Chapter Two

Paul Coffman

Rocketdyne - J-2 Saturn V 2nd & 3rd Stage Engine

Paul Coffman earned experience during many Apollo-era assignments. He served as lead engineer on the J-2 thrust chamber assembly development, supervisor of engineering test for J-2 components and engines, and manager of J-2 engine development and flight support. Following the Apollo era, some of Coffman's program management and business development assignments included serving as director of controls and engineering, program manager of the compact steam generator, engineer coordination for the space shuttle main engine, and project engineer on the gas dynamic laser. He also led successful proposal efforts on an advanced storable propellant spacecraft engine and a system test bed for potential space station propulsion.

The J-2 engine was unique in many respects. Technology was not nearly as well-developed in oxygen/hydrogen engines at the start of the J-2 project. As a result, it experienced a number of "teething" problems. It was used in two stages on the Saturn V vehicle in the Apollo Program, as well as on the later Skylab and Apollo/Soyuz programs. In the Apollo Program, it was used on the S-II stage, which was the second stage of the Saturn V vehicle. There were five J-2 engines at the back end of the S-II Stage. In the S-IV-B stage, it was a single engine, but that single engine had to restart. The Apollo mission called for the entire vehicle to reach orbital velocity in low Earth orbit after the first firing of the Saturn-IV-B stage and, subsequently, to fire a second time to go on to the moon. The engine had to be man-rated (worthy of transporting humans). It had to have a high thrust rate and performance associated with oxygen/hydrogen engines, although there were some compromises there. It had to gimbal for thrust vector control. It was an open-cycle gas generator engine delivering up to 230,000 pounds of thrust.

We delivered 152 production engines. The specific impulse of 425 seconds did represent a small compromise. Chamber pressure was relatively modest, a little over 700 pounds per square inch absolute (psia), and the mixture ratio was 5.5:1 at that chamber pressure. It also had the capability of operating at a mixture ratio 4.5:1. Basic engine weight was relatively light, and if you make the numbers, you will come out a little bit over 80 percent for thrust-to-weight ratio. Propellants were liquid oxygen and liquid hydrogen. Nozzle area ratio was 27.5:1. Because of the difference in densities, there was a requirement for two different turbopumps. They were separate and located, as conventional wisdom would have it, on opposite sides of the engines just for safety sake. The bearing lubrications were liquid oxygen and liquid hydrogen, so there was no separate lubrication system. The gas generator burned main propellants. (See Slides 2 and 3, Appendix D)

As far as the engine operating in the main stage, there was a gas generator that was fed with fuel and oxygen off the main propellants ducts. The gas generator drove the turbomachinery, in a series turbine arrangement, through the fuel turbopump and then over into the oxidizer turbopump, with a bypass for calibration and a heat exchanger to heat up oxygen for tank pressurization (or helium in some instances), before dumping into the thrust chamber at the nozzle midpoint. It was a fully tubular thrust chamber with a fuel inlet at the reduced epsilon[1] portion of the nozzle. It was one tube down for every two tubes up. There was a 2:1 split, and we used the opening at the 2:1 split to dump the hot gas into the nozzle of the thrust chamber. The oxidizer turbopump was a fairly conventional centrifugal device. There was a rather strange shape for the fuel turbopump. The fuel turbopump was an axial machine. It was derived from machines that had been used in the nuclear rocket programs. While that was deemed a sound idea at the start, it resulted in probably the single major development problem – a start sensitivity for the engine. (See Slide 4, Appendix D)

Again, the engine had to restart. It started at altitude on the S-II and S-IV-B first burn and, then, on the S-IV-B restart after one and one-half to two and one-half orbits. The energy for start was supplied by cold hydrogen that had been loaded either on the ground or reloaded from the main fuel injection manifold during the initial burn operation, and it was this start tank that would discharge cold hydrogen through the two turbines to get the engine started. Inside that start tank was a small helium tank because the engine carried its entire helium supply with it in a cold helium tank.

[1] Epsilon refers to nozzle area ratio, exit area to throat area.

Altogether, 152 engines were produced. The engine development, while it may have ceased with the qualification efforts, continued with the F-1 and persisted throughout the entire program. It also spawned a J-2S engine and the Linear Aerospike Engine test beds that were operated at Rocketdyne as well. From the start in 1960, the first 250-second duration test occurred a couple of years later, not a year as the F-1 managed. The 500-second full duration test happened another year after that. We then began a series of tests to demonstrate formally the engine readiness through preliminary flight rating tests (PFRTs). Those were followed by flight readiness tests (FRTs), followed by qualification tests I (qual). Those tests were on the 225,000-pound version of the engine, fifty-nine of which were delivered. Qual II tests came in 1966. Those overlaid with the first production engine delivery, the first flight of the S-I-B, and the first Saturn V flight in 1967, with the moon landing coming in 1969. At the time we landed on the moon, we were very close to delivering the last of the J-2 engines. Perhaps interestingly enough, from a contract standpoint, the engine deliveries were incentivized in the initial contract, so there was a bounty on delivering them on time, even though the capsule-related delays that were experienced by the Apollo Program caused them to be delivered to warehouses in many cases. (See Slides 5 and 6, Appendix D)

This was a schedule-driven program; cost was no object.

We had thirty-eight development engines. This was not a hardware-poor program. This was a schedule-driven program; cost was no object. We had 1,700 tests through qualification. That was the qual II series. We continued testing and had over 3,000 tests by the time the program ended. We had multiple test facilities and a lot of component facilities as well. We had five engine test stands at Rocketdyne Santa Susana in California, including one that was a simulated altitude facility. We had test facilities at NASA's Marshall Space Flight Center in Alabama. We had test facilities at NASA's Stennis Space Center in Mississippi, and one at Sacramento, California, for the S-IV-B. We also did some very significant testing at the J-4 cell at the Arnold Engineering Development Center (AEDC) in Tennessee. That was significant because that was where we finally resolved the start problems that were the major issue with the engine. (See Slide 7, Appendix D)

The engine was required to complete thirty tests and 3,750 seconds of operation for the formal demonstrations. If you added up all of these numbers, it came to something over forty tests because there were more than one at a time. Indeed, we managed to do that to the point that some of them were just start-stop tests as we actually finished the program. The most significant ones were what we blandly called "the safety limits tests." Those were the extremes of the boxes for start energy and build up energy. The actual demonstration engine was Engine 2072, which did all but the last altitude simulation test. The program office was very, very conscious of, and was not comfortable with, the fact that we ran 57.4 seconds over the requirement.

