
4. Restart tests
5. NPSH demonstration

The flight rating test program performance ratings are the same as the pre-flight rating test program, with the exception of specific impulse which is increased.

The J-2 engine qualification program is scheduled to be completed December 31, 1965. The qualification demonstration will have 3,750 seconds of mainstage operation. Since the inception of the program in September 1960, a total of 999 single-engine tests have been conducted for a total duration of 56,463 seconds.

Demonstration of the J-2 engine as an acceptable item for manned application is being accomplished by a systematic test program. As a part of this test program, a major emphasis is placed on limits testing as a means of demonstrating reliability and confidence without a prohibitively large test sample. Limits testing offers us a judicious method of establishing confidence to meet the end goal of a successful flight and still allows us the control of cost and development time.

An analytical and experimental program to evaluate advanced component designs and systems operating concepts, which are applicable to the J-2 propulsion system, is being conducted. Principal engine features which differ from the present J-2 engine will be:

1. Tapoff turbine drive with the attendant simplified control system
2. Catalytic thrust chamber ignition
3. Minimize chilldown requirements

As part of the various studies to increase the payload capability of the Saturn

vehicles, J-2 engine uprating has been investigated. These studies have shown that the engine can be uprated sufficiently to obtain appreciable payload gains.

APOLLO SPACECRAFT ENGINES

Engines discussed up to now are used in the launch vehicle stages and are designed to perform only a portion of the mission requirement. There are also three main propulsion systems contained within the spacecraft portion of the vehicle. These are the service module propulsion engine and the lunar excursion module (LEM) descent and ascent engines. There are major differences in the launch vehicle propulsion systems and in the spacecraft propulsion systems. These differences are, of course, related to the particular requirements, mission and environments of the stage. All three of the spacecraft propulsion systems are pressure-fed bi-propellant systems, utilizing storeable hypergolic propellants and have redundant control valves. The propellants used are nitrogen tetroxide and a 50/50-percent blend of unsymmetrical dimethyl hydrazine and anhydrous hydrazine.

SERVICE MODULE ENGINE

The Apollo service module engine is a pressure-fed system which develops a nominal vacuum thrust of 21, 900 pounds. The engine is utilized in several operating modes and is capable of multiple starts and a relatively long service life. A single engine (Figure 24) is provided on the Apollo service module. The engine (Figure 25) has been under development since 1962 and is presently undergoing extensive altitude simulation testing.

Various missions for the service module require that the engine be utilized in four operating modes. The system normally provides the required propulsion for mid-course velocity corrections, during the translunar and transearth phases of the lunar mission and is employed to achieve lunar orbit injection and ejection. The system is also utilized to achieve changes in the lunar orbit plane. The present lunar mission concept does not require a final retrograde maneuver from earth orbit during the return trip; however, the system supplies the necessary deceleration capability during retrograde from the initial earth orbit missions.

In addition to the four main operating modes, the service module engine can be utilized for two important emergency operating conditions. If an abort maneuver is required after the solid propellant launch escape system has been jettisoned, (jettisoning of the launch escape system occurs during second stage boost), then the service module

propulsion system can be utilized to achieve separation from the launch vehicle. In addition, the service module engine may be employed for recovery of a lunar excursion module which has been disabled after achieving lunar orbit, but prior to accomplishing rendezvous with the command and service modules.

Development of the service module engine was begun in April 1962, by the Aerojet-General Corporation. The engine is gimbal-mounted with a throat-mounted gimbal ring and utilizes an ablative thrust chamber with a removable radiation-cooled nozzle extension. Propellant valve redundancy is achieved by the use of a series-parallel arrangement of fuel and oxidizer valves. The valve configuration allows continued engine operation with any one set of linked fuel and oxidizer valves failed-closed and also prevents unscheduled depletion of propellants with any one set of valves failed-open. A redundant electrically-operated flow control valve is installed in the oxidizer supply line to provide a means of compensating for variations in engine mixture ratio.

Due to the high expansion ratio and low chamber pressure, complete engine system testing must be performed under simulated altitude conditions. At the time of this writing, the service module engine was undergoing pre-qualification testing at Arnold Engineering Development Center in Tullahoma, Tennessee. Testing to date in the pre-qual program has been successful. These tests included gimbaling and mixture ratio shifts and durations up to 180 seconds. Extensive component, subscale, and fullscale engine testing has been conducted to get us to the present pre-qual program. Some of the major milestones in the program are shown in Figures 26 and 27. The next few months should see us through the qualification program and we will then be operating on a routine basis with engine acceptance and deliveries.

DESCENT ENGINE

The engine system required for the lunar descent and landing maneuver has to be capable of multiple starts, variable thrust and thrust vector control. The engine must provide the propulsion to decelerate from lunar orbit for mid-course correction during descent to the lunar surface, descent thrusting and hover to landing. In addition, the engine is capable of placing the LEM (Figure 28) stage back into lunar orbit, if during a portion of the landing sequence, it is determined that it is unsafe to continue the landing mission.

Development of the descent engine was begun in June 1963, by the Space Technology Laboratory. The LEM descent engine is pressure-fed and operates at a chamber pressure of 110 psia. The expansion ratio is 47 to 1 with ablative thrust chamber and nozzle

section out to an expansion ratio of 16 to 1. The nozzle extension from the 16 to 1 out to the 47 to 1 area ratio is radiation-cooled. The nozzle extension is flange-mounted to the thrust chamber (Figure 29), and designed to crush, should it contact the lunar surface upon landing.

A schematic of the engine system is presented in Figure 30. Fuel and oxidizer are introduced to the engine through flixible inlet lines located near the gimbal ring at the engine throat and then directly into the flow control valves. After flowing through the venturis, the propellants pass to a series-parallel shut-off valve assembly and then to the variable area injector. The shut-off valves are fuel pressure-actuated. The fuel is introduced to the valve actuators through solenoid actuated pilot valves. The series-parallel redundancy in the valve arrangement provides for positive start and cut-off.

Throttling requirement for the LEM descent engine was probably the single most stringent requirement for the engine. To assure high performance and stable combustion of all thrust settings, it was necessary to maintain uniform injector patterns over the entire range of propellant flow rates. Initially, two parallel approaches were undertaken to insure the success of the program. One approach employed the injection of an inert gas (helium) into the propellant manifolds to sustain propellant injection velocities. A second method uses a mechanically throttled engine which has a variable area injector. Both approaches were tested and evaluated, and the mechanically throttled engine selected for development.

The mechanical throttling scheme utilizes variable area, cavitating venturi flow-control valves in addition to the variable area injector. This allows separate propellant flow control and propellant injection functions. Each function may then be optimized without compromising the other and insures that propellant flow rates are made insensitive to downstream pressure variations in the injector and combustion chamber. The variable area coaxial injector performs the metering function down to a 70-percent thrust level. At this point, cavitation commences in the valve throats, and the valve then functions as a cavitating venturi. Once cavitation begins, the propellant metering function is removed from the injector. Flow is then controlled entirely by the cavitating venturi valves. The engine throttling capability extends over a 10 to 1 thrust range.

