VII. CASING JOINT DESIGN

Discussion

(a) Introduction

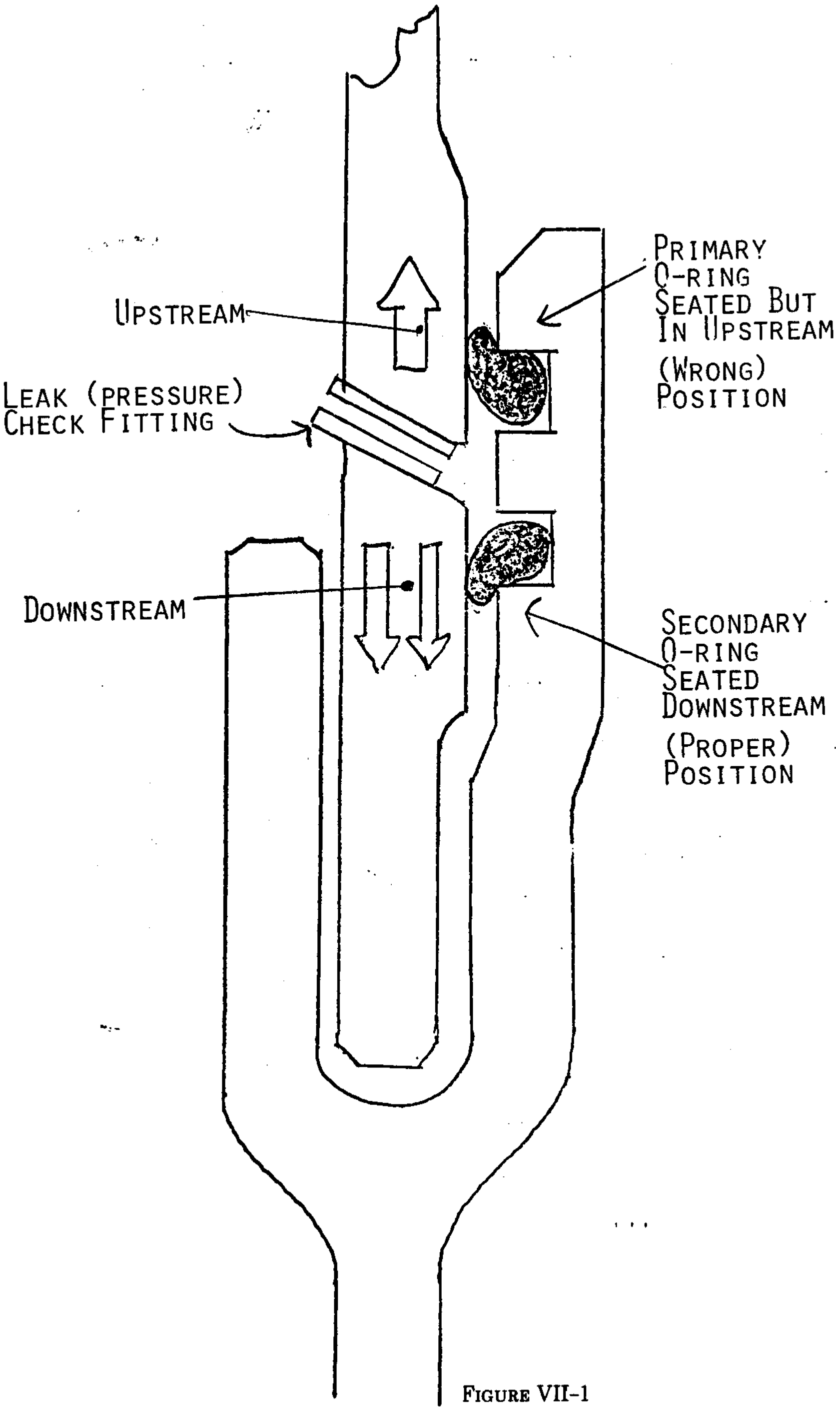
The fact that the aft field joint of the right-hand Solid Rocket Booster failed at the 300 degree location is overwhelmingly supported by the evidence. Retrieval of two large pieces of the joint clearly show that they were destroyed by the heat and velocity of the gas flame emanating from the right-hand booster. Additional supporting evidence was found by reviewing the telemetry data and the photographs taken during launch and flight.[1]

For the purpose of redesigning the joint it is important that the way in which the joint failed be determined as closely as possible. This determination, however, is difficult, if not impossible, to make with one hundred percent certainty. The evidence to support progress of the failure through the joint is incomplete. However, based on the recorded history of the joint problems encountered in flight and in test, based on the laws of physics, and based on behavior of the materials used in the joint, the following PROBABLE CAUSE is offered.