There were four key development issues. (See Slide 8, Appendix D) The first was thrust chamber ignition detection. It was a good idea to have the thrust chamber lit before the main propellants were added. The thrust chamber was ignited by an augmented spark igniter in the middle with two spark plugs that used fuel and hydrogen and oxygen. It also had a spot for an ignition detection device. The development of ignition detection devices was quite significant in that the simplest device was a fusible link. Burning the link showed there was a fire. This process was used for most ground tests, except at AEDC, an Air Force-owned test site, where we wanted to run multiple tests within a given air-on period, or vacuum period. At this point, it seemed advisable to have an ignition detection device for all tests subsequent to the first one. There was also an idea that we would like to detect ignition on flight engines as well, since a reusable probe underwent a lot of development. That was used for the AEDC engine testing, but it never demonstrated enough reliability for flight. The net result was that I don't believe there was ever a ground test conducted without an ignition detection device, but none of the engines were flown with one. *That's a point to consider as we move forward to a new crew launch vehicle*[2]. (See Slide 9, Appendix D)

The engine start was a significant development issue, and one we didn't do tremendously well.

The engine start was a significant development issue, and one we didn't do tremendously well. I think Dr. Richard Gilbrech[3] mentioned that, in this era, the analytical capabilities and modeling were somewhat limited. That was probably a very positive statement, but a sensitive one as well. There was a very strong potential for gas generator burnout if you didn't tiptoe through the start transient very carefully. There were several components that contributed heavily to that sensitivity. The first was the axial fuel turbopump. Fuel turbopumps of that configuration have a tendency to stall if there's too much downstream resistance. That required a great deal of thrust chamber assembly thermal conditioning on the ground beforehand, and a variation in fuel lead time, depending on how long it had been since it had been conditioned on the ground, in order to make sure that it was not too cold or too warm.

A complex two-position main oxidizer valve was utilized. It featured a pneumatically actuated butterfly valve. To operate, the valve had to be moved from the closed position to a fourteen-degree position and the flow had to start to begin the engine. It had to dwell there for about one-half second, then go to the full-open position required during one and one-half seconds. It is characteristic of butterfly valves, however, that if the flow force increased too rapidly on them, they would just stop moving, unless the actuator force was sufficient to move them. Now, if the actuator force was pneumatic, a force balance issue occurred.

[2] Italicized text represents "lessons learned" by the conference presenters.
[3] Dr. Richard Gilbrech served as director of Stennis Space Center from January 2006 through August 2007. He recently returned to Stennis Space Center as associate director.

The third issue was the turbopump start energy from the high-pressure cold hydrogen. The dump of that energy had to be precisely accommodated, and it had to be both the requisite temperature and pressure in order to get the satisfactory start energy. That process was controlled by the start tank vent relief valve. People have wondered why we used two valves. Well, we knew there was going to be a warm-up of the cold hydrogen during the boost phase, and we wanted to compensate for that. We had a big valve that was going to dump immense quantities of cold hydrogen, and we had a leak rate that had to be calibrated as well. With all of these issues combined, extensive thermal conditioning was needed. We were conditioning a thrust chamber with cold helium on the launch pad. (See Slide 10, Appendix D)

The initial problems were encountered at Santa Susana and at Marshall. The simulated altitude testing at the Santa Susana Vertical Test Stand 3, 3A position, was a small capsule. The engine was mounted horizontally, and it had a steam ejector to pull the initial altitude. It really didn't do a very good job of simulating the start transient with the spit-back. When we started flying the S-I-Bs, *which shows the wisdom of flying early if you possibly can*, we indicated the problem was not very well understood, and certainly was not resolved. Bottom line: we started putting immense activity into AEDC's J-4 cell. For those who are not familiar with it, there was a hole in the ground about 100 feet in diameter and 300 feet deep. We could attach a very strong vacuum to it. There was a huge bell jar configuration metal container, inside of which very easily fit an S-IV-B stage with a J-2 attached. We did a year and one-half worth of testing there and, concurrent with that, fruitfully advanced our analytical modeling capability, both at NASA and at Rocketdyne. The net result was that we were able to achieve satisfactory test starts on all the ground and flight applications.

The engine started relatively slowly. It could go from 10 percent to 80 percent in about one and one-half seconds. Although it was relatively slow, it was quite repeatable. The bandwidth was fairly repeatable for 152 engines, which was pretty tight, especially considering that the inlet conditions to the engine depended to some extent on the stage application. Again, we were able to replicate this and advance the cause by advancing our analytical modeling. (See Slides 11 and 12, Appendix D)

Another interesting little problem we encountered was high transient sideloads. We realized that to get at a twenty-seven and one-half epsilon nozzle and still facilitate testing at sea level conditions, it was going to take some doing. That was a goal. We used a conventional nozzle, and at a 27.5:1 ratio, it was below the three-tenth level. We assumed we were going to have full flow in the nozzle. We wanted to accomplish that, so we developed an adverse pressure gradient, or APG, nozzle that would be back up over that range as a result of over-expansion in the nozzle. There was about a two- to three-second specific impulse performance loss attributed to that. We proved the concept with both analysis and model hot fire testing.

We had tenth-scale, solid-wall nozzles of several configurations that were evaluated before the APG configuration was chosen. However, once we started testing, we noticed some fairly high sideloads during start transients. Similar sideloads could occur also when thrust was reduced by changing the propellant utilization to run at the 4:5 mixture ratio and dropping the thrust level about 20 percent. The thrust chamber damage that we incurred was rather dramatic. The attach points for the actuators on the J-2 were at the upper end of the engine, and the clevis pins started elongating the holes. That may have been due to the fact that we were putting more than 100,000 pounds of pressure through them, and they weren't designed for that. (See Slides 13, Appendix D)

The second thing was spotted by an associate of mine in combustion devices at the time. He stopped to tie his shoelaces after he looked at the thrust chamber to make sure the injector was okay. When he happened to glance back over his shoulder, he noticed the exit of the thrust chamber had a unique configuration. It was no longer straight; it was concave, and he said to himself – being a MIT grad – "There's something wrong there." There's just no discounting a good education. The net result was a panic because the conventional diffuser to attach to the chamber was going to be approximately twenty feet long, and that wasn't going to fit real well into the test stands of the day. The original configuration had a very clever turnout at the tail end of the tubes. We began with the subscale model that had been used to define the contour in the beginning, and we started testing, first with wood, then eventually with metal. We finally settled on something that was about a quarter of an inch long on the subscale model – six inches long on the actual full-scale nozzle. That eventually evolved into a water-cooled version. We reinforced the hat-bands by cutting the top out of them, and welding in tubing. We also attached clevis pins to the area where the fuel injection and hot gas manifolds, which allowed us to tie into a sideload attachment mechanism, or SLAM, system. (See Slides 14, Appendix D)

