ENGINE PROGRAM OFFICE

DEVELOPMENT OF LOX-HYDROGEN ENGINES FOR THE SATURN APOLLO LAUNCH VEHICLES

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Abstract

During the development of the RL-10 and J-2 engines, many problems were encountered. Solutions to the significant problems are contained. A description of these LOX-Hydrogen engines, outlining the unique features of each will be given. Performance parameters for both engine systems are tabulated. Specific applications to various stages are shown. Start and restart conditions at altitude are a very important adjunct to the engine development and are presented in the paper. Testing an engine designed to operate at altitude at ambient sea level conditions presented some interesting problems and required peculiar test equipment. The solution to these problems and a description of the test equipment will be covered. Flight data revealed some anomalies that were later verified at the altitude test facility at Arnold Engineering Development Center, Tullahoma, Tennessee. A description of the anomalies and verification testing will be made.

Introduction

The nature of propulsion system development is such that it becomes the pacing item in a new vehicle program. This has been true in aircraft as well as missile programs. The extensive testing, design, redesign, and retest have shown that propulsion systems generally must lead by several years in the development cycle. This necessitates the initiation of an engine development program prior to the detail design of the vehicle. This presents the obvious problems of interface, power requirement definition, control system relationships, etc., which must be again evaluated as the vehicle design is finalized. Engine redesign or compromises usually result from these iterations.

Once the technical need has been established for the development of a new propulsion system and the economics of the system have been evaluated, then the design trade-offs begin. Vehicle requirements which include such basic factors as staging, payload weight, mission flexibility, and reliability must be fed back into the basic engine design as the system evolves. Wherever possible, engines are designed to be utilized in more than one stage. Following this philosophy the RL-10 engine found use on two different vehicles and the J-2 engine is utilized on three stages of two vehicles.

This "common engine" approach allows the contractor to concentrate on one design, results in higher reliability due to repetitive testing of a single configuration, and reduces production costs since the engines can be bought in larger quantities. A common engine configuration also simplifies the spare parts and field support requirements, and permits interchangeability of basic components.

Thus new engine development programs are optimized by a meld of the practical use of the best technology available within the constraints involved and a vigorous trade-off of the performance parameters. The choice of expansion ratio, for example, was one not only of performance, but of the economics involved in building the test facilities required for engine development. The RL-10, being a smaller engine, could use a higher expansion ratio (and thus take advantage of the higher specific impulse) because it was practical to build steam ejectors and diffusers for engines in this size range. Such a choice for the J-2 engine would have been prohibitive from the standpoint of cost due to the size of the facilities involved. Hence, one of the reasons a 27.5/1 expansion ratio was chosen for the J-2 engine was to accommodate testing at sea level conditions.

The guiding philosophy of the NASA-Marshall Space Flight Center in developing engines is to schedule a detailed component test program followed by an in-depth engine test program to ferret out as many latent hardware defects as possible before flight testing. A vigorous production support program is maintained concurrent with the development phase and the improvements are incorporated into the embryonic design as soon as practical. This plan has worked successfully because it continues to challenge the best technical minds to reach out "over the horizon" for better ideas which have resulted in continual payload increases for NASA's launch vehicles. This production support effort has been the backbone for resolving the problems resulting from engine-stage integration and testing. This process continues as long as vehicles using the engines are flying. The feedback of production and flight problems into the engine programs is an essential element in deriving a reliable system, and also serves the cofunction of allowing the engine design, testing, and manufacturing procedures to catch up with the state of the art.

Part I: Development of the RL-10 Engine

The Apollo Program was conceived with the idea that a design evolution would be required before the large vehicle suitable for manned lunar landing could become a reality. The planned evolution was from the Saturn I to the Saturn IB, and finally to the Saturn V. Since the liquid oxygen (LOX)-liquid hydrogen (LH_2) RL-10 engine was already in development on the Centaur program when the Saturn I program was started, an uprated version (LR-119) was chosen for the high performance upper stage propulsion. The optimistic development program for the LR-119 failed to materialize and coupled with major difficulties in the RL-10 development program resulted in a program redirection. A common version of the RL-10 engine was defined to meet the requirements of both the Centaur and Saturn I programs and allowed the contractor to concentrate on one development program.

An all cryogenic propellant system of liquid hydrogen and liquid oxygen was chosen because of the high specific impulse attainable. Major breakthroughs in liquid hydrogen technology made both propellants readily available and relatively inexpensive. In addition these propellants are nontoxic and give stable combustion.

The RL-10 engine was successfully used on the Surveyor Program in an Atlas-Centaur vehicle (Figure 1) to boost the payload to a lunar landing. It was also used in the Saturn I vehicle (Figure 2) to place three Pegasus meteoroid technology satellites into orbit.

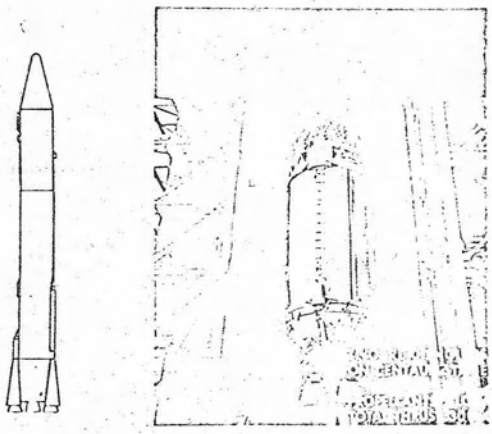

Figure 1. Atlas Centaur Engine/Stage Application

In the Saturn I program the RL-10 engine became an excellent test-bed for Apollo hardware development. The S-IV stage of the Saturn I was a forerunner of the S-IVB stage in the Saturn IB and V vehicles. Likewise, experience gained in developing the S-IV and Centaur stages was utilized in designing the Saturn V S-II stage.