(b) Probable Cause of Failure

1. Both the primary O-ring and the secondary O-ring were seated when the steel casings were mated. The pressure check verified this fact. However, from experience, the primary O-ring was seated in the upstream position as had been previously recognized by NASA and Thiokol engineers. (See Figure VII–1.)

[1] Rogers Commission Report, Volume I, pp. 22–23 and 78–79.



PRIMARY O-RING SEATED BUT IN UPSTREAM (WRONG) POSITION
UPSTREAM
LEAK (PRESSURE) CHECK FITTING
DOWNSTREAM
SECONDARY O-RING SEATED DOWNSTREAM (PROPER) POSITION

FIGURE VII-1

2. Upon ignition, the primary O-ring could not reseat at the 300 degree location in the downstream position, where it needed to be in time to prevent blowby. At this point there were too many deficiencies acting in unison which prevented the O-ring from reseating in that location. First, proper spacing between the inner face of the tang and the opposing face of the inner leg of the clevis approximately 0.020 inches, is critical. That spacing for Flight 51-L was too small, at the 300 degree location where the smoke was observed, to facilitate prompt reseating of the primary O-ring. Calculations of segment diameters indicate the gap spacing was only 0.004 inches, near metal-to-metal contact. The ignition gases passed the O-ring at this location (See Figure VII-2). This condition did not exist elsewhere in the joint around the casings since the primary O-ring was able to seat around the joint in other locations.

Second, the low temperature throughout the night prior to launch left the fluorocarbon elastomer primary O-ring stiff and lacking in ability to spring into the downstream (seated) position at the 300 degree location in time, relative to the buildup of motor pressure, to provide a tight seal. The temperature of the aft field joint at time of launch was calculated by Thiokol after the accident to be 16 degree F. Part of the reason for this low temperature was the heat transfer away from the joint, by conduction through the aft attachment strut. The conduction was driven by liquid hydrogen, which remained in the external tank overnight. The supercold fuel created a 430 degree temperature differential across the ship, drawing heat out of the joint and O-rings.

At ignition, blowby occurred, either with erosion of the primary O-ring or without erosion.