There was a strange device hanging on the backend of the chamber. We put it on to allow us to get through start transient. We "belt-and–suspendered" it for both single and multi-engine ground tests, which solved the problem with no further difficulty. We needed to have the capability for this to release during main stage, so we could demonstrate gimbal. If we didn't have to demonstrate gimbal, we would usually just leave it attached at all times. As an aside, the initial version of this cheap metal monstrosity was welded to a combustion devices chamber on a test stand in the area, a pressure-fed test stand. The first test had been quite satisfactory. We planned on coming back the next day and testing it more. However, when we got back, there was nothing but a jagged edge. *The engine systems people had been over and cut it off and it now was welded on the engine next door.*

Now, the final major issue of the J-2 came when the Apollo 502 vehicle was launched in 1968. It was the second launch of the Saturn V vehicle. It was unmanned and was planned to be the last unmanned launch. One of the five J-2 engines cut off prematurely, and the S-IV-B engine operated apparently satisfactorily on the first burn, then failed to restart. We really went after that one. Flight data indicated there was probably a similar failure mode that caused both malfunctions. Hard as it was to believe, there were certain similarities in the engine area temperatures and in the engine operating characteristics that led us to believe that it was very likely the case. What we wanted to do was verify the anomalies with ground tests, define the failure mechanism analytically at the same time, verify it, and *then please, please, please incorporate corrective action, as soon as possible.* (See Slides 15 and 16, Appendix D)

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A good example of this was the S-IV-B stage timeline. The engine experienced five engine tests and two stage acceptance tests prior to flight, and it had been pretty nominal. We saw engine chilling beginning about sixty-five seconds into flight. Then, there was a heating that began about forty seconds after that, and we got into chilling again during the restart. We saw very small performance shifts of about 4 psi and 12 psi. We also saw a little bit of change in the required thrust vector control. There was a slight shift in actuators that indicated something changed in the nozzle. Everything else was dead nominal. As a matter of fact, we noticed that we turned loose the start tank discharge valve and waited a number of seconds before there was a second burn cutoff initiative because it didn't go.

We started looking at the augmented spark igniter. The fuel line to the augmented spark igniter had a single-ply bellows, and we said, "Feed until that bellows fails." The idea was if that bellows failed, it would account for burnout of the augmented spark igniter, and that would certainly mirror, or have the possibility of mirroring, the chilling and heating that we observed. We set up an engine at Santa Susana to investigate that. We planned a sequence of events where we would have sixty-five seconds of normal operation, then we would incur a small augmented spark igniter fuel leak. We would increase the leak subsequently to complete the failure and allow backflow of the augmented spark igniter combustion products for the final thirty seconds of the operation. I had never been so happy to see copper in a flame of a rocket engine in my whole life. It went right down the line. There was substantial erosion of the device. The cavity burned out. The performance loss was similar and so was the thrust vector change. We figured we had it nailed. Within a day at Marshall Space Flight Center, the S-2 engine that had been set up and tested showed the same characteristics, so we figured we had done a good job in duplicating the failure on the ground. The analysis did indicate that the single-ply bellows could fail and that probably liquid air, since there was hydrogen passing through it on the ground, could damp the oscillations. That hypothesis was tested on similar

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bellows both with inert fluids and liquid hydrogen, and showed the failure. Some people have asked why the bellows were there in the first place. Well, our analysis initially indicated the bellows was a wise idea because of thermal changes that were going to occur during engine operation. So, the bellows were put in to compensate for that. In hindsight, a good idea turned out to be a bad idea. We got rid of the design modification. Solid-line Class 1 welds, and the revised configuration was qualified and incorporated on all the engines in the field. Eight months later, the first manned Apollo 503 flew – a pretty good response to the issue. (See Slides 17 and 18, Appendix D)

We began this project in 1960, and it was schedule-driven. It was what we would characterize today as a low technology readiness level. The thermodynamic characteristics of para-hydrogen were under investigation by the National Bureau of Standards[4]. Given the criteria, the development program was quite satisfactory: two years to the first 250-second test, formal development completed in six years. The J-2S and linear spawned off from that, to investigate and keep people sharp. There were over eighty modes of unplanned cutoffs. The engine and the analytical tools utilized to develop it enabled us to be satisfactory on the space shuttle main engine, the RS-68 engine, the X-33 engine, and what is now being characterized as the J-2X engine. (See Slide 19, Appendix D)

[4] Currently known as National Institute of Standards and Technology.

Editor's Note: *The following information reflects a question-and-answer session held after Coffman's presentation.*

STEVE FISHER[5]: I'd like to introduce Mr. Manfred "Fred" Peinemann, a guest of The Aerospace Corp. Please stand up, Fred, so we can pinpoint you while you ask your question.

QUESTION: Did you encounter interesting materials compatibility problems during the development with the J-2? Did you encounter hydrogen embrittlement?

COFFMAN: We didn't encounter hydrogen embrittlement. The first problem we encountered was the first summer we started trying to do engine testing. We found that we couldn't get liquid hydrogen through the engine because the facility dump system and the engine bleed valves weren't big enough to allow it in a Santa Susana summer[6] to achieve liquid at the inlet to the engine.

QUESTION: This goes basically to both F-1 and J-2. Can you comment on the philosophy when you had failures and anomalies? How fast you turned those around? From what we can tell, it looks as though you were continually testing. You didn't stop and have failure investigation teams, and factor all that back in. How did your process for getting that corrective action work into the design occur?

COFFMAN: It is true that we were constantly testing, but there was a formal Unexplained Condition Report (UCR) system. When a component failed in an engine test, or in a component test, the UCR was generated. The first thing we had to determine was whether the latest configuration had failed, or whether it was something else that had already been replaced on the engine and the failure wasn't useful. There was enough hardware richness that we had five engine test stands, running two shifts. I remember one of the technicians looked up and said, "Wow, Friday. Only two more work days until Monday," which characterized the era pretty well. The attempt was to incorporate any corrective actions immediately, and put out kits for retrofit of everything in the field. The logbooks of which kits applied to which engines were pretty interesting. But, certainly, we had a very strong cadre of developmental personnel, at both the component and engine system levels, and no problem was ignored.

[5] Steve Fisher served as facilitator during the *On the Shoulders of Giants* seminar series.
[6] Meaning the hot climate in north Los Angeles desert.

BOB BIGGS: I think for major problems we have always - we did in the F-1 program, we do in space shuttle main engine program - established an investigation team with an autonomous team leader to run the investigation. We assigned on the order of twenty to thirty people to work the problem, and it is generally totally resolved within thirty days. Today, we would not test for a period of time. I think on F-1 that was rare. We would continue testing while the investigation was going on.