The injector for the LEM descent engine is somewhat unique. It has a movable sleeve; oxidizer flows through the center of the sleeve and is sprayed into the chamber through ports whose area increase with sleeve motion for maximum thrust and decrease for minimum thrust. The orifice for the fuel is an annular opening between the sleeve and injector face. This annular orifice is contoured so that as the sleeve moves upward

(away from the injector tip), the fuel passages enlarge. This action permits increased propellant flow, while maintaining optimum velocity and impingement angles. The movable injector is linked directly to the metering pins of the venturi valves and this linkage is connected to a single actuator. Thrust is thus regulated by actuator movement through control of both the injector and the cavitating venturi.

Some of the major milestone achievements are shown in Figures 31 and 32. As is apparent, the program is progressing at a fast pace. The program plan calls for extensive testing during the remainder of this year and in 1966 and qualification testing in mid 1966. The development is progressing reasonably well and no major difficulties are expected.

LEM ASCENT ENGINE

One of the most critical maneuvers in the complex lunar landing mission is the lift off from the lunar surface and injection into lunar orbit. This is the one phase of the mission where there is no back-up mode provided. The LEM ascent engine (Figure 33) is required to provide the propulsive forces necessary to lift the LEM, less the landing stage, from the lunar surface to a lunar orbit and to provide the gross orbit adjustments required for rendezvous with the Apollo command and service modules. The required reliability of this engine system is considered to be the most significant of all the Apollo engines.

Development of LEM ascent engine was awarded to Bell Aerosystems Company. The LEM ascent propulsion system is a pressure-fed constant thrust, bi-propellant rocket engine with a restart capability. The propellants are N_2O_4 and 50/50-percent blend of N_2H_4 and UDMH, which are the same as those used for the service module and the descent engines. The engine is not required to provide thrust vector control. The lack of complexity of the engine allows for a more inherently reliable design. The relative simplicity of the control system and the design redundancy of the control valves further enhance the goal of achieving a highly reliable engine.

Control of the propellant flow to the engine combustion chamber is achieved by the valve package, trim orifice and injector assemblies. The valve package consists of a redundant series-parallel ball valve arrangement. The ball valves are arranged in fuel/oxidizer pairs as can be seen in Figure 34. Each pair is simultaneously opened and closed on a common crankshaft by an actuator that uses fuel as the actuating medium. The actuating fuel flow is controlled through three-way solenoid valves. Failure of any

one set of interlinked fuel/oxidizer valves in either the open or closed position still allows for the safe completion of the mission.

The trim orifices are provided to adjust the thrust level and mixture ratio of the engine by trimming out the pressure drop of the propellant feed section. This "fine tuning" is accomplished in static test on each engine.

The propellants are supplied from the main inlet lines and are divided on both the oxidizer and fuel sides. The flow is then through the series valves, trim orifices, and then through the injector in the combustion chamber. The injector is a fixed orifice design with the orifices arranged in a radial pattern on the injector face and drilled in triplets. Near the outer periphery of the injector face, a doublet pattern is used. This helps to provide a temperature barrier near the ablative chamber wall with a lower temperature, fuel rich combustion zone.

The engine utilizes an ablative thrust chamber and nozzle. The expansion ratio of 45.6 to 1 is designed for an operating chamber pressure of 120 psia. The relatively high expansion ratio and low chamber pressure yield a large engine for the thrust which is obtained.

The development program on the LEM ascent engine is presently undergoing redesign to incorporate a baffled injector and increased burn time. Qualification is expected to begin March 1966.

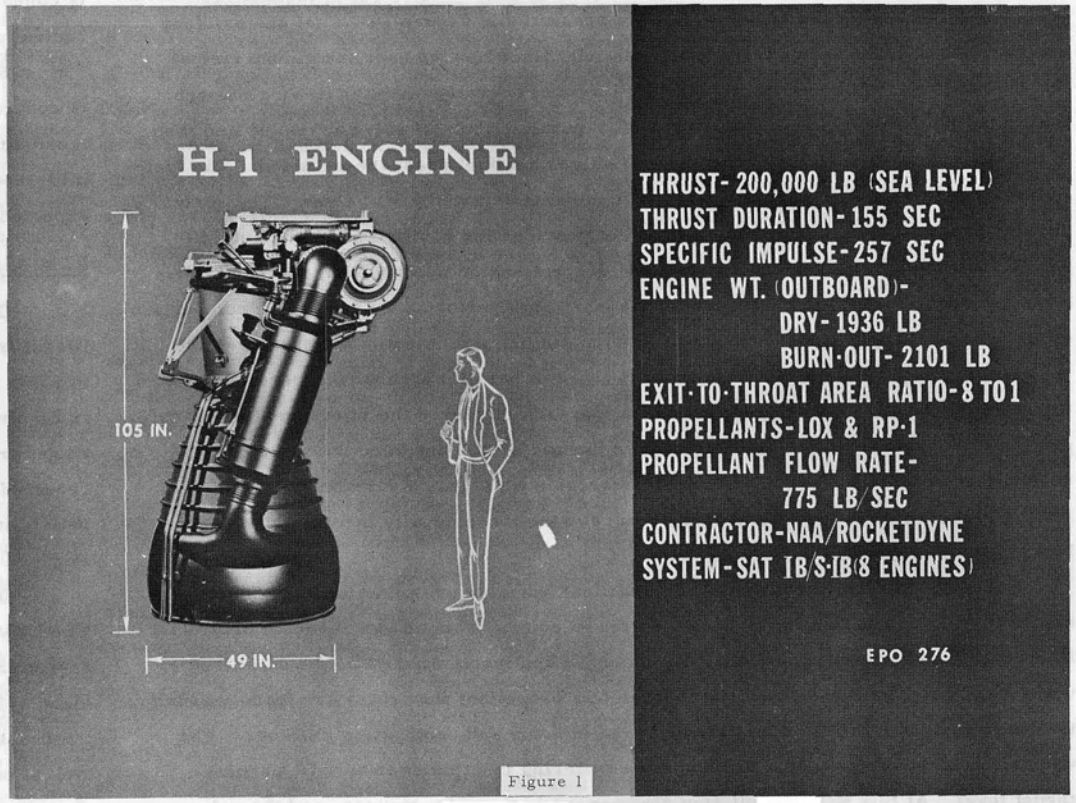
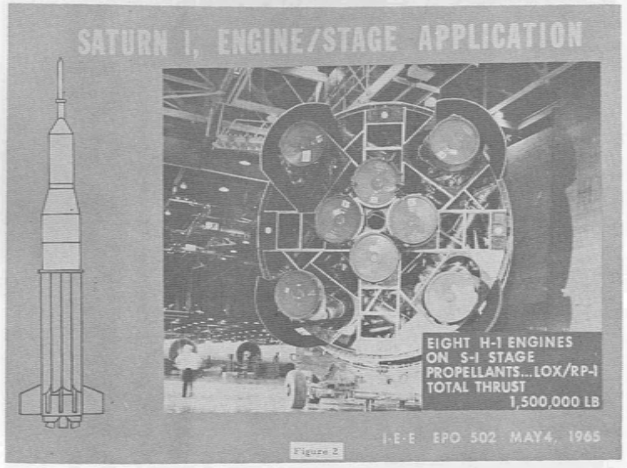
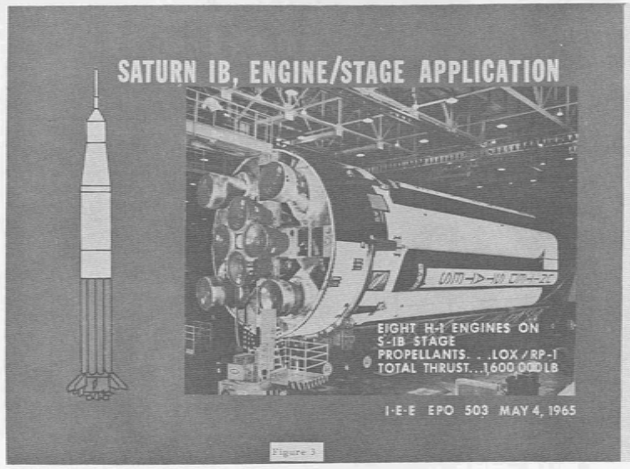
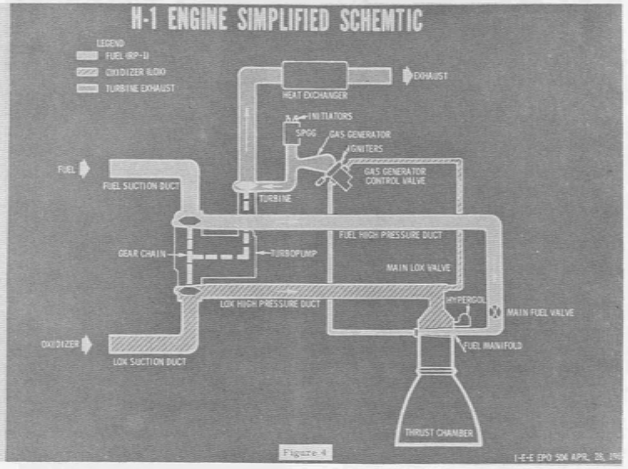
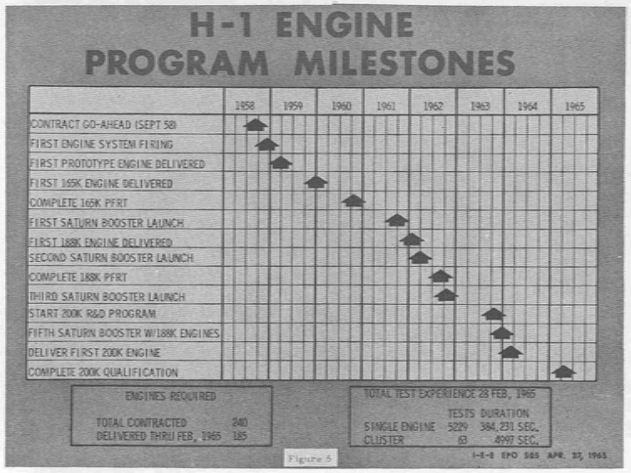