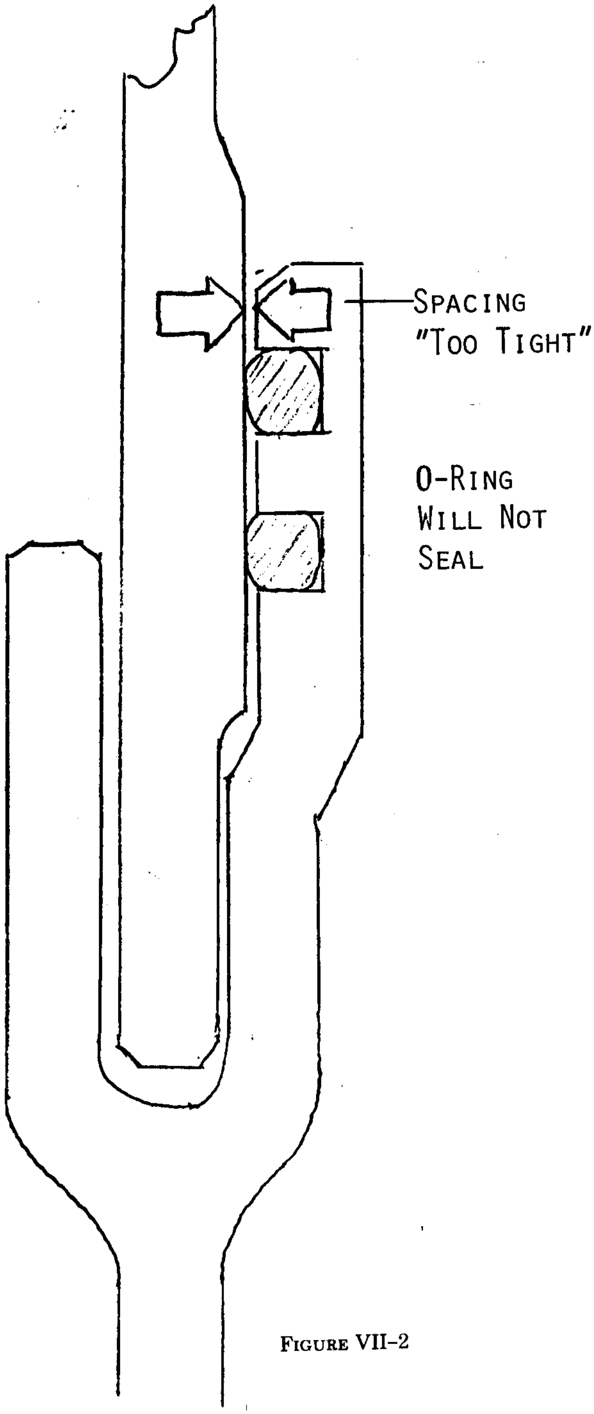


Figure VII–2

3. However, there could have been one or more than one blowhole through the zinc chromate putty before ignition. One such blowhole could have been made at the 300 degree location either during the leak check at 200 psi prior to the seating of the primary O-ring or prior to the leak check when the casings were brought together. If so, there was a high probability that the primary O-ring eroded at this point. The phenomena of blowing holes in the putty had been observed many times at post-flight dismantlement and had dramatically increased when the procedures were changed to increase the test pressure to 200 psi. Additionally, the Randolph putty had been found unsatisfactory on numerous occasions.[2]

4. Upon ignition of the Solid Rocket Motor, this blowhole would have facilitated and concentrated the hot propellant gas flame on the primary O-ring, and possibly the secondary O-ring as well. Alignment of O-ring erosion with the location of blowholes had been observed on numerous occasions.[3]

5. Between the time the casings were assembled and the launch, the secondary O-ring was unseated from its previously sealed position. The fact that it had been sealed has been verified by the pressure check made 28 days before when the casings were joined. Either the secondary O-ring was unseated by joint rotation coupled with O-ring stiffness or by the formation of ice in the joint.

During the intervening period, as the Shuttle stood on Pad 39B waiting for the launch, 7 inches of rain had fallen and some could have easily penetrated the joints. The access of rain water into the joints was proved when STS-9 was disassembled and water poured out of the assembly pin holes. In tests conducted after the accident, it was confirmed that the water in the aft field joint would have turned to ice, and that the ice could have dislodged the secondary O-ring, pushing it upstream into a non-sealed position. In this position, it is doubtful that the secondary O-ring could have sealed at ignition.

6. One of the three Solid Rocket Booster to External Tank aft attachment struts is also connected at the 300 degree location, just a few inches below the aft field joint. As the Space Shuttle system stood on the launch platform at Pad B on January 28, the large External Tank was gradually filled with liquid hydrogen and liquid oxygen. Liquid hydrogen, at a temperature of 423 deg F below zero, and liquid oxygen, at a temperature of 297 deg F below zero, caused the tank to contract as it was filled. Since the Solid Rocket Boosters are firmly bolted to the launch platform, a lateral force of approximately 190,000 pounds pulled sideways on the aft attachment strut and the Solid Rocket Motor casing, including the joint that failed.[4] Refueling of the tank was accomplished early on the morning of January 28.

At ignition, the 190,000-pound force was instantly released when the SRB hold-down bolts were blown loose. For the next two and a half seconds the right Solid Rocket Motor field joints experienced a 3 cycle per second vibratory load caused by the sudden release of

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[2] NASA, MSFC Memo, Miller to Horton, April 12, 1984.

[3] Thiokol, "Erosion of SRM Pressure Seals," TWR 15160, Chart A-9, August 19, 1985: "Seal damage always has associated putty blowhole."

[4] NASA, MSFC, "51-L Analysis Overview," April 25, 1986, p. H-203.

the lateral force.[5] The ignition pressure increased the joint spacing. Also, the flow of motor gases through the blowhole at the 300 degree location could have resulted in damage to the primary O-ring. The evidence of smoke at the 300 degree location is unlikely without O-ring damage.