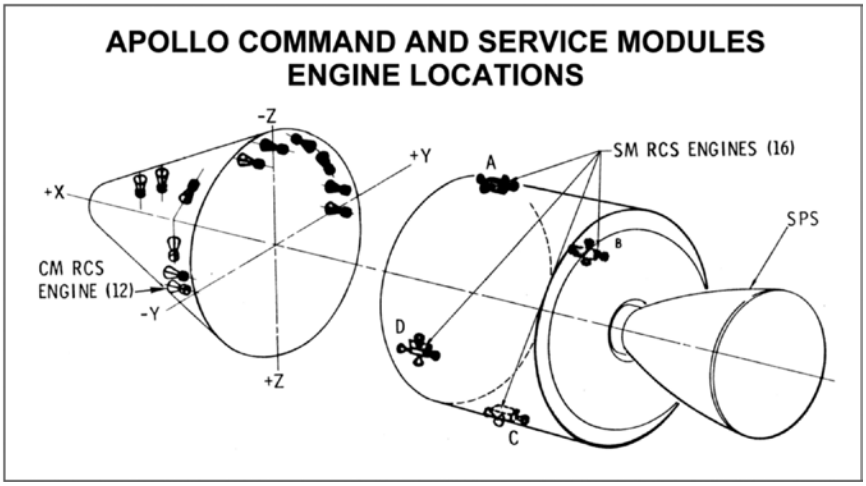
QUESTION: Paul, from your talk, it appeared that you did a lot of testing over at Santa Susana and, obviously, we all know that the facility is not in great shape right now. You mentioned you had as many as five test stands going at the same time. Do you have any recommendations for as to how to get prepared for those tests?

COFFMAN: I think you are starting from a much more advanced position with respect to looking at developing future engine and stages. You also are not pushing the envelope nearly as much. So, I think that from what I have seen of the test planning for the J-2X, for example, I think that the group is well-connected between Marshall Space Flight Center and Rocketdyne in some of the net meetings where I have been an observer to make sure they do a thorough job of testing. Now, the question of "thorough" will always have different meanings to different people. *I would encourage, until we are very satisfied with the start characteristics of that engine at altitude, that we keep open the vacuum capabilities.* I think the test planning is headed in that direction.

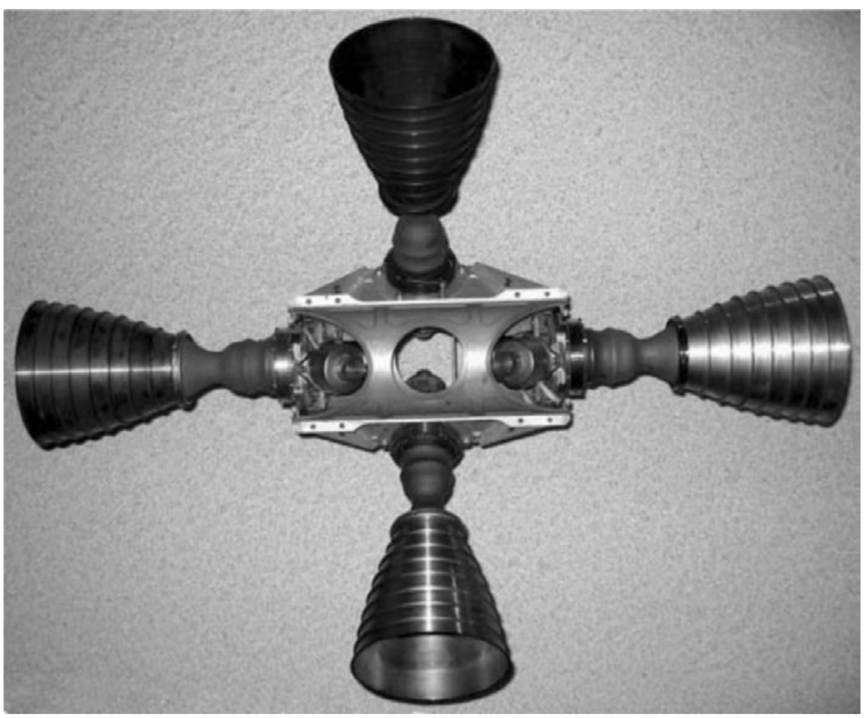
FISHER: This is my opportunity as a moderator to throw in my two cents. I think, as Paul mentioned, we have come a long way. Our analytical tools, especially our transient modeling and understanding of the processes, are much better than they were. However, I think that with some of our recent successes in engine programs, we have done very well. We tend to get a little bit optimistic on a no-failure plan. I think one of our weaknesses over the years is we haven't planned for failure. If you are in a program, and you have a lot of extra hardware, and you have money to resolve it like we did back in those days, that is one thing. In today's world, where we plan for every engine test being a success, that may rise up to bite us. *So, for my two cents, a little bit of reserve is required for those failures that are unknown. It would be nice if we could carry that. I don't know how you go get that.*

QUESTION: I'm Thomas Carroll. I work systems for J-2X testing. Were there any compromises that were made for any of the previous J-2 programs, other than the planning for failures? Are there any compromises that were made that we are looking at fixing for J-2X that we want to fix, that we want to change, and want to make better?

COFFMAN: I don't think so, at least as the engine is evolving now. There seems to be a rather thorough study of the "S" (i.e. J-2S) and the basic J-2. You must recognize that the "S" was commissioned to be physically and functionally interchangeable, so it had to meet the 80-inch diameter, and the same length, and the same attach points, and it had to fit the same buildup characteristics, and so forth. There may be some real possibilities for engine definition for the "X" (i.e. J-2X) in coordination with the stage to be mutually advantageous, but I don't have any concrete examples of compromises that would cause a problem going forward.