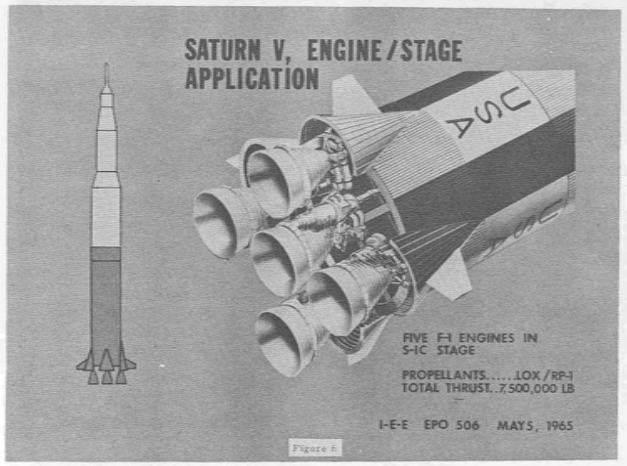
Figure 1

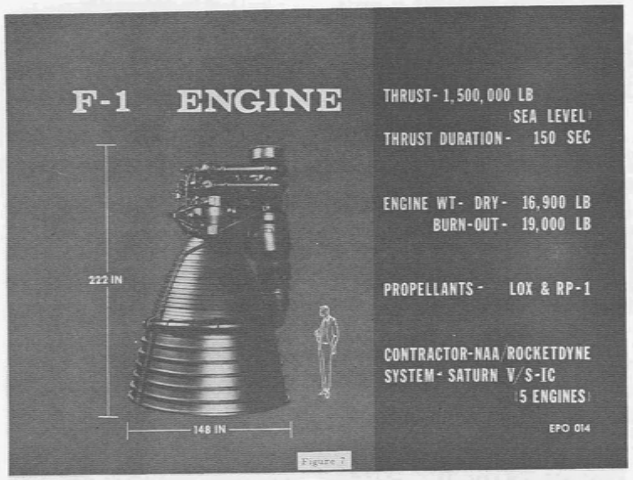

SATURN I, ENGINE/STAGE APPLICATION
EIGHT H-1 ENGINES ON S-I STAGE
PROPELLANTS...LOX/RP-1
TOTAL THRUST 1,500,000 LB
I-E-E EPO 502 MAY4, 1965
Figure 2

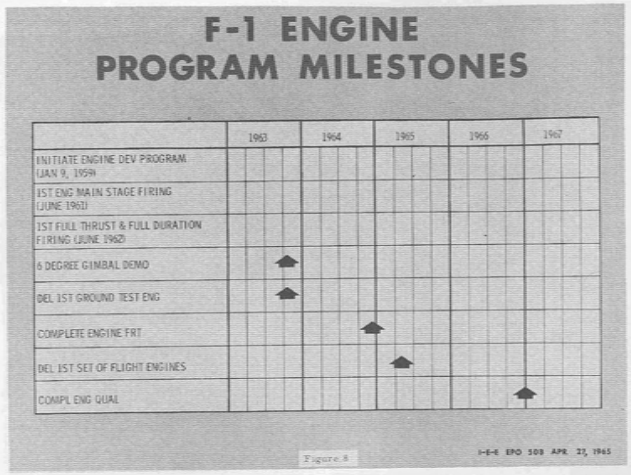

SATURN IB, ENGINE/STAGE APPLICATION
EIGHT H-1 ENGINES ON S-IB STAGE
PROPELLANTS. . .LOX/RP-1
TOTAL THRUST...1,600,000 LB
I-E-E EPO 503 MAY 4, 1965
Figure 3