7. Smoke at launch, clearly visible in the photographs, stopped at 2.7 seconds when the vibratory load damped out and the joint sealed. The sealing of the breach at the 300 degree location was made possible by blockage from burned material, probably consisting of a mixture of insulation and aluminum oxide. Post-accident tests performed by Morton Thiokol proved that aluminum oxide could have successfully plugged the joint at 2.7 seconds. While the smoke at ignition appeared to be intermittent, that appearance was probably a result of air and main engine exhaust currents.

8. At T+37 seconds into the flight, the Shuttle encountered wind gust loads in conjunction with planned maneuvers. Components of these gust and maneuvering loads were transmitted to the Solid Rocket Booster through the External Tank attachment strut. Based on the prescence of smoke at liftoff, these forces were transmitted to a joint already weakened by erosion and heat damage.

9. At 43 seconds into the flight, the main engines throttled back as the Shuttle reached "Max q" (maximum dynamic pressure). Four seconds later, the main engines had throttled up to 104% power and the geometry of the Solid Rocket Motor propellants had increased thrust. At this point, the motor pressure increased to 609 psi.

Additional structural loads resulted from turbulence. Flight 51-L experienced the most severe turbulence of any Shuttle flight and, although the loads were within the allowable design limits, those design limits did not consider a joint that had already failed.[6] It is unknown how much the combined effect of wind gust loads, maneuvering loads and an increase in thrust contributed to the accident. But the combined effects of these forces could have dislodged the burned material at the previously breached section of the joint.

10. Shortly after the vehicle was loaded by these turbulent forces, at T+58 seconds, a flame appeared from the same general region where the puffs of smoke had been seen. But, this time the joint was continuously breached by the burning propellant gases. In a little over two seconds, the flame had grown and acted as a blowtorch to burn through the hydrogen tank. The appearance of the flame at this time is also indicative of a damaged primary O-ring and failure of the secondary O-ring to seal, for reasons explained in the Critical Items List dated December 17, 1982.[6a]

The telemetry data, photographs and cockpit voice recordings support evidence of turbulent conditions and the manner in which the Shuttle failed.

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[5] The joint was designed to accommodate these loads.

[6] NASA, MSFC, "51-L Analysis Overview, STS 51-L—Wind Shears, April 25, 1986, p. H-597.

[6a] NASA, "SRB Critical Items List," December 17, 1982, page A-6A, sheet 1.

Time (min:sec)	Crew position [7]	Crew comment
T+19	PLT	Looks like we've got a lotta wind here today.
T+20	CDR	Yeah.
T+22	CDR	It's a little hard to see out my window here.
T+28	PLT	There's ten thousand feet and Mach point five.
T+30		(Garble)
T+35	CDR	Point nine.
T+37		[High thrust vector control (steering) activity noted. This was caused by upper atmoshpere wind gusts and planned maneuvers.]
T+40	PLT	There's Mach one.
T+41	CDR	Going through nineteen thousand.
T+43	CDR	OK, we're throttling down.
T+57	CDR	Throttling up.
T+58	PLT	Throttle up.
T+58		[Vehicle loaded by dynamic pressures.]
T+58		[A flame appeared from same general region where the puffs of smoke had been seen earlier.]
T+59	CDR	Roger.
T+60	PLT	Woooohoooo.
T+60		[Right SRM internal pressure began to diverge from that of left SRM]
T+61		[Well-defined plume was deflected indicating the plume had burned through the liquid hydrogen tank structure.]
T+62	PLT	Thirty-five thousand going through one point five.
T+64.7		[Liquid hydrogen leak noted.]
T+66.8	CDR	Reading four eighty six on mine.
66.800		[Leak confirmed when hydrogem tank leak pressurization system was unable to maintain normal pressurization rate.]
T+67	PLT	Yep, that's what I've got, too.
T+70	CDR	Roger, go at throttle up.
T+72.2		[Right Solid Rocket Booster motion differed from Orbiter and left Solid Rocket Booster, indicating failure of lower attachment structure.]
T+72.6		[Liquid hydrogen tank pressure fell. Leak was growing rapidly.]
T+73		[Liquid hydrogen and liquid oxygen pressure to main engines showed significant drop.]
T+73	PLT	Uh Oh
T+73.1		[Circumferential white pattern around the External Tank aft bulkhead suggested liquid hydrogen tank structure failure.]
T+73.1		[Vapor observed at inter-tank which was indicative of the liquid oxygen tank failing. Liquid oxygen then observed.] [8]

[7] (CDR) Commander Scobee, (PLT) Pilot Smith, (MS 1) Mission Specialist Onizuka, (MS 2) Mission Specialist Resnik.
[8] NASA, D.M. Germany, STS 51L Incident Investigation, Integrated Events Time Line, Johnson Space Flight Center, June 4, 1986, as modified.