Apollo Spacecraft/LM Adapter
from Apollo Training Manual "Apollo Spacecraft & Systems Familiarization"
(March 13, 1968)



Apollo Reaction Control System Engines

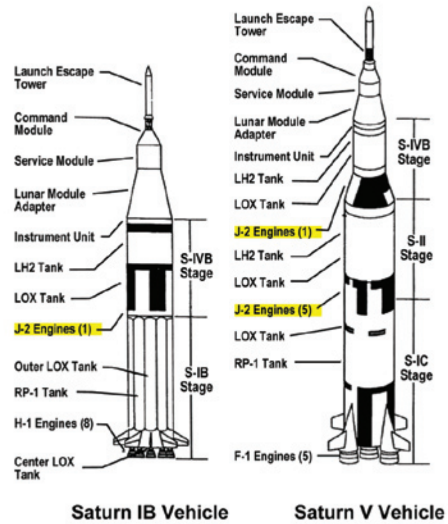
Appendix D

Paul Coffman's Presentation Viewgraphs

The J-2 Engine

APOLLO 11

- Requirements
  - Man rated
  - High thrust / performance
  - Multiple space starts
  - Gimbal for TVC
  - Application to both SIVB & S-II stages
- Usage on Apollo, Skylab & ASTP
  - SIB/SIVB – 9 flights
  - SV/SII – 13 flights
  - SV/SIVB – 12 flights



Rocketdyne

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J-2 Basic Engine Features

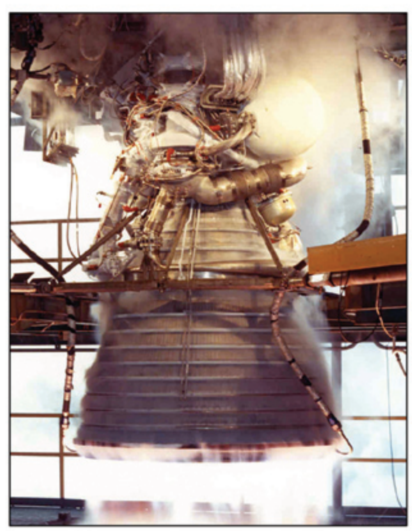
APOLLO 11

Performance & Weight

Nominal vacuum thrust (lb)	230,000
Nominal vacuum specific impulse	425
Chamber pressure (psia) (nozzle stagnation)	717
Engine mixture ratio calibration (O/F)	5.5:1
Basic engine dry weight (lb)	2,754
Engine dry weight (lb) (including accessories)	3,492

Description

Pump-fed, liquid-propellant rocket engine
Propellants: liquid oxygen & liquid hydrogen
Nozzle area ratio: 27.5:1
Tubular-wall thrust chamber, regeneratively cooled
Separate oxidizer & fuel turbopumps
Bearing lubrication: liquid oxygen & liquid hydrogen
Turbine drive: gas generator burning main propellants



34 Flights, 86 Engines, 98% Reliability

Rocketdyne

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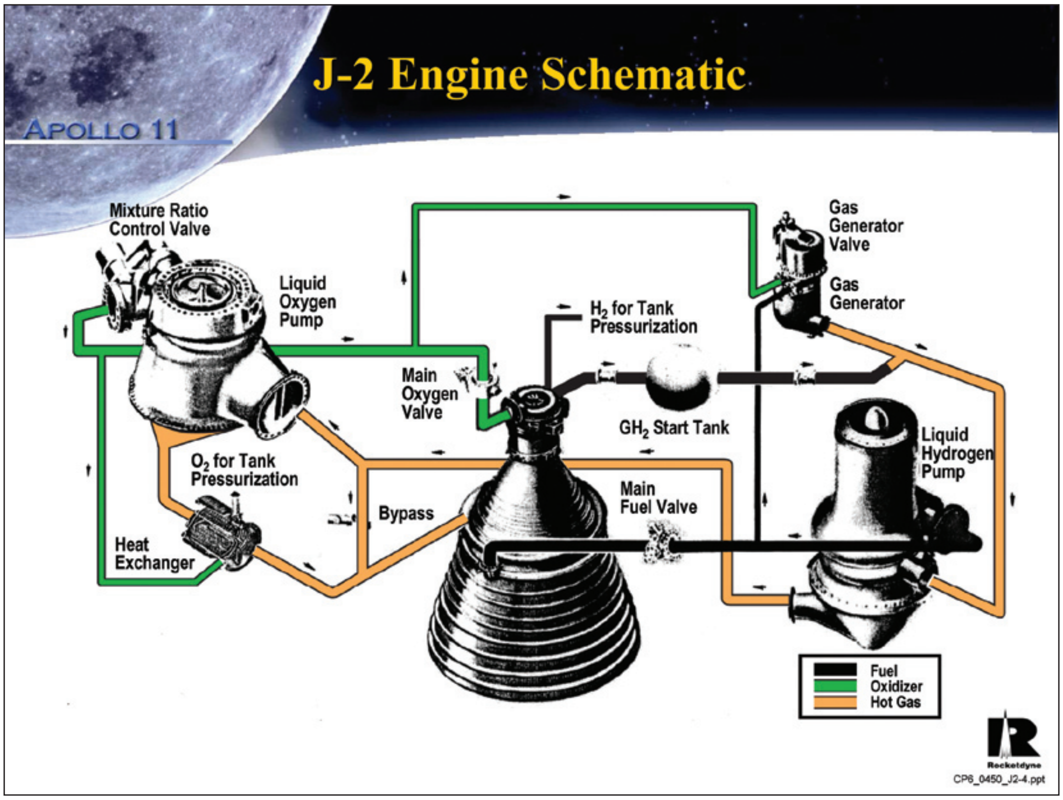
J-2 Engine Schematic
APOLLO 11
Mixture Ratio Control Valve
Liquid Oxygen Pump
Gas Generator Valve
Gas Generator
H_2 for Tank Pressurization
Main Oxygen Valve
GH_2 Start Tank
O_2 for Tank Pressurization
Bypass
Main Fuel Valve
Liquid Hydrogen Pump
Heat Exchanger
Fuel
Oxidizer
Hot Gas
Rocketdyne
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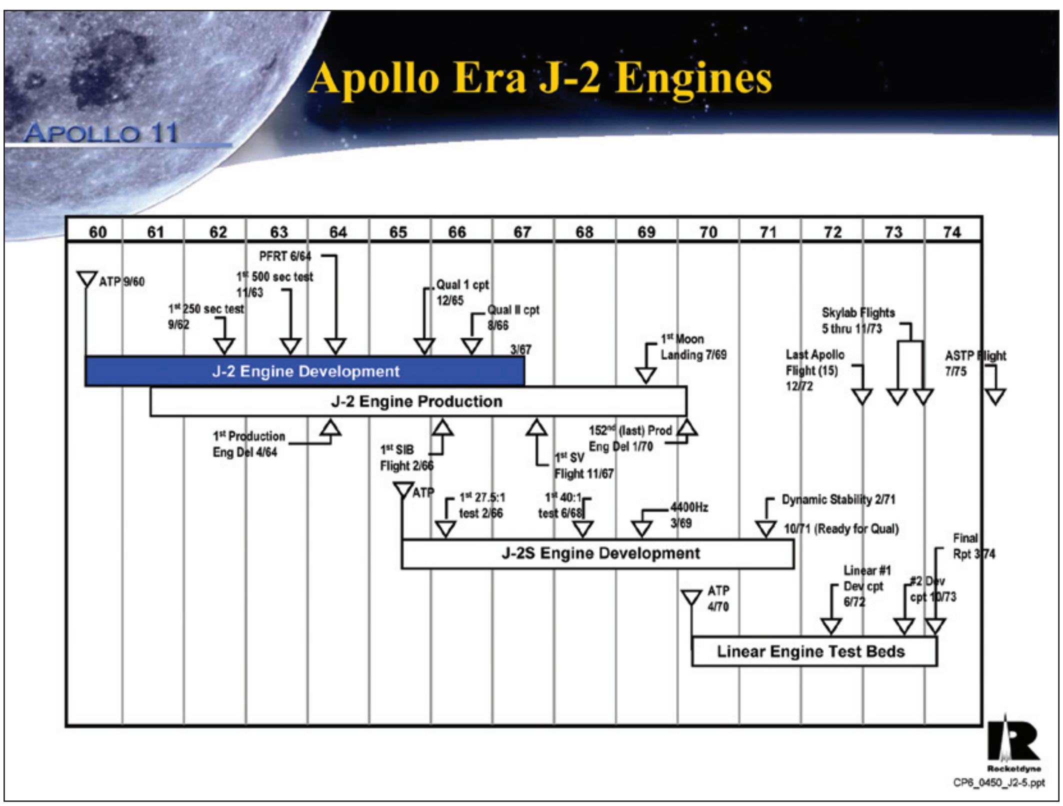
Apollo Era J-2 Engines
APOLLO 11
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ATP 9/60
1st 250 sec test 9/62
1st 500 sec test 11/63
PFRT 6/64
Qual 1 cpt 12/65
Qual II cpt 8/66
3/67
J-2 Engine Development
J-2 Engine Production
1st Production Eng Del 4/64
1st SIB Flight 2/66
1st SV Flight 11/67
152nd (last) Prod Eng Del 1/70
1st Moon Landing 7/69
Skylab Flights 5 thru 11/73
Last Apollo Flight (15) 12/72
ASTP Flight 7/75
ATP
1st 27.5:1 test 2/66
1st 40:1 test 6/68
4400Hz 3/69
Dynamic Stability 2/71
10/71 (Ready for Qual)
J-2S Engine Development
Final Rpt 3/74
ATP 4/70
Linear #1 Dev cpt 6/72
#2 Dev cpt 10/73
Linear Engine Test Beds
Rocketdyne
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J-2 Engine Development