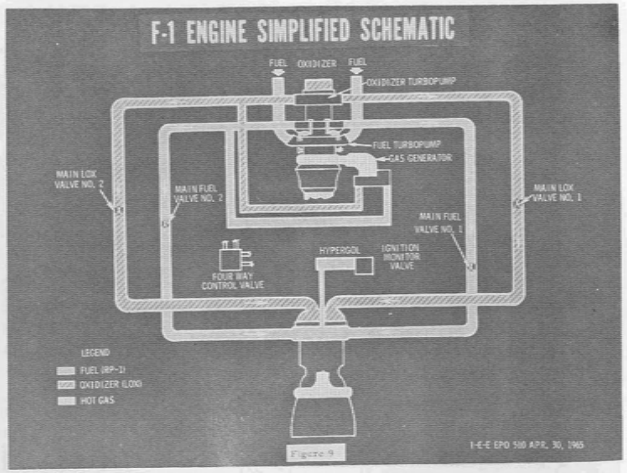

H-1 ENGINE SIMPLIFIED SCHEMTIC
LEGEND
FUEL (RP-1)
OXIDIZER (LOX)
TURBINE EXHAUST
HEAT EXCHANGER
EXHAUST
INITIATORS
SPGG
GAS GENERATOR
IGNITERS
GAS GENERATOR CONTROL VALVE
FUEL
FUEL SUCTION DUCT
TURBINE
FUEL HIGH PRESSURE DUCT
GEAR CHAIN
TURBOPUMP
MAIN LOX VALVE
LOX HIGH PRESSURE DUCT
HYPERGOL
MAIN FUEL VALVE
OXIDIZER
FUEL MANIFOLD
LOX SUCTION DUCT
THRUST CHAMBER
Figure 4

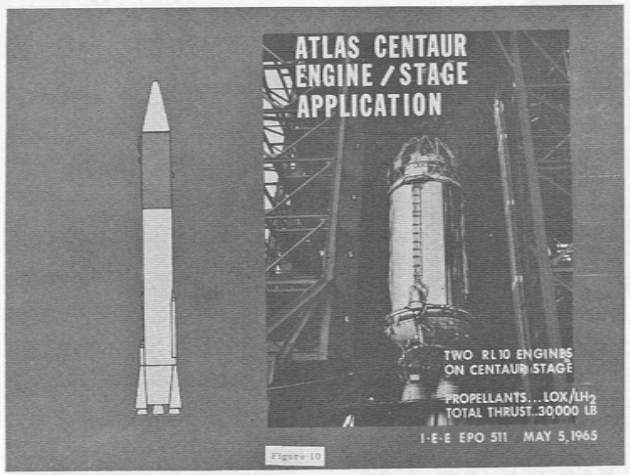

H-1 ENGINE PROGRAM MILESTONES
1958
1959
1960
1961
1962
1963
1964
1965
CONTRACT GO-AHEAD (SEPT 58)
FIRST ENGINE SYSTEM FIRING
FIRST PROTOTYPE ENGINE DELIVERED
FIRST 165K ENGINE DELIVERED
COMPLETE 165K PFRT
FIRST SATURN BOOSTER LAUNCH
FIRST 188K ENGINE DELIVERED
SECOND SATURN BOOSTER LAUNCH
COMPLETE 188K PFRT
THIRD SATURN BOOSTER LAUNCH
START 200K R&D PROGRAM
FIFTH SATURN BOOSTER W/188K ENGINES
DELIVER FIRST 200K ENGINE
COMPLETE 200K QUALIFICATION
ENGINES REQUIRED
TOTAL CONTRACTED 240
DELIVERED THRU FEB, 1965 185
TOTAL TEST EXPERIENCE 28 FEB, 1965
TESTS DURATION
SINGLE ENGINE 5229 384,231 SEC.
CLUSTER 63 4997 SEC.
I-E-E EPO 505 APR. 27, 1965
Figure 5

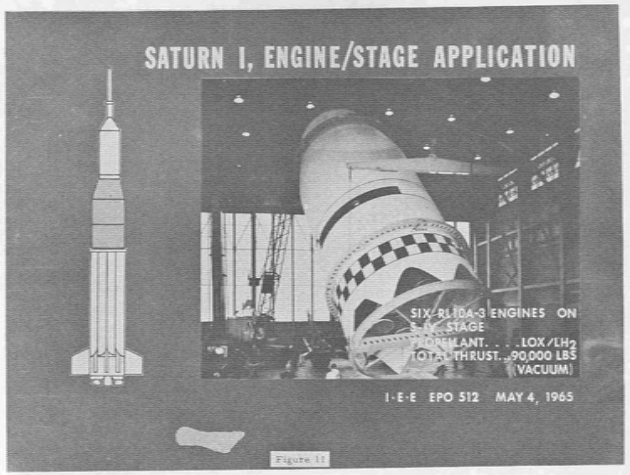

SATURN V, ENGINE/STAGE APPLICATION
FIVE F-1 ENGINES IN S-IC STAGE
PROPELLANTS.....LOX/RP-1
TOTAL THRUST..7,500,000 LB
I-E-E EPO 506 MAY5, 1965
Figure 6


F-1 ENGINE
THRUST- 1,500,000 LB
(SEA LEVEL)
THRUST DURATION- 150 SEC
ENGINE WT- DRY- 16,900 LB
BURN-OUT- 19,000 LB
PROPELLANTS- LOX & RP-1
CONTRACTOR-NAA/ROCKETDYNE
SYSTEM- SATURN V/S-IC
(5 ENGINES)
EPO 014
222 IN
148 IN
Figure 7

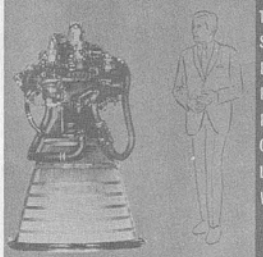

F-1 ENGINE
PROGRAM MILESTONES
1963
1964
1965
1966
1967
INITIATE ENGINE DEV PROGRAM (JAN 9, 1959)
1ST ENG MAIN STAGE FIRING (JUNE 1961)
1ST FULL THRUST & FULL DURATION FIRING (JUNE 1962)
6 DEGREE GIMBAL DEMO
DEL 1ST GROUND TEST ENG
COMPLETE ENGINE FRT
DEL 1ST SET OF FLIGHT ENGINES
COMPL ENG QUAL
Figure 8
I-E-E EPO 508 APR. 27, 1965


F-1 ENGINE SIMPLIFIED SCHEMATIC
FUEL
OXIDIZER
FUEL
OXIDIZER TURBOPUMP
FUEL TURBOPUMP
GAS GENERATOR
MAIN LOX VALVE NO. 2
MAIN FUEL VALVE NO. 2
MAIN LOX VALVE NO. 1
MAIN FUEL VALVE NO. 1
HYPERGOL
IGNITION MONITOR VALVE
FOUR WAY CONTROL VALVE
LEGEND
FUEL (RP-1)
OXIDIZER (LOX)
HOT GAS
Figure 9
I-E-E EPO 510 APR. 30, 1965


ATLAS CENTAUR
ENGINE / STAGE
APPLICATION
TWO RL10 ENGINES
ON CENTAUR STAGE
PROPELLANTS... LOX/LH2
TOTAL THRUST..30,000 LB
I-E-E EPO 511 MAY 5, 1965
Figure 10


SATURN I, ENGINE/STAGE APPLICATION
SIX RL10A-3 ENGINES ON
STAGE
PROPELLANT. . . .LOX/LH2
TOTAL THRUST...90,000 LBS
(VACUUM)
I-E-E EPO 512 MAY 4, 1965
Figure 11

RL10 ENGINE



	RL10A3	RL10A3-3
THRUST (ALTITUDE)	15,000 LB	15,000 LB
THRUST DURATION	470 SEC	470 SEC
SPECIFIC IMPULSE	433 SEC	444 SEC
ENGINE WEIGHT DRY	295 LB	300 LB
EXIT-TO-THROAT AREA RATIO	40 TO 1	57 TO 1
PROPELLANTS	LOX & LH_2	LOX & LH_2
CHAMBER PRESSURE	300 PSI	400 PSI
LENGTH	68 IN.	70.6 IN.
WIDTH	37.8 IN.	38.9 IN.