(c) Problems Discovered

The design of the joint was based on the successful design of the joints used on the Titan III booster rocket.[9] That design was similar except that the tang pointed upward, instead of down, and the clevis pointed downward, instead of up, as in the case of the Shuttle booster. Another difference was that the design of the Shuttle joint included two O-rings instead of one as provided for in the

[9] NASA, "SRB Critical Items List," December 17, 1982, p. A-6A, Sheet 1.

Titan design. But, the most important difference was the use of putty in the Shuttle design. While the Titan employed the NBR insulation to close the gap between segments, the Shuttle design called for filling a gap between insulation with putty.

The Shuttle design was changed to accommodate manufacturing constraints.[10] The Shuttle booster is larger, 146 inches in diameter as compared to 120 inches for the Titan. As a result of its larger size, the Shuttle booster uses more steel. While this requirement for more steel had no impact on other booster components, it did have an impact on the joint design. The maximum billet size (a piece of metal made from an ingot) commercially available to manufacture the large, one-piece, weld free forward dome with an integral forward skirt tang was less than that needed for the Shuttle Solid Rocket Boosters. However, it was found that by turning the casings upside down, there would be just enough metal to manufacture a forward dome because that component would then only have to incorporate the single joint element, the tang, instead of the double joint element, the clevis.

It is good engineering practice to design products to accommodate manufacturing tooling capabilities and methods. Furthermore, with the clevis facing up and the tang down, field assembly at the Kennedy Space Center was simplified. Combined with the extra O-ring, the design change appeared reasonable. But it is also good engineering practice to accommodate all the forces and conditions that the product must perform under during its useful life. The design of the Shuttle Solid Rocket Motor, as opposed to the Titan, had to provide for reuse of the propellant casings, including the wearing of joint surfaces and distortion of the case in handling and shipment. It had to accommodate heavier propellant loads. The design was more susceptible to water entry during storms. And, most significant, the design had to accommodate a combination of dynamic structural loads significantly different than those encountered by the Titan.

It is always a simple task to find fault with someone else's work; especially after an accident occurs. It is quite another matter to originate the work and produce a useful product.

The joint design provided a direct path between the combustion chamber, consisting of an annulus with propellant surrounding it, and the outside of the steel motor casings. That path was sealed with putty and two circular fluorocarbon elastomer (rubber-like) bands called O-rings. While O-rings are frequently used to retain pressures much higher than those present in the Shuttle Solid Rocket Motor, thermal and structural forces acting on the Shuttle joints are formidable. These joints must carry and transfer these loads between the casings.

Another essential ingredient of good engineering practice is to use material suited to the function. Some O-rings can withstand high temperatures. But "all . . . elastomers become brittle at low [temperatures]. . . . Elastomers, like natural rubber, nitrile rubber, and Viton A . . . that become brittle at low [temperature]

[10] Staff discussion with E.G. Dorsey, Thiokol Wasatch Operations, Brigham City, Utah, September 4, 1986.

can be used for static seal gaskets when highly compressed at room temperature prior to cooling."[11]

But the Shuttle's O-rings were not used in a "static" system as evidenced by the variations in gap spacing between the tang and clevis. Nor would they always be highly compressed at room temperature prior to the cooling. Furthermore, the O-rings could not withstand burning propellant temperatures in the range of 5800° F. The design of the joint therefore provided for putty to insulate the O-rings from the burning gases.

This putty did not always perform as had been expected, and evidence of hot gas passing the putty and getting to the first, or primary, O-ring along the path to the outside of the rocket chamber was discovered. Once the putty was breached, the joint was not working as it had been designed. This failure, although recognized by NASA and its contractor Morton Thiokol, was neglected on March 8, 1984 when they chose to accept an "allowable degree of erosion," which meant there was an allowable percentage of failure.[12]

O-rings become effective (are seated) when pressure is applied to them as they sit in a groove provided to house them.

One question that the design was intended to answer was whether or not the O-ring was seated properly in its groove. An opening, with a fitting much like a valve stem on a tire, was provided to allow pressure testing between two such O-rings, the primary and the secondary.