APOLLO 11

- Development engines - 38
- Engine tests – 1730 through qualification
- Engine (and stage) test facilities
  - Rocketdyne/ Santa Susana Field Lab
    - Including one (1) simulated altitude facility
  - NASA / MSFC
  - NASA / MTF (now SSC)
  - USAF / AEDC simulated altitude facility

Rocketdyne
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J-2 Engine J2072 Qualification Program

APOLLO 11

	Requirement	Demonstrated
Endurance Tests		
Total Tests	30	30
Total Seconds	3750	3807.4
Total Duration At High Mixture Ratio	1500	1503.9
Operating Limits and Performance Tests		
Programmed Mixture Ratio; 470 Second Duration Tests	3	3
Gimbaling Pattern Cycles	8	8
Mixture Ratio Control Valve Calibration Point Tests	5	5
Mainstage Performance Tests	4	7
Heat Exchanger Pressurization Performance Point Tests	2	2
Hydrogen Tapoff Pressurization Performance Point Tests	1	1
Safety Limits Test	16	16
Start-restart Test Couples	2	2
Demonstration Of Restart Capability Tests	1	1
Start-stop Tests	As necessary	5
Altitude Simulation Test (Engine J-2073)	1	1

Rocketdyne
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Selected J-2 Engine Development Issues

APOLLO 11

- Thrust chamber assembly ignition detection
- Engine start transient
- Engine sideloads
- AS 502 failure – ASI fuel flex line

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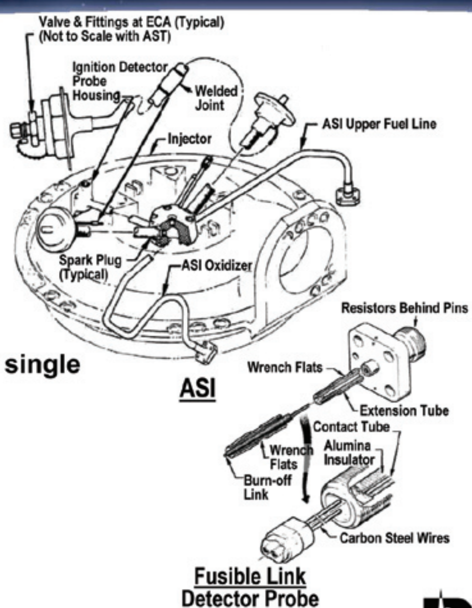
J-2 Engine TCA Ignition Detection

APOLLO 11

- TCA ignition accomplished by ASI
  - Spark plugs
  - Hydrogen & oxygen
  - Ignition detectors developed
- Fusible link detector
  - Used on most ground tests, except at AEDC
  - Satisfactory reliability, but limited to single use
- Reusable IR detector
  - Used at AEDC for multiple testing
  - Unsatisfactory reliability for flight
- Result: ALL ground tests used ignition detector, NO flights used one!!



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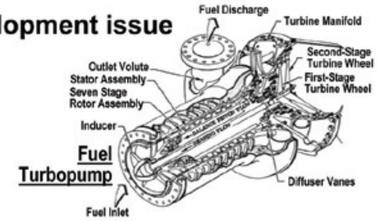
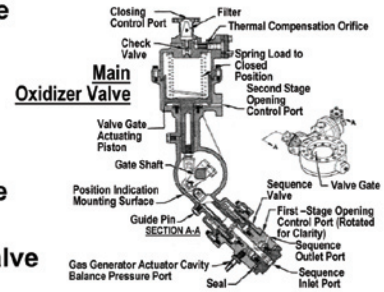
J-2 Engine Start Transient

APOLLO 11

- Engine start sensitivity – <u>THE</u> major J-2 development issue
  - Potential gas generator overtemp and fuel turbine burnout
- Contributing component characteristics:
  - Axial fuel turbopump stall
    - TCA thermal conditioning
    - Fuel lead time variation dependent on use
  - Complex main oxidizer valve pneumatic actuator operation
    - Two position
    - Extended ramp to full open
  - Turbopump start energy from high pressure cold GH_2
    - Controlled by start tank vent and relief valve
    - Secondary Helium temperature effect
  - Extensive thermal conditioning and sequence control required





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J-2 Start Transient Development