Figure 12

EPO 390

RL10 ENGINE MILESTONES ACHIEVED

OCTOBER 1958	CONTRACT AF-18 (600)-1774 WAS LET BY THE AIR FORCE TO PRATT & WHITNEY AIRCRAFT FOR THE DEVELOPMENT OF A 15,000-POUND THRUST ENGINE. THE ENGINE WAS DESIGNATED THE LR115, AND WAS TO BE USED IN A TWO-ENGINE CLUSTER FOR THE CENTAUR VEHICLE.
AUGUST 1959	ENGINE LR115 WAS FIRED SUCCESSFULLY.
JULY 1960	FIRST LR115 GROUND TEST ENGINE DELIVERED.
JANUARY 1963	FIRST SUCCESSFUL CLUSTER FIRING OF SIX RL10A-3 ENGINES CONDUCTED.
NOVEMBER 1963	FLIGHT OF CENTAUR AC-2 (FIRST SUCCESSFUL CENTAUR FLIGHT).
JANUARY 1964	FLIGHT OF SATURN SA-5 (FIRST TWO STAGE SATURN I FLIGHT).
APRIL 1964	INITIATE RL10A3-3 MODIFICATION (11 SEC. Isp INCREASE FOR CENTAUR).

Figure 13

I-E-E EPO 514 APR. 27, 1965

RL10 ENGINE SIMPLIFIED SCHEMATIC

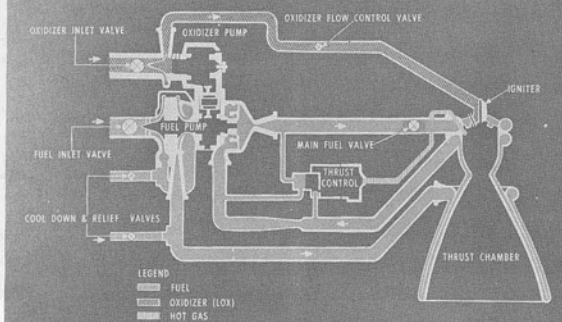


Figure 14

EPO 278

RL10 ENGINE PROGRAM MILESTONES

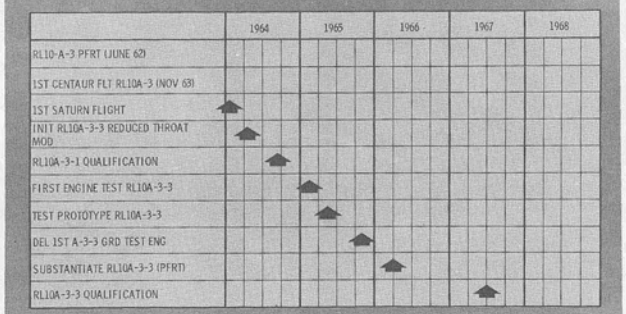


Figure 15

I-E-E EPO 516 APR. 27, 1965

RL10 ENGINE FIRING SUMMARY

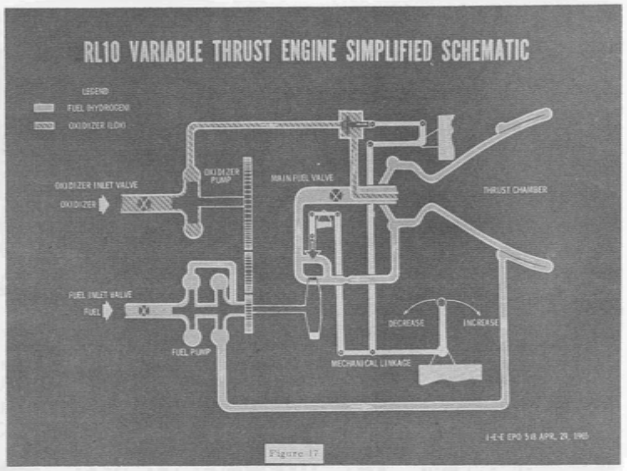
- FIRINGS AT PRATT & WHITNEY FLORIDA RESEARCH AND DEVELOPMENT CENTER
  1. RL10A-1 FIRINGS 1700
  2. RL10A-3 FIRINGS 2425
  3. RL10A-3-1 FIRINGS 985
  4. RL10A-3-3 (PROTOTYPE FIRINGS 47
  5. RL10A-4 (VARIABLE THRUST) FIRINGS 123
- FIRINGS ACCOMPLISHED IN THE FIELD

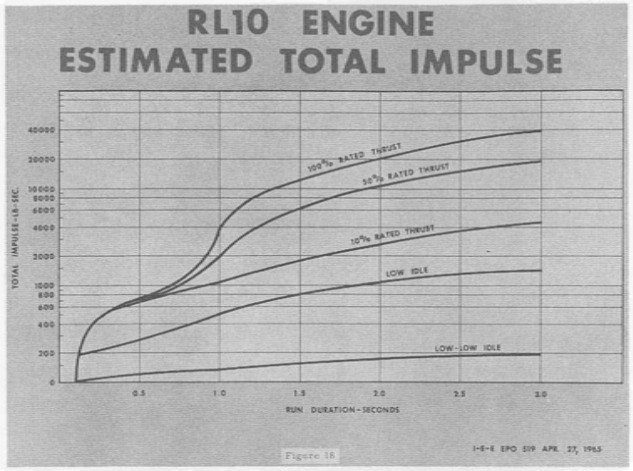
  TOTAL FIELD FIRINGS 384
- FIRINGS ACCOMPLISHED DURING VEHICLE FLIGHT
  1. SATURN 24
  2. CENTAUR 6
- TOTAL ACCUMULATED FIRING TIME

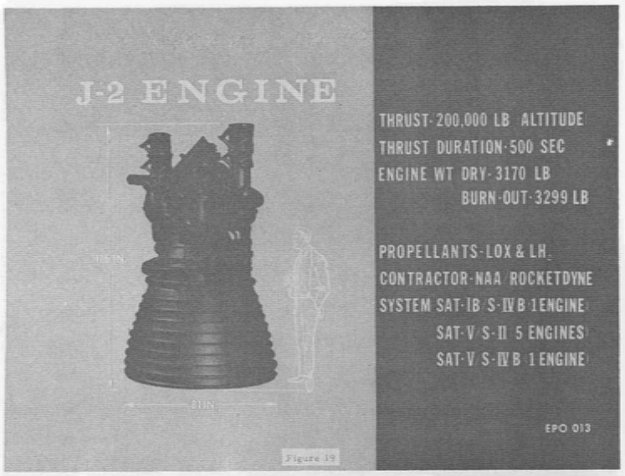
  ALL ENGINE FIRINGS - APPROXIMATELY 800,000 SECONDS
- MAXIMUM DURATION OF SINGLE FIRINGS - 1,680 SECONDS
- MAXIMUM THRUST ON ANY FIRING - 20,000 LBS.
- MINIMUM THRUST ON ANY FIRING WITH TURBO-PUMP OPERATING - 1500 LBS.
- THROTTLING RANGE (RL10A-4) - 10 TO 1
- LOW IDLE THRUST (PRESSURIZED MODE) - 300 LBS. TO 500 LBS.

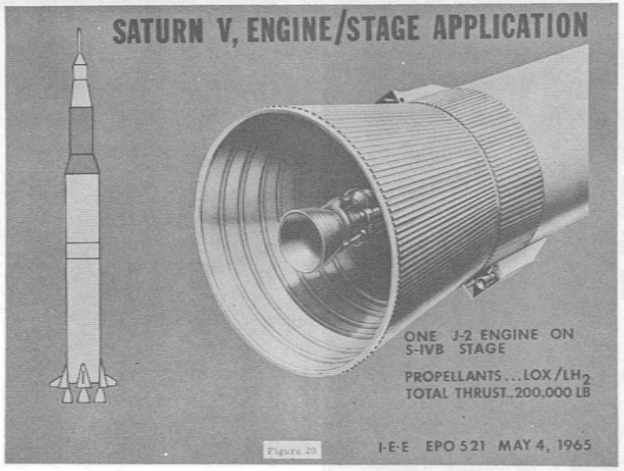
Figure 16

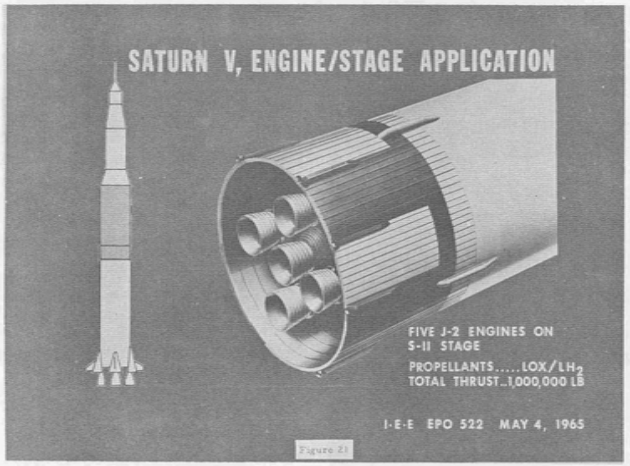
I-E-E EPO 517 APR. 12, 1965

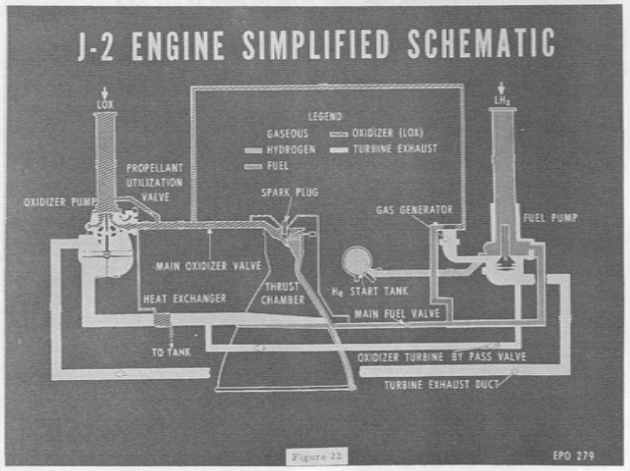

RL10 VARIABLE THRUST ENGINE SIMPLIFIED SCHEMATIC
LEGEND
FUEL (HYDROGEN)
OXIDIZER (LOX)
OXIDIZER INLET VALVE
OXIDIZER
OXIDIZER PUMP
MAIN FUEL VALVE
THRUST CHAMBER
FUEL INLET VALVE
FUEL
FUEL PUMP
DECREASE
INCREASE
MECHANICAL LINKAGE
I-E-E EPO 518 APR. 29, 1965
Figure 17

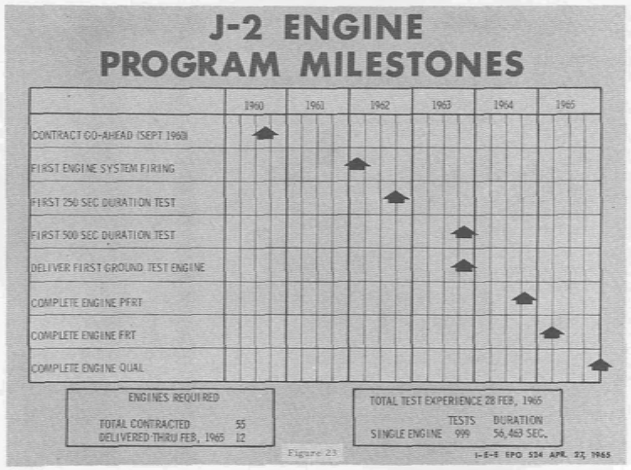

RL10 ENGINE
ESTIMATED TOTAL IMPULSE
TOTAL IMPULSE-LB-SEC
40000
20000
10000
8000
6000
4000
2000
1000
800
600
400
200
0
100% RATED THRUST
50% RATED THRUST
10% RATED THRUST
LOW IDLE
LOW-LOW IDLE
0.5
1.0
1.5
2.0
2.5
3.0
RUN DURATION-SECONDS
I-E-E EPO 519 APR. 27, 1965
Figure 18

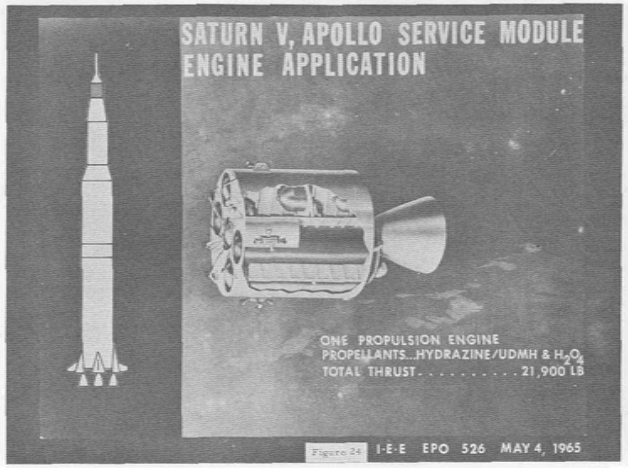

J-2 ENGINE
THRUST-200,000 LB ALTITUDE
THRUST DURATION-500 SEC
ENGINE WT DRY-3170 LB
BURN-OUT-3299 LB
PROPELLANTS-LOX & LH2
CONTRACTOR-NAA ROCKETDYNE
SYSTEM SAT-IB S-IVB 1 ENGINE
SAT-V S-II 5 ENGINES
SAT-V S-IVB 1 ENGINE
EPO 013
Figure 19

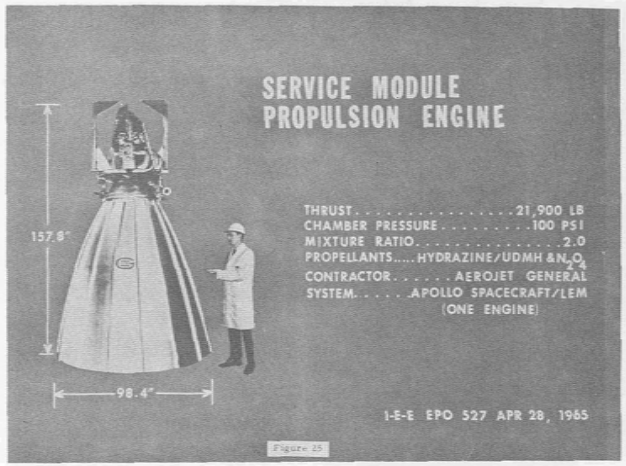

SATURN V, ENGINE/STAGE APPLICATION
ONE J-2 ENGINE ON
S-IVB STAGE
PROPELLANTS...LOX/LH2
TOTAL THRUST..200,000 LB
I-E-E EPO 521 MAY 4, 1965
Figure 20

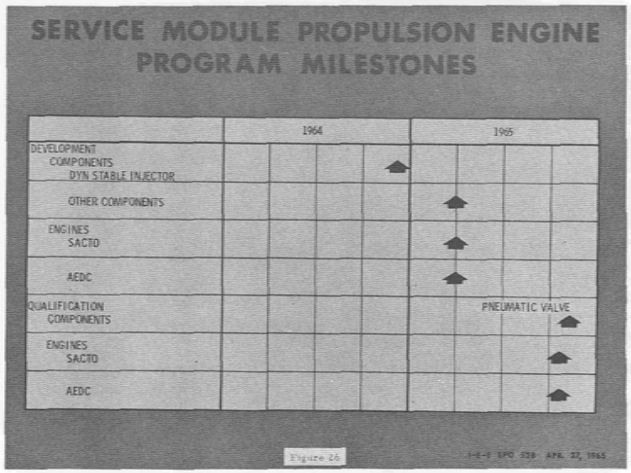

SATURN V, ENGINE/STAGE APPLICATION
FIVE J-2 ENGINES ON
S-II STAGE
PROPELLANTS.....LOX/LH2
TOTAL THRUST..1,000,000 LB
I-E-E EPO 522 MAY 4, 1965
Figure 21

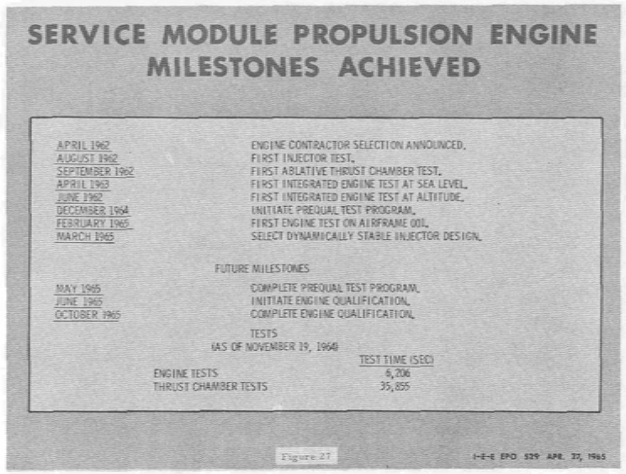

J-2 ENGINE SIMPLIFIED SCHEMATIC
LOX
LH2
LEGEND
GASEOUS
HYDROGEN
FUEL
OXIDIZER (LOX)
TURBINE EXHAUST
PROPELLANT UTILIZATION VALVE
OXIDIZER PUMP
SPARK PLUG
GAS GENERATOR
FUEL PUMP
MAIN OXIDIZER VALVE
HEAT EXCHANGER
THRUST CHAMBER
He START TANK
MAIN FUEL VALVE
TO TANK
OXIDIZER TURBINE BY PASS VALVE
TURBINE EXHAUST DUCT
EPO 279
Figure 22


J-2 ENGINE
PROGRAM MILESTONES
1960
1961
1962
1963
1964
1965
CONTRACT GO-AHEAD (SEPT 1960)
FIRST ENGINE SYSTEM FIRING
FIRST 250 SEC DURATION TEST
FIRST 500 SEC DURATION TEST
DELIVER FIRST GROUND TEST ENGINE
COMPLETE ENGINE PFRT
COMPLETE ENGINE FRT
COMPLETE ENGINE QUAL
ENGINES REQUIRED
TOTAL CONTRACTED 55
DELIVERED THRU FEB, 1965 12
TOTAL TEST EXPERIENCE 28 FEB, 1965
TESTS DURATION
SINGLE ENGINE 999 56,463 SEC.
Figure 23
I-E-E EPO 524 APR. 27, 1965


SATURN V, APOLLO SERVICE MODULE
ENGINE APPLICATION
ONE PROPULSION ENGINE
PROPELLANTS...HYDRAZINE/UDMH & H_2O_4
TOTAL THRUST. . . . . 21,900 LB
Figure 24
I-E-E EPO 526 MAY 4, 1965


SERVICE MODULE
PROPULSION ENGINE
157.8"
98.4"
THRUST. . . . . 21,900 LB
CHAMBER PRESSURE. . . . . 100 PSI
MIXTURE RATIO. 2.0
PROPELLANTS. . . . . HYDRAZINE/UDMH & N_2O_4
CONTRACTOR. . . . . AEROJET GENERAL
SYSTEM. . . . . APOLLO SPACECRAFT/LEM
(ONE ENGINE)
I-E-E EPO 527 APR 28, 1965
Figure 25


SERVICE MODULE PROPULSION ENGINE
PROGRAM MILESTONES
1964
1965
DEVELOPMENT COMPONENTS
DYN STABLE INJECTOR
OTHER COMPONENTS
ENGINES
SACTO
AEDC
QUALIFICATION COMPONENTS
PNEUMATIC VALVE
ENGINES
SACTO
AEDC
Figure 26
I-E-E EPO 528 APR. 27, 1965


SERVICE MODULE PROPULSION ENGINE
MILESTONES ACHIEVED
APRIL 1962 ENGINE CONTRACTOR SELECTION ANNOUNCED.
AUGUST 1962 FIRST INJECTOR TEST.
SEPTEMBER 1962 FIRST ABLATIVE THRUST CHAMBER TEST.
APRIL 1963 FIRST INTEGRATED ENGINE TEST AT SEA LEVEL.
JUNE 1962 FIRST INTEGRATED ENGINE TEST AT ALTITUDE.
DECEMBER 1964 INITIATE PREQUAL TEST PROGRAM.
FEBRUARY 1965 FIRST ENGINE TEST ON AIRFRAME 001.
MARCH 1965 SELECT DYNAMICALLY STABLE INJECTOR DESIGN.
FUTURE MILESTONES
MAY 1965 COMPLETE PREQUAL TEST PROGRAM.
JUNE 1965 INITIATE ENGINE QUALIFICATION.
OCTOBER 1965 COMPLETE ENGINE QUALIFICATION.
TESTS
(AS OF NOVEMBER 19, 1964)
TEST TIME (SEC)
ENGINE TESTS 8,206
THRUST CHAMBER TESTS 35,855
Figure 27
I-E-E EPO 529 APR. 27, 1965

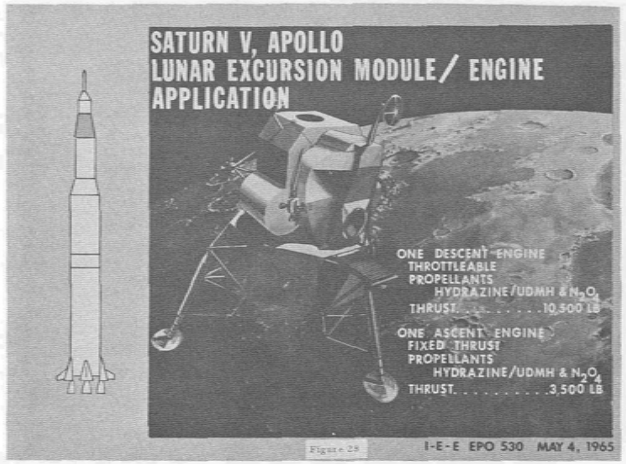

SATURN V, APOLLO
LUNAR EXCURSION MODULE/ ENGINE
APPLICATION
ONE DESCENT ENGINE
THROTTLEABLE
PROPELLANTS
HYDRAZINE/UDMH & N_2O_4
THRUST. . . . . 10,500 LB
ONE ASCENT ENGINE
FIXED THRUST
PROPELLANTS
HYDRAZINE/UDMH & N_2O_4
THRUST. . . . . 3,500 LB
Figure 28
I-E-E EPO 530 MAY 4, 1965

Figure 29

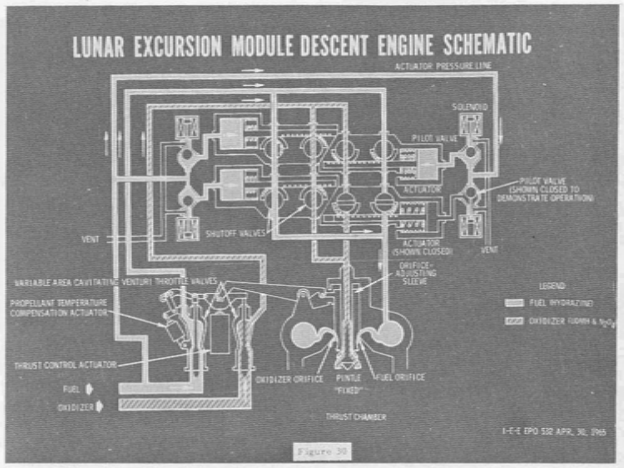


Figure 30

LUNAR EXCURSION MODULE DESCENT ENGINE PROGRAM MILESTONES

	1965	1966
ENGINE		
ENGINE DEVELOPMENT AND PERFORMANCE TESTS	▲	
FINAL DEVELOPMENT - ENVIRON. TREATMENT AND TEST	▲	
ENGINE QUAL - ENVIRONMENTAL TREATMENT AND TEST		▲ STL ▲ AEDC
MAJOR COMPONENT DEVELOPMENT	▲	
QUAL LEVEL	▲	

Figure 31

I-E-E EPO 533 APR. 27, 1965

LUNAR EXCURSION MODULE DESCENT ENGINE MILESTONES ACHIEVED

JUNE 1963	ENGINE CONTRACT AWARDED.
JANUARY 1964	BASIC CONCEPT OF VARIABLE AREA INJECTOR PROVED FEASIBLE.
FEBRUARY 1964	FIRST FULL-SCALE ABLATIVE NOZZLE TEST.
MAY 1964	FIRST ABLATIVE COMBUSTION CHAMBER FIRING.
SEPTEMBER 1964	FIRST 10 TO 1 THROTTLING TEST AT VACUUM.
OCTOBER 1964	DEMONSTRATION OF MOST STRINGENT DUTY CYCLE (ABORT DUTY CYCLE).
NOVEMBER 1964	DEMONSTRATION OF COMBUSTION CHAMBER PERFORMANCE OPERATING AT PROPELLANT TANK RELIEF PRESSURE.

TESTS
(AS OF DECEMBER 1964)

	NUMBER	DURATION
THRUST CHAMBER	66	12,340 SEC.
ENGINE	15	1,978 SEC.

Figure 32

I-E-E EPO 534 APR. 27, 1965

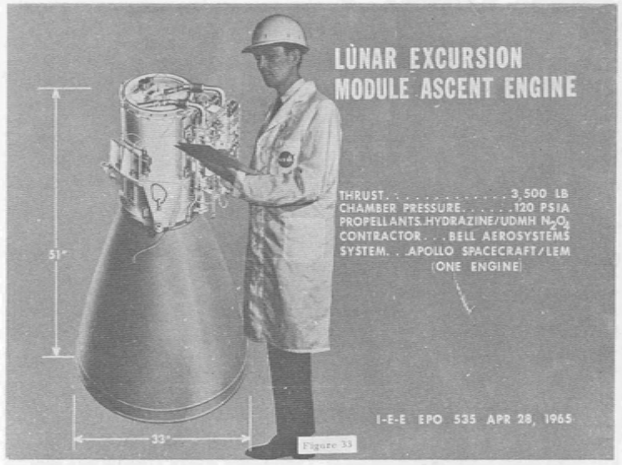


Figure 33

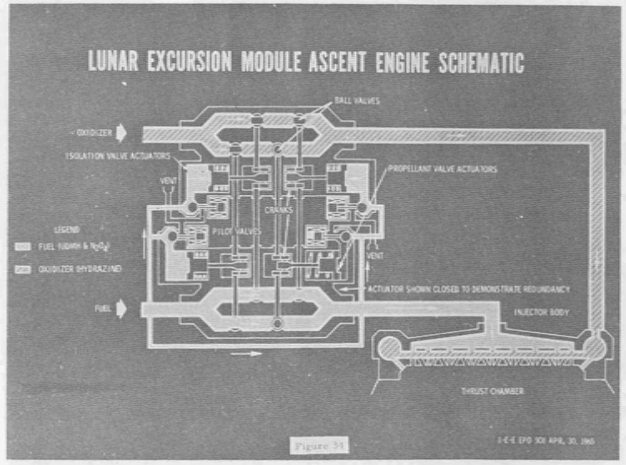


Figure 34