But this design did not always answer the question: was the primary O-ring seated? Did it seal or not? Notice how the primary O-ring in Figure VII–1 (p. 176) is forced upward (shown by the single arrow). That is opposite to the normal direction that the propellant pressure acts (notice the double arrows). Even with an acceptable pressure check result, the primary O-ring would still be unseated for a fraction of a second when the motor pressure pushed the O-ring in the opposite direction from that which took place during the leak check.

The second reason the assumption concerning the leak check as "proof of sealing" could be erroneous was that the primary O-ring did not really have to seat at all if the putty behind it (toward the inside of the case) held the pressure during the leak check. So earlier in the program there really was no way to know whether the primary O-ring seated or not.

What appeared to be a rather straightforward joint was far from simple. If the primary O-ring did not seat during the leak check, and the pressure test succeeded, then the putty was doing the work of sealing. But it still was not possible to determine from outside the casings whether the putty or the O-ring was holding the pressure. But, if the leak check failed, then the O-ring was not seated and there was a blow-hole through the putty.

To resolve this concern, NASA and its contractor, Morton Thiokol, changed the leak-check procedure by increasing the pressure until a pressure of 200 pounds per square inch (psi) was accepted as

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[11] Theodore Baumeister, Editor, *Standard Handbook for Mechanical Engineers*, 7th Ed., (New York: McGraw-Hill, 1967), pp. 18–35.

[12] Rogers Commission Report, Volume II, p. H–1.

the standard. They had ascertained that this was sufficient pressure to blow a hole through the putty.[13] Then, if the O-ring failed to seat, the pressure would blow a hole through the putty and the test would disclose an unseated O-ring (a failed seal). But if the O-ring held the higher pressure, the O-ring would still have been seated in the upper position instead of the downward position. That would be contrary to the way the O-ring would have to be seated to contain the propellant pressure during launch of the Shuttle.

In summary, there was still no way to verify whether the primary O-ring was seated properly, meaning in the downstream position after the cases were joined together in the field. In the beginning of the development program the concept was that the putty would act somewhat like a "piston in a cylinder" when the propellant was ignited. As the chamber pressure built up, the putty was to move downstream and compress the air in the path between it and the primary O-ring. The compressed gas was to seat the O-ring and thereby seal the joint. Besides, even if the primary O-ring didn't seal, surely the secondary O-ring would, since it had already been pressure checked, which verified it was seated in the downstream position.

There was no direct evidence that the primary O-ring was not holding the pressure off the secondary ring until Flight 51-B. That was the first flight when erosion of the secondary O-ring had been observed, even though erosion of the primary O-ring had occurred before.[14]

Thiokol had considered the joint design to be Criticality 1R,[15] meaning that there was redundancy. While the second O-ring was redundant by design, the joint as a whole was still Criticality 1, since if it failed, it would mean the loss of the Shuttle and crew. In other words, there was no backup for the joint.

The joint was designed to mate two rocket motor segment cases, one to the other, where the lower edge of the upper case consisted of a tang and the upper edge of the lower case consisted of a clevis. After the tang was inserted into the clevis (which housed the two O-rings), 177 steel pins, each approximately 1 inch in diameter, were inserted from the outside through aligned holes which went through the outer leg of the clevis, the tang and partly into the inner leg of the clevis. The spacing between the inner face of the tang and the mating face of the inner leg of the clevis where the O-rings were housed was critical to the integrity of the joint because that spacing, in part, determined whether the O-rings could function properly to seal against the propellant gas pressures. Not only was the initial static spacing critical, but maintaining the proper spacing during launch and flight under dynamic structural loadings was necessary for an effective seal.

Upon ignition of the Solid Rocket Motor fuel the operating pressure increases to 922 psi at 40 degrees F within a little over one-half second (0.648 sec).[16] The effect of this pressure increase is to

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[13] NASA, MSFC, Problem Assessment System Record No. A07934, January 23, 1986, p. 6.
[14] Rogers Commission Report, Volume V, p. 1510.
[15] Cmte Hgs, Transcript, June 18, 1986, p. 51.
[16] Morton Thiokol, TWR-10212 (CD), Table 4-9, *Typical Propellant Design Data.*

cause the casings to bulge out around their midsections while being constrained by the thicker steel sections at the ends, much like a