APOLLO 11

- Initial problems encountered during ground test
  - SSFL and "sea level" test sites
  - Simulated altitude testing at SSFL/VTS-3A addressed the issue
    - Small capsule
    - Steam driven hyperflow ejector
  - Early S-IB flights indicated problem <u>NOT</u> resolved (or well understood)
    - Additional simulated altitude test conducted at AEDC/J-4
      - Large vacuum facility
    - Extensive analytical modeling capability developed
      - Facilitates modern engine development

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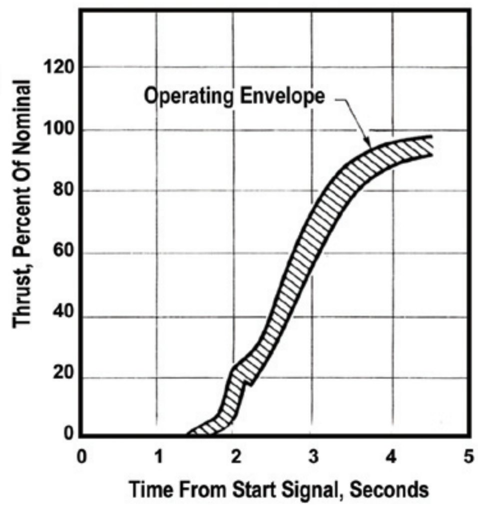
APOLLO 11

J-2 Start Transient Control Achieved

- **Satisfactory starts on all ground and flight applications**
  - **Relatively slow (3.5 – 4 sec to 90% P_c)**
  - **Repeatible, with narrow bandwidth**
- **Analytical model development**



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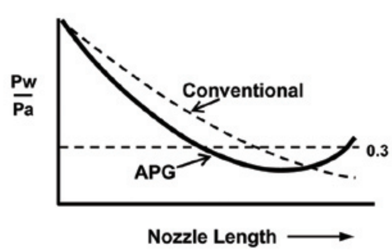
APOLLO 11

J-2 Engine Sideloads

- **Unique J-2 Thrust chamber nozzle configuration**
  - **Adverse pressure gradient (APG)**
    - **Enable "sea level" operation with full flow 27.5:1 nozzle**
    - **Facilitate development**
  - **Acceptable minor performance loss**
    - **Analysis**
    - **Model hot fire test**
  - **Unstable separation and high sideloads experienced**
    - **During start transient operation**
    - **During low thrust (PU) operation**
    - **Thrust chamber damage incurred**
      - **Actuator arm attach points (>100Klb loads)**
      - **Nozzle deformation**



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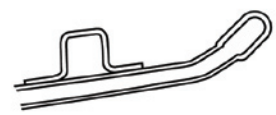
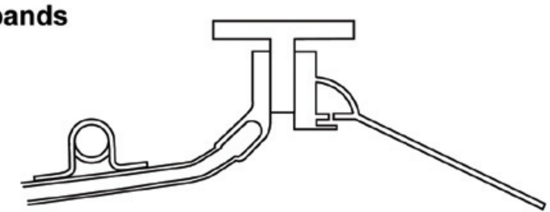
J-2 Engine Sideload Resolution

APOLLO 11

- Water cooled bolt-on "diffuser" developed
  - Classic diffuser >20ft long!
  - Subscale models (wood & metal) tested
  - Uncooled full scale verification
    - Six (6) inch length
- Strengthened nozzle
  - Tubing welded into hat bands
  - "T-ring" bolted to exit
- Restrained thrust chamber
  - Side Load Attach Mechanism (SLAM)
  - Release to allow gimbal operation



Original Configuration



Evolved Configuration

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J-2 Engines on AS 502

APOLLO 11

- AS 502 launched April 4, 1968
  - Second launch of Saturn V vehicle (unmanned)
  - One of the five SII engines cutoff prematurely
  - The SIVB engine did not restart after its initial burn & nominal coast
- Fight data analysis showed significant anomalous operation
  - Similar failure mode projected for <u>both</u> engines!
- Expedited effort to:
  - Verify observed anomalies with ground test simulations
  - Define the failure mechanism analytically
  - Incorporate corrective action for subsequent Saturn flights

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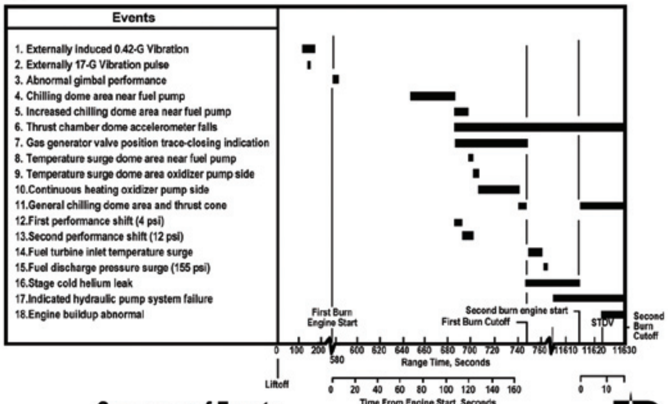
Engine J2042 on AS 502 SIVB Stage

APOLLO 11

- Engine operation nominal during pre-launch operation (704.6 sec mainstage)
  - Five engine and two stage acceptance tests
- Flight data anomalies observed
  - Engine area temperatures
    - Chilling at 645 sec
    - Heating at 696 sec
    - Chilling at restart
  - Performance decay
    - Starting at 684 sec
  - Restart failure despite
    - Proper conditions
    - Valve operation
- Failure mode hypothesis
  - Small ASI fuel bellows leak at 645 sec, with failure at 696 sec
  - ASI burnout



Sequence of Events

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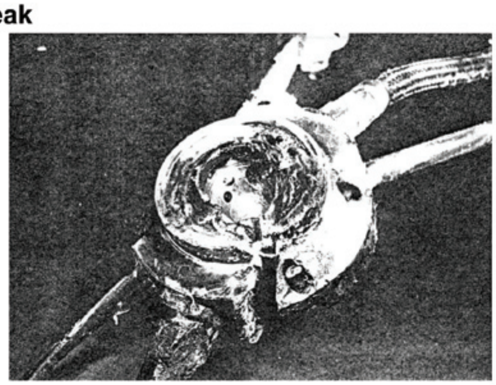
Failure Simulation

APOLLO 11

- Engine J016-4 configured to simulate J2042
  - Test equipment added to allow ASI flow control & overboard dump
  - Sequence of events planned:
    - 65 sec of normal operation
    - 35 sec of 0.6 lb/sec ASI fuel leak
    - Increase leak to complete line failure
    - Backflow of ASI combustion products for 29 sec
- Posttest evaluation
  - Substantial main injector ASI cavity erosion
  - ASI burnout
  - Performance loss similar to J2042 experience
- Similar test demonstration duplicated SII anomaly