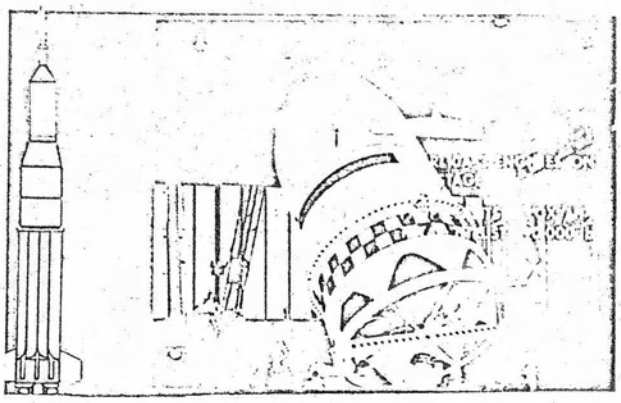

Figure 2. Saturn I Engine/S-IV Stage Application

Description

The RL-10 engine utilizes a regeneratively cooled thrust chamber and a turbopump-fed propellant flow system. Due to its high heat capacity, the liquid hydrogen very effectively cools the thrust chamber. While passing through the thrust chamber tubes, the hydrogen picks up heat and is expanded in a two-stage turbine to drive a single geared turbopump. The fuel is then injected into the combustion chamber. This "topping" cycle provides a performance gain of approximately 1/2 to 1 percent over that of a conventional gas generator type cycle. Oxidizer is pumped directly to the propellant injector through the mixture ratio control valve. Thrust control is achieved by regulating the amount of fuel bypassed around the turbine as a function of combustion chamber pressure in order to vary turbopump speed and thereby control engine thrust. Ignition is accomplished by means of an electric spark-torch igniter recessed in the propellant injector face. Starting and stopping are controlled by pneumatic valves which receive their supply of helium through electrically operated valves. Major parameters of the RL-10 engine are depicted in Figure 3.

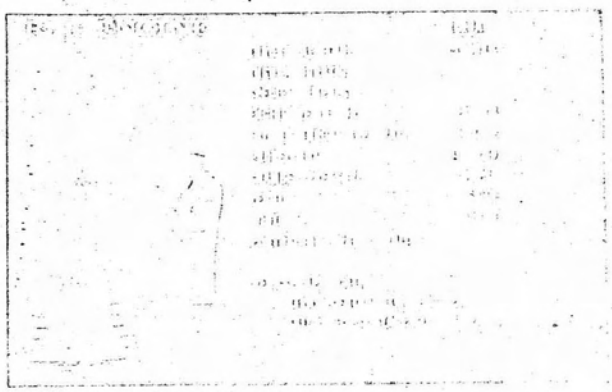

Figure 3. RL-10 Engine Major Parameters

A functional description of the engine is shown in Figure 4.

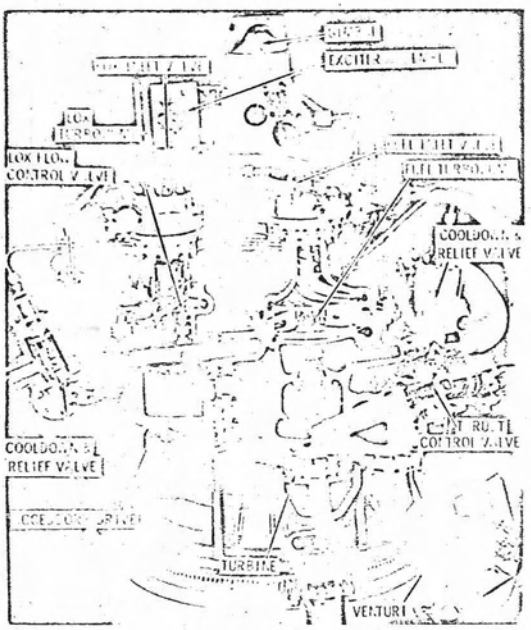

Figure 4. RL-10 Engine

Development Program

Figure 5 shows the development cycle for the RL-10 engine which falls within the typical 5-7 year span required for propulsion systems.

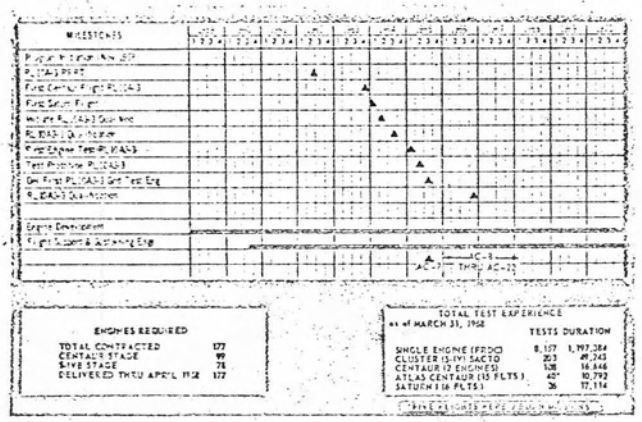

Figure 5. RL-10 Project Milestones

The first Centaur flight utilizing an RL-10 engine occurred five years after program initiation. Approximately $240 million dollars was spent on the RL-10 engine program of which about seventy percent was spent on the development phase.

The engine, in its various versions has completed over one million seconds of total firings; and has been flown on 15 Centaur flights and six Saturn flights. A total of 66 engines have been flown with no malfunction which affected the mission.

Testing under simulated Centaur stage conditions on a dual-position vertical test stand contributed significantly to the lack of engine problems once vehicle ground and flight testing commenced. Vehicle components such as the hydraulic power pack, propellant utilization drive motor, recirculation ducts and diffusers, vehicle boost pumps, and retromaneuver discharge ducts were tested on the dual-engine stand under conditions closely approximating those found on the actual stage in flight. This not only allowed interactions on the engines to be identified, but permitted accurate simulation data for purposes of developing the stage components.

Production support effort was initiated to minimize the impact of production problems and continually improve system reliability and flight worthiness. This included establishment of vehicle starting sequences and limits, optimization of prelaunch chilldown and boost pump settings, and investigation of flight problems.

Basic technology was developed in special test facilities, hydrogen injectors, ignition systems, idling and throttling, reduced net positive suction head (NPSH) operation, the use of cryogenics for a bearing and gear coolant and the feasibility for a "zero chilldown time" engine.

The RL-10 has not only been a reliable engine system for the Saturn and Centaur programs, but has served as a "test-bed" for other areas of technology. The use of fluorine as an oxidizer was proven on modified engines of an early vintage to provide one of the first "flight weight" fluorine prototype engines. This was accomplished by modifying the pump seals in the oxidizer system to provide fluorine compatibility and by changing the engine "trim".

A modified RL-10 engine has run successfully in the throttling mode down to one percent of the rated engine thrust. The feed system was stable at all thrust levels in this range. Further, a modified RL-10 has operated in the low-idle mode, in which the turbopump does not rotate but the engine operates on tank pressure with gaseous, liquid, or mixed-phase propellants.

Other areas, such as hypergolic ignition through the use of a small percentage of trioxygen difluoride in the oxidizer and instant-start or zero cooldown time accomplished by insulating the fuel pump internal surfaces, were investigated using the RL-10 engine as a "test-bed" and were proven feasible.

Problems

The RL-10 engine encountered a wide gamut of problems during development. Some of the major problem areas are listed below.

(1) thrust overshoot
(2) gimbal block lubrication
(3) spark igniter deficiency

Thrust Overshoot. The Centaur vehicle was unable to accept more than 15 percent thrust overshoot. Thrust exceeding the specified limit occurred because the engine thrust control valve (Figure 6) was set to maintain a chamber pressure of 300 psi and did not by-pass any turbine drive gas until the nominal thrust level was reached. The system momentum was not retarded early enough to prevent the overshoot. Adjustments in spring pressure on the thrust control piston prevented excessive thrust overshoot, but caused the system to stabilize below the nominal thrust level.

The problem was solved by the incorporation of a "pneumatic reset" on the thrust control. The thrust control by-pass was set to open at a lower chamber pressure (approximately 270 psi), and the desired nominal 300 psi chamber pressure was reestablished by use of the pneumatic reset system.

The reference pressure on the back side of the thrust control by-pass piston is atmospheric pressure which is essentially zero psi at the engine operating altitude. When the thrust controller by-pass is actuated at the lower chamber pressure level, hydrogen is vented into the thrust control body. By orificing the thrust control valve body vent, the body pressure and piston reference pressure are increased approximately 30 psi which resets the by-pass relief pressure to obtain the nominal 300 psi chamber pressure. By this method the reset pressure lags the body pressure during engine acceleration which allows the thrust control valve to return to a nominal position before the desired 300 psi chamber pressure is reached.

Development of a backup electronic thrust controller was initiated in the production support program and the design had progressed to the point where a decision could have been made to use it when the modified pneumatic controller proved satisfactory.

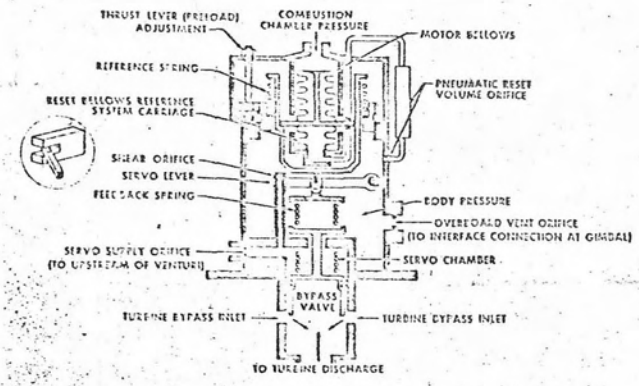

Figure 6. Thrust Control Valve Schematic

Gimbal Block Lubrication. The gimbal assembly is the component with the highest unit loading and is the only loaded component that is exposed to high vacuum during engine operation. Space vacuum effect on materials is significant in two respects.

(1) evaporation of solids
(2) vaporization of surface gas layers

Evaporation of material is not a significant problem because of the rate of material loss is negligible at temperatures below 300°F and the engine surface temperatures remain continuously below this critical range. Vaporization of surface gas layers of engine components is a significant problem because of the associated phenomenon of "cold welding." Degassing a metal surface removes the oxide coating that is characteristically present within the atmosphere of the earth and protects the metal surface against molecular attraction of similar clean materials. Also the friction between the rubbing metals increases proportionately to the extent that the surfaces are degassed. A dry film lubricant (molybdenum disulphide) was developed for the gimbal block to reduce the increased torque encountered under space conditions to a tolerable level well within the structural capability of the engine and the vehicle.

Spark Igniter Deficiency. The first static test stands that were built for the RL-10 development program positioned the engines in a horizontal attitude for firing into a long diffuser. The diffusers were used to create a vacuum around the engine prior to start. Later a dual engine test stand was built in which the engines were positioned vertically and fired downward into diffusers from an elevated facility. The first attempt to fire an RL-10 engine vertically resulted in an explosion that extensively damaged the engine and test facilities. The ensuing investigation proved that LOX was introduced into the thrust chamber in such a way that it bypassed the igniter. During the start of a horizontal engine firing, the LOX formed a pool inside the thrust chamber. Boil-off from this pool mixed with the hydrogen to form a combustible mixture for ignition. When the engine was fired vertically, the LOX dropped to the bottom of the diffuser tubes, and the ignition occurred at that point. The explosion progressed up through the diffusers to the engines and test facility. The engine injector was modified so that both oxygen and fuel were routed past a recessed spark-torch igniter during start and solved the problem.

Uprating

There has been a sustained spectacular payload increase over the years attributed to engine improvement. As shown in Figure 7, an increase of almost 800 lbs. of payload has resulted from engine improvements made over a six year period. Essentially all of these ameliorations resulted from improvements in efficiency with the ISP progressing

step-wise from the original design specification limit of 412 lb. sec/lb to a delivered nominal of 442 lb sec/lb.

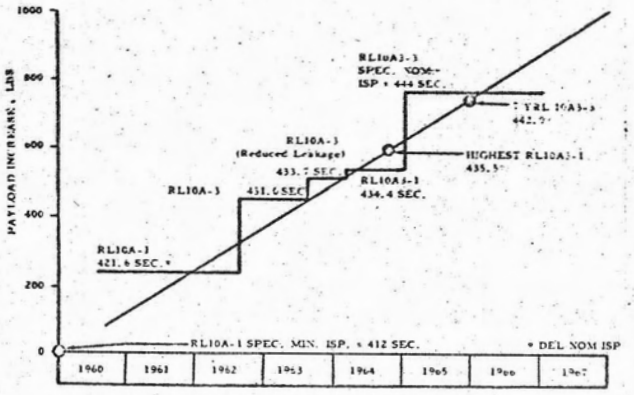

Figure 7. RL-10 Engine Contribution to Centaur Payload Increase

Accomplishments

Some of the accomplishments of the RL-10 engine program are listed below:

(1) First LH_2/LOX engine
(2) Developed LH_2 technology
(3) Combustion Stability
(4) Multiple start capability
(5) Idling and throttling capability
(6) Test-bed for other areas of technology

The RL-10 engine has become a work-horse in the propulsion field. In addition to being an excellent experimental tool it also was the forerunner of the J-2 and M-1 engines.

Part II: Development of the J-2 Engine

Design Philosophy

The J-2 engine is the second major propulsion system developed using liquid hydrogen as the fuel. Development of the J-2 engine was undertaken to satisfy the need for a high-thrust, high-performance upper stage engine which would be capable of restarting in space. Propellant utilization and thrust programming required an additional flexibility for the engine system. Much of the experience and technology gained on the RL-10 program was directly applicable to the J-2 engine. Since the J-2 was to be used on a manned vehicle, considerable attention had to be given to achieving a high reliability through extensive component and engine systems ground testing to resolve any potential problems before flight testing. In the design of the J-2, attention has been focused on potential failure modes and inherent design characteristics which could prevent these failures. Welded joints are used throughout the engine to prevent leaks. Dual seals, with intermediate bleeds, are used at all hot gas and propellant separable connections.

Design flexibility had to be maintained since the J-2 engine was to be used in the second stage of the Saturn IB and the second and third stages of the Saturn V, as shown in Figures 8 and 9.

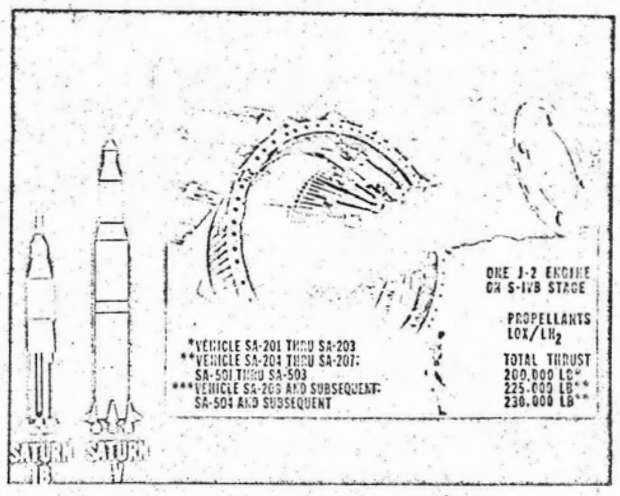

Figure 8. Saturn IB-V, Engine/S-IVB Stage Application

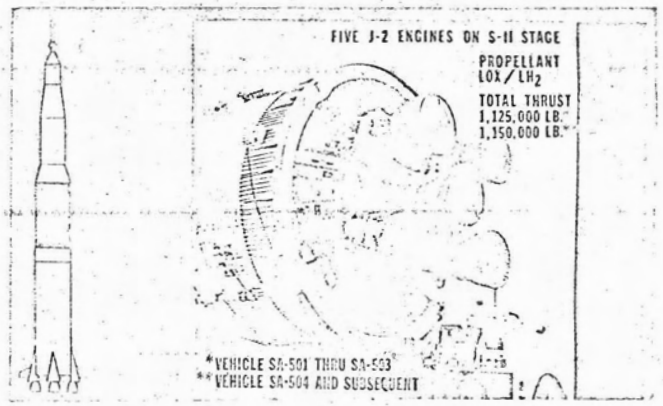

Figure 9. Saturn V, Engine/S-II Stage Application

It was decided in the early stages of the Apollo Program that the engines for the second and third stages would be completely interchangeable. The J-2 engine requires only minor changes to permit application in the second and third stages. Changes in the insulation, orificing, and heat exchanger connections are required, and provisions must be made for refilling the hydrogen start bottle for the S-IVB restart.

Description

The J-2 rocket engine, shown in Figure 10, is a 230,000-pound thrust, multiple-restart, gimballed engine utilizing liquid hydrogen and

liquid oxygen as propellants and is designed to be used singularly or clustered. The engine has a regeneratively cooled thrust chamber, separate LH_2 and LOX pumps driven by a gas generator connected in series to turbines powering each pump, gaseous hydrogen for engine start, and an integral helium control bottle for pneumatic valve operation. An electrical control assembly (ECA) package controls the engine sequencing and provides the high voltage transformers for the spark ignition system.

The major parameters are shown in Figure 11.

Figure 10. J-2 Engine

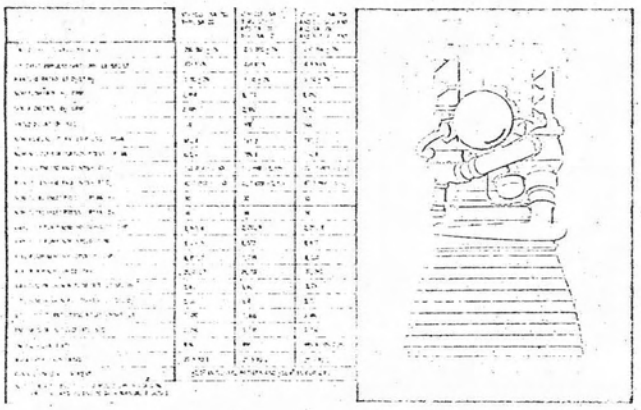

Figure 11. J-2 Engine Parameters

Development Program

Significant developments within the J-2 program are shown in Figure 12. The development test program has three distinct facets: Component testing, engine systems testing, and flight testing. The component testing consists of qualifying all major components and verification testing for flight engines. Both the first flight configuration and the uprated configuration engines have been qualified. The flight test program to establish overall vehicle performance has verified the engine's flight capabilities. The J-2 engine has been under development for almost eight years.

Extensive component and system testing have demonstrated a reliability of 0.9950 at a 50 percent confidence level. There have been 3,183 single-engine tests with an accumulated test time of 309,140 seconds. There have been 57 cluster firing tests, consisting of the full five-engine clustered configuration conducted for 10,113 seconds of firing time. The J-2 engine has made four Saturn IB flights and two flights in the Saturn V vehicle.

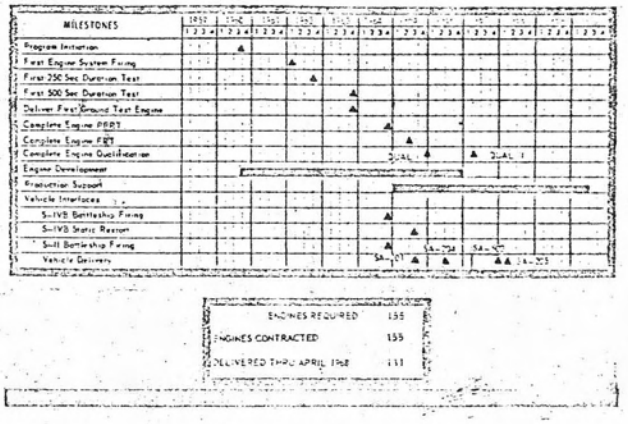

Figure 12. J-2 Engine Project Milestones

Problems (Non-Flight). The course of the J-2 development has not been without incidents. The first engine tests exhibited side loads during start at sea level conditions; however, there have been no side load problems while testing at altitude. The side loads were due to gas flow separations inside the thrust chamber. 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 arms to physically hold the engine during engine start.

Fuel pump stall was a development problem early in the program. The fuel pump flow entered the regeneratively cooled jacket before passing into the injector. As the fuel pump delivered the first fuel to the relatively warm chamber, a considerable volume of hydrogen gas was created. The gas could not pass through the injector at a rate sufficient to keep up with the flow. The solution was to prechill the pump and chamber with liquid hydrogen to limit the temperature conditions under which a start would be attempted.

Another problem involved sequencing of the main liquid oxygen valve opening, allowing an excessive amount of liquid oxygen to be passed on to the gas generator. This caused excessive gas temperature and pressure spikes in the gas generator. The problem was corrected by changing the sequencing of the main liquid oxygen valve to aid in regulating the flow of liquid oxygen to the gas generator. This has also helped to alleviate the previously discussed fuel pump stall problem.

Three serious component problems which occurred during the development of the J-2 engine were ECA solder joints failures, LOX turbine wheel cracks, and ECA timer failures.

ECA Solder Joint Failures. Spurious engine cutoff signals were encountered during checkout of a vehicle at Cape Kennedy. The signals were traced to cracked solder joints in the engine ECA package. The electrical control assembly, which is the nerve center of the engine, sequences the engine operations throughout the complete cycle.

As shown in Figure 13, the crack occurred in a joint where a component was soldered into a printed circuit board. Subsequent temperature stressing caused the joint to fail as shown in Figure 14.

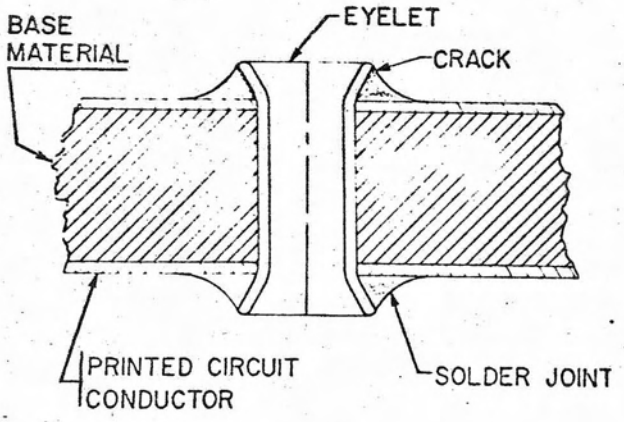

Figure 14. Typical Solder Joint Failure

Since this had been recognized as a potential problem area, a substitute shown in Figure 15 was being developed in the production support program. The terminal solid pin acts as a heat sink and results in a vastly improved joint in which microscopic cracks that are no deterrent to good continuity are occasionally encountered.

Figure 13. ECA Solder Joint Crack

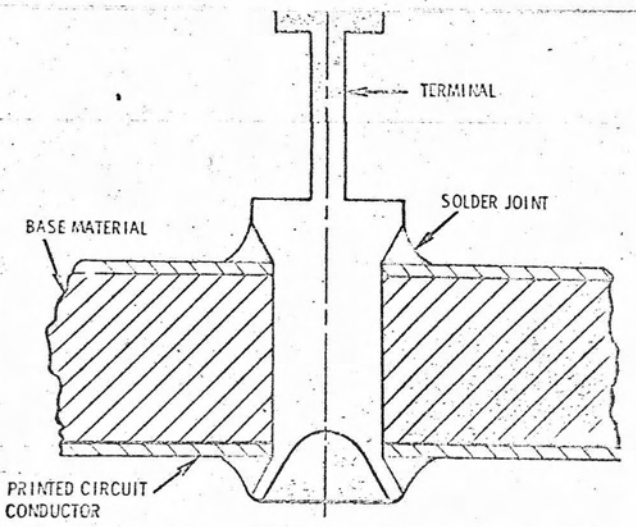

Figure 15. Improved Solder Joint Design

LOX Turbine Wheel Cracks. Cracking was detected during a routine inspection of the second stage LOX turbine wheel (Figure 16).

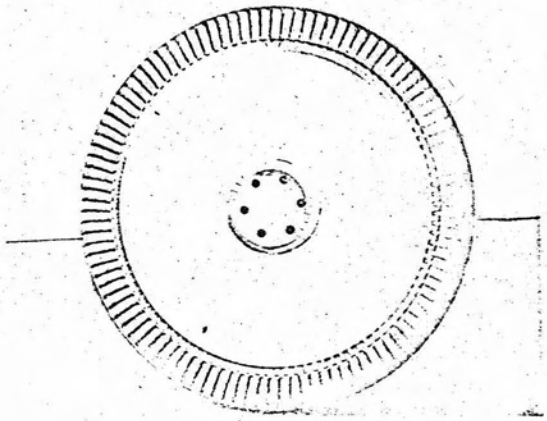

Figure 16. LOX Turbine Wheel

As shown in Figure 17 the wheel hub was cracked around the entire periphery. Since this wheel operated at about 8,000 rpm, the potential energy available could cause catastrophic results in case of failure. It was discovered that in going through the mixture ratio excursion on the J-2 engine, a standing wave was generated in the turbine wheel which continued to flex the metal at the hub area until fatigue occurred. By changing the web thickness and the stiffening characteristics of the wheel the destructive standing wave vibration was eliminated.

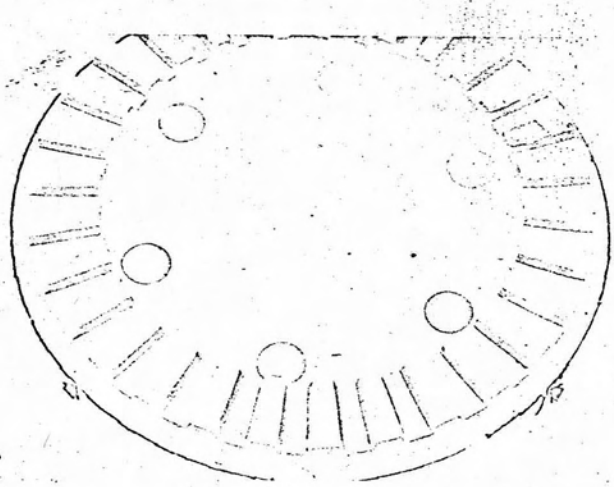

Figure 17. LOX Turbine Wheel Hub Crack

ECA Timer Failures. An electrical control assembly timer failure occurred during checkout at Cape Kennedy. Throughout the history of the timer development encompassing about eight years only eleven failures were noted. Of these failures six were attributed to transistors in the timer circuit. A thorough investigation of the problem attributed the cause to be whisker growth within the case enclosing a CK 65 transistor which is an integral part of the timer. The whisker growth is shown in Figure 18.

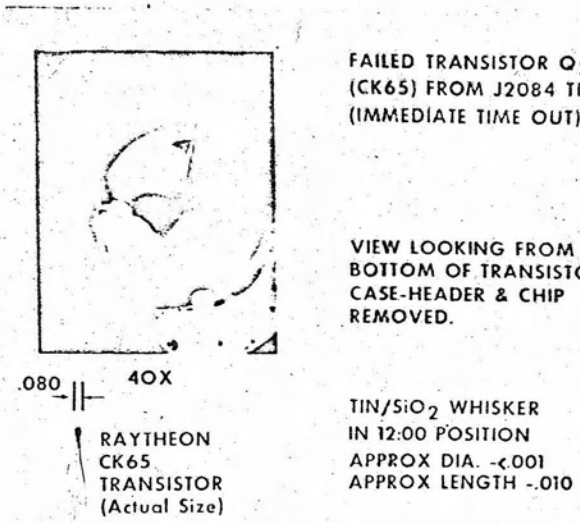

Figure 18. Failed ECA Timer Transistor

This whisker growth phenomenon is a characteristic of noble base metals such as tin, indium, lead and silver. The growth is known to be accelerated by (x-ray), moisture, electrical potential difference, and the presence of silica. Since the transistor has a tin plated case, contains silica gel desiccant is assembled in an uncontrolled humidity environment, has a poor hermetic seal, and is exposed to an electrical potential difference it makes an excellent test-bed for whisker growth. There was concern over replacing a qualified component which had 2296 engine hot fire tests for a demonstrated reliability of 0.9999 at a 50 percent confidence level and 50,000 exposures in non-hot fire tests for a reliability of 0.9998 at a 50 percent confidence level. This is another example of the Apollo program pushing the state of the art. The transistors were the best available in the industry at the time they were chosen for the application. However, the long lead time involved in manufacturing an item as complicated as the Saturn V vehicle resulted in long storage time for numerous sub-components. This time period was sufficiently long enough to allow the whisker to grow to the extent that it caused a short between the transistor element and the case. The transistors were replaced with a high reliability improved unit made under more rigid manufacturing specifications and with a case material which did not support whisker growth.

Problems (Flight).

AS-203 hot crossover duct. Flight data from AS-203 indicated that the crossover duct (See Figure 10) which connects the fuel turbine to the oxidizer turbine did not cool as rapidly as anticipated and the additional energy in the system influenced the engine start transient.

Due to excellent heat transfer characteristics in a sea level environment, the duct had always cooled quickly before a restart test was initiated. Tests were conducted at the Arnold Engineering Development Center in an environmental test cell in which the entire Saturn S-IVB stage can be maintained at a simulated altitude of 100,000 feet during engine start transient and steady state operation. These tests verified that the J-2 engine would start and restart satisfactorily under large temperature extremes by starting with the propellant utilization valve wide open. This operation insured minimum energy in the system at startup since the engine would be programmed for an initial fuel rich mixture ratio which would put less heat into the gas generator turbine exhaust gases that pass through the crossover duct. In addition the crossover duct was painted black to enhance the transfer of heat out of the system.

AS-502 Flight. Anomalies were encountered in both the S-II and S-IVB stages during flight. The S-II anomalies included environmental changes, indicated yaw actuator malfunction, performance shifts, and premature cutoff.

Environmental Changes. The 502 flight was normal by 501 standards until 225 seconds after liftoff. At that time temperature began to drop rapidly in selected engine areas as shown in Figures 19, 20, and 21. Engine 2 cutoff was preceded 0.3 seconds by indications of hot gas impingement as depicted in Figures 22 and 23.

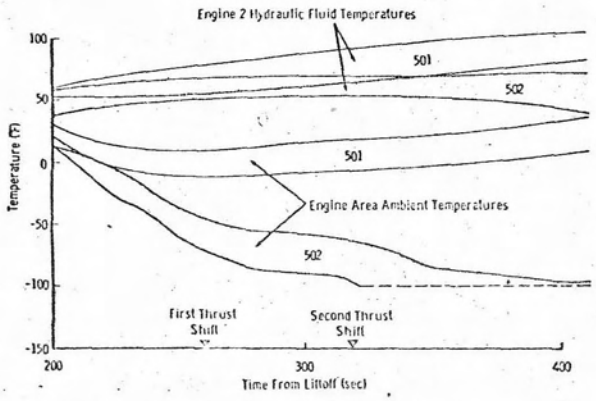

Figure 19. Cooling Trends Prior to Engine Number 2 Cutoff

Cooling trends were established by 280 seconds after liftoff and were accelerated at 320 seconds. Both time intervals correspond closely with thrust shifts. Just prior to engine number 2 cutoff the engine and thrust cone area indicated hot gas impingement. Following engine number 2 cutoff there was a cooling trend in the engine 2-5 quadrant indicating a cryogenic line rupture.

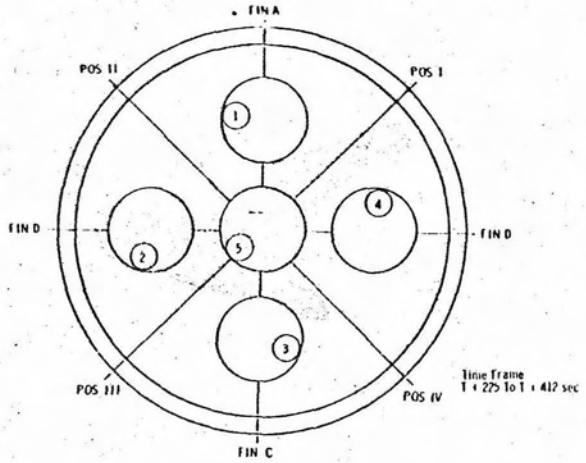

Figure 20. Bottom View of Area Cooled Prior to Engine Number 2 Cutoff

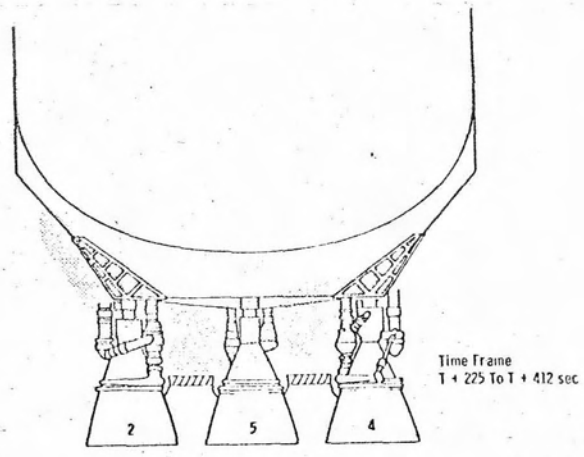

Figure 21. Side View of Area Cooled Prior to Engine Number 2 Cutoff

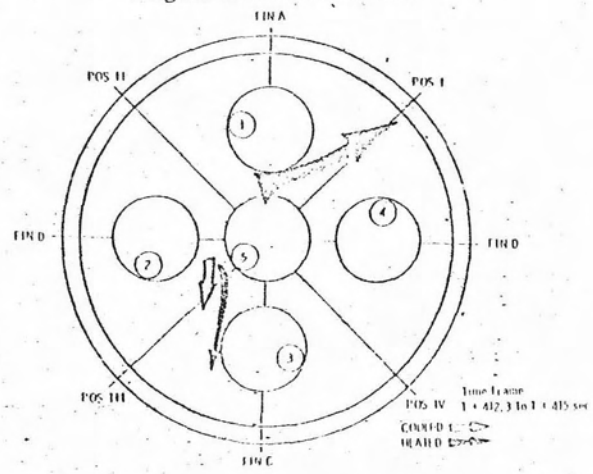

Figure 22. Bottom View of Area Heated and Cooled at Engine Number 2 Cutoff

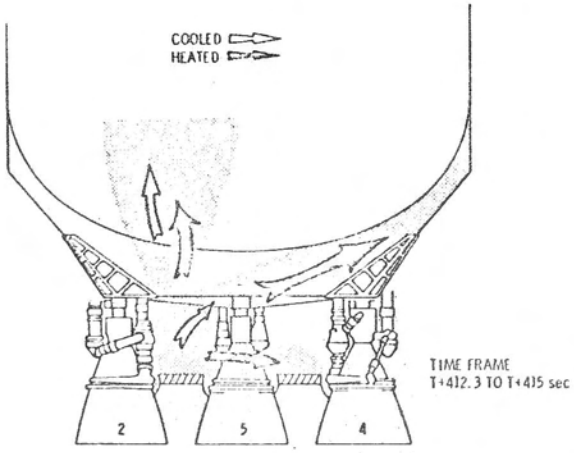

Figure 23. Side View of Areas Heated and Cooled at Engine Number 2 Cutoff

The most probable suspects for causing this type of anomaly is a leak in the augmented spark igniter (ASI) fuel or oxidizer line with the fuel line the prime candidate. (Figures 24, 25, 26, and 27.

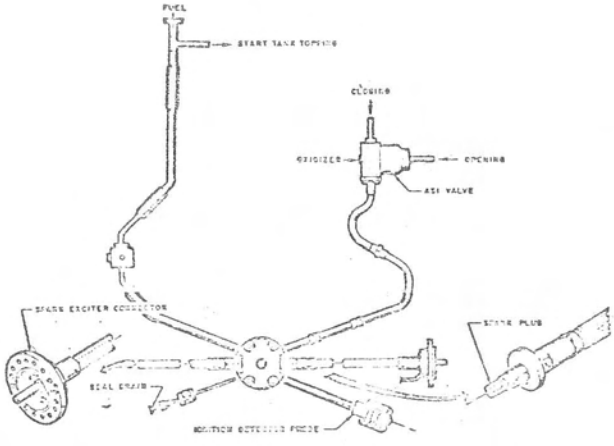

Figure 24. J-2 Engine Ignition System

It has been postulated that a small leak occurred in the line and continued to manifest itself until there was a complete rupture. At this point hot gases from the combustion chamber would backflow through one leg of the ruptured line in the vicinity of the ASI port and result in hot gas spewing out one part of the break and raw cryogenic propellant coming out of the other severed end. This phenomena has not been completely duplicated under controlled conditions at the time this paper was written.

Figure 25. Lower end of ASI Fuel Line

Figure 26. Upper end of ASI Fuel Line

Figure 27. ASI LOX Line

Indicated Yaw Actuator Malfunction. As shown in Figure 28, coincident with a sharp increase in cooling, the yaw actuator ΔP began to increase rapidly and reached a maximum point corresponding to the second thrust shift exhibited by the engine. It was postulated that the indicated malfunction was the result of a cryogenic propellant leak in the vicinity of the actuator.

An actuator pressure transducer was subjected to a stream of liquid nitrogen to simulate a liquid hydrogen leak impingement. The pressure transducer (Figure 29) is composed of two concentric helix bourdon pressure tubes with one attached to the dash pot wall and the other fastened to a sliding rod. As the outer helix tube cooled rapidly it contracted faster than the inner tube and gave an indication of a differential pressure. As the inner helix tube approached the temperature of the outer one, the transducer indicated a null position. When the cryogenic source was removed, the process was reversed with the outer tube heating up faster, expanding, and giving an erroneous differential pressure. As the inner tube temperature approached the outer one a null position was again initiated. It is believed that the cryogenic spray of the actuator ΔP transducer and/or an engine performance shift could explain all the indicated yaw actuator anomalies.

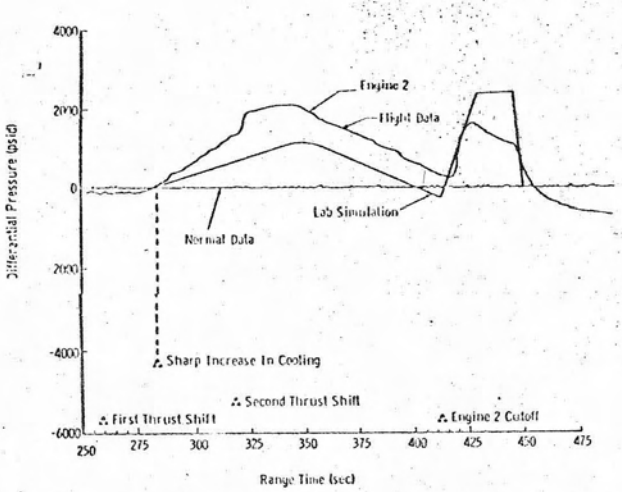

Figure 28. Yaw Actuator ΔP

Performance Shifts. Three performance shifts occurred during the S-II burn. The first shift was gradual starting at 260 seconds after liftoff and continuing for about 60 seconds to a maximum of 6 psi drop. A sudden 20 psi decrease in chamber pressure occurred approximately 319 seconds after liftoff, and a 10 psi drop occurred 0.5 seconds before cutoff of the number 2 engine. As shown in Figure 30, all parameters indicated a drop in thrust at the 319 second data slice which was closely simulated by the math model when programmed for a large fuel leak. A preliminary conclusion is that a leak in the ASI fuel line caused the anomaly. An extensive test program is still in progress to completely verify the cause of the malfunction.

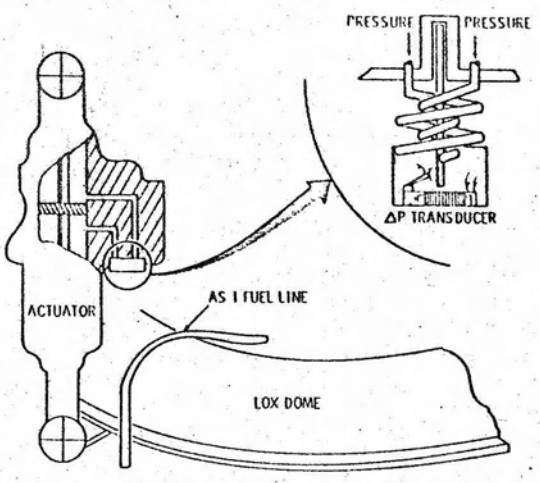

Figure 29. Pictorial Schematic of ΔP Transducer

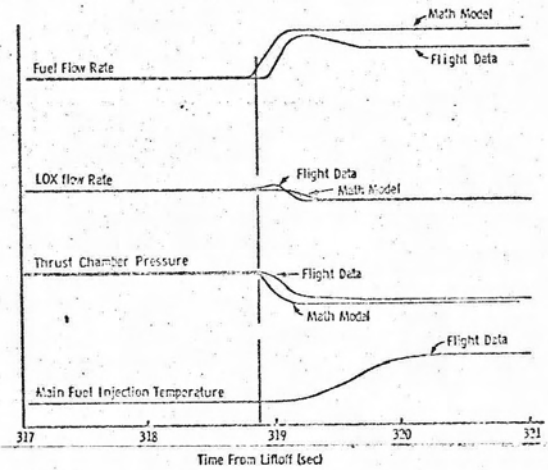

Figure 30. Engine Number 2 Thrust Shift

Premature engine cutoff. Engine number 2 cutoff at 412.8 seconds and engine number 3 cutoff at 414.28 seconds after liftoff which was short of the required mission burn time by 107 seconds and 105.5 seconds respectively. The reason for the number 2 engine cutoff was activation of the mainstage thrust O.K. pressure switch which had decayed to the cutoff level as a result of the malfunction(s). Engine 3 was cutoff as a result of a human error in the stage assembly. When solenoid switches were changed out of the stage the LOX prevalve control command cable on the number 2 engine was erroneously connected to the number 3 engine solenoid and the cutoff signal initiated in the number 2 engine prevalve control circuit shutdown engine number 3.

AS-502 Anomalies (S-IVB Stage) Anomalies which occurred on the AS-502 flight in the S-IVB stage include performance shift during first burn, changes in environmental conditions during first

burn, hydraulic system failure, and failure of the engine to restart on second burn.

The first burn of the J-2 engine was normal up to 695 seconds after liftoff. At that time there was an engine performance shift and a disturbance in environmental conditions.

The chamber pressure dropped, fuel pump discharge pressure increased, both the fuel and LOX injector pressure decreased, the fuel injector temperature rose, and the fuel pump speed went to zero rpm. Figure 31 indicates that both heating and cooling occurred external to the engine during a 15 second time interval.

A sharp drop in the cylinder oil temperature of the yaw actuator was recorded at 695 seconds to correspond with the previously mentioned environmental changes.

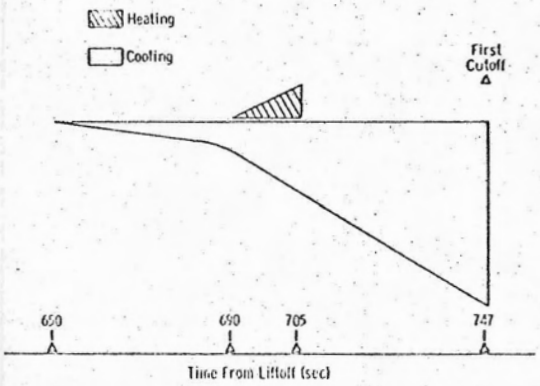

Figure 31. Summary of Environmental Condition Changes

All preconditioning requirements were satisfactory for the second burn and all valves cycled properly. The start transient was normal until the beginning of mainstage except for chamber pressure which failed to rise to the desired level indicating no ignition in the main chamber. The most probable cause for the first burn anomalies and the failure to restart in the second burn is an ASI fuel line failure. Math models produce similar performance shifts for failed ASI fuel lines; however, all malfunctions have not yet been satisfactorily explained. Test programs are underway to demonstrate flight failure modes.

Uprating

Consistent with the pattern of the other large liquid engines developed by NASA, the J-2 engine has been steadily improved since its conception and contributed markedly to vehicle payload increases as shown in Figure 32. Payload capability in the Saturn V vehicle has been increased almost 7000 lbs. through improvements in the J-2 engine. As a result of an intensive simplification program, the J-2 engine is programmed for a demonstration of design in an uprated version in the 1970 time frame.

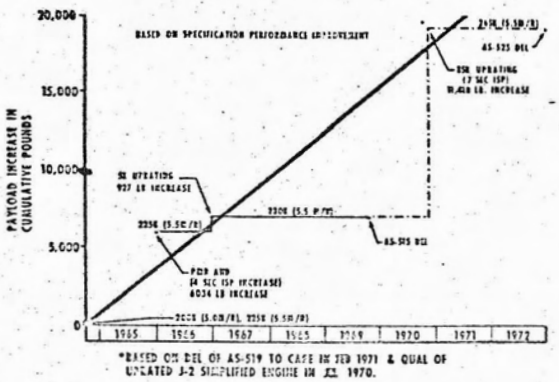

Figure 32. J-2 Engine Contribution to Saturn V Payload Increase

Summary

The J-2 engine development program has encountered problems typical of a new propulsion unit. To date no problem has approached the technical complexity of the combustion instability phenomena found in the LOX-RP-1 engines. The presence of two cryogenics makes the dynamic balance of the propulsion system under a wide operating band a very difficult problem especially during the start transient phase. The basic engine design has been thoroughly evaluated and tested.

Future Outlook

The development of liquid oxygen-liquid hydrogen engines is but another tick mark in the milestones of propulsion. The past trend has been to larger engines but now a plateau seems to be forming. For the near future, developments will probably evolve along more versatility in the system, higher impulse, and more reliability. One of the most promising fields is the low cost booster utilizing either packaged storables or solid propellants.

Acknowledgements

The assistance of the many people who contributed to the preparation of this paper is appreciated. The author would particularly like to thank personnel in the Propulsion and Vehicle Engineering Laboratory and other laboratories at the Marshall Space Flight Center and the many Saturn contractors who contributed data and analyses for the preliminary conclusions drawn in the flight problem area.