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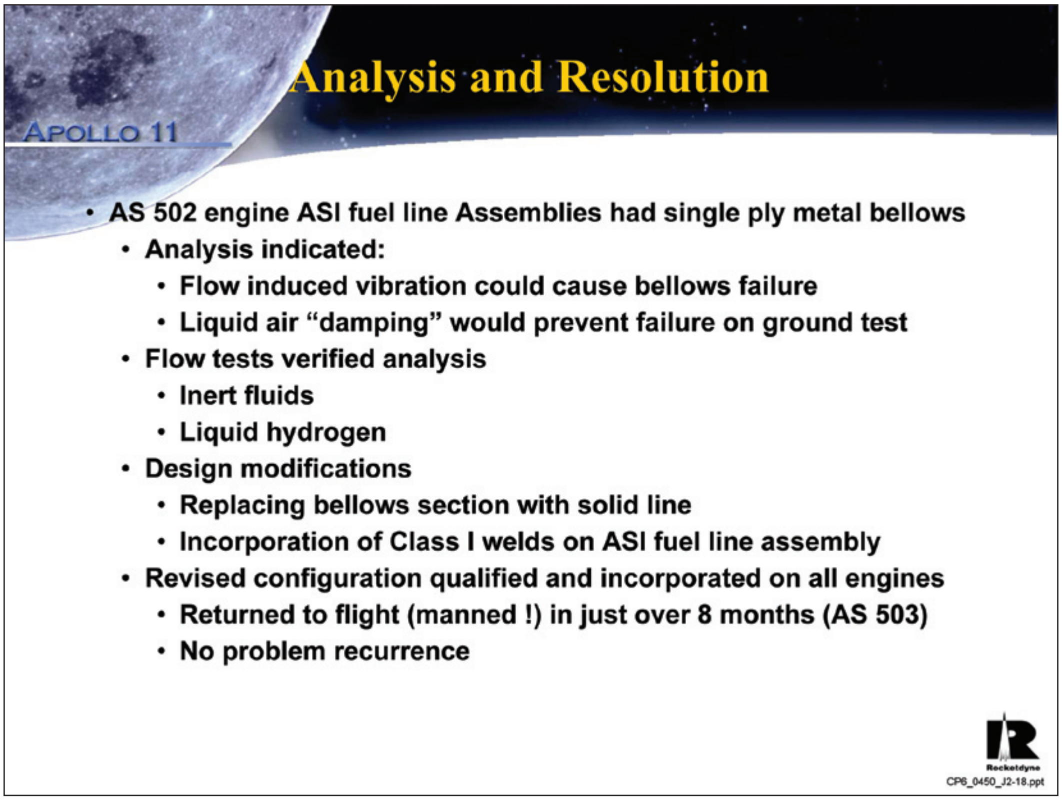
Analysis and Resolution
Apollo 11
• AS 502 engine ASI fuel line Assemblies had single ply metal bellows
• Analysis indicated:
• Flow induced vibration could cause bellows failure
• Liquid air "damping" would prevent failure on ground test
• Flow tests verified analysis
• Inert fluids
• Liquid hydrogen
• Design modifications
• Replacing bellows section with solid line
• Incorporation of Class I welds on ASI fuel line assembly
• Revised configuration qualified and incorporated on all engines
• Returned to flight (manned !) in just over 8 months (AS 503)
• No problem recurrence
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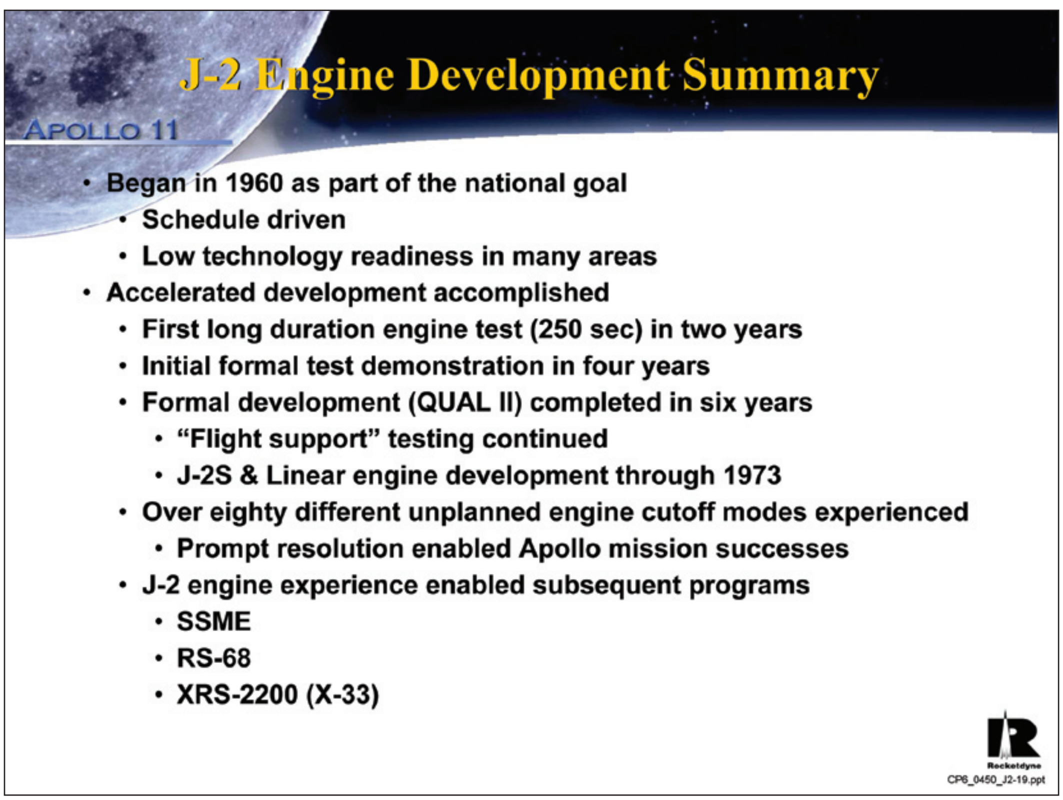
J-2 Engine Development Summary
Apollo 11
• Began in 1960 as part of the national goal
• Schedule driven
• Low technology readiness in many areas
• Accelerated development accomplished
• First long duration engine test (250 sec) in two years
• Initial formal test demonstration in four years
• Formal development (QUAL II) completed in six years
• "Flight support" testing continued
• J-2S & Linear engine development through 1973
• Over eighty different unplanned engine cutoff modes experienced
• Prompt resolution enabled Apollo mission successes
• J-2 engine experience enabled subsequent programs
• SSME
• RS-68
• XRS-2200 (X-33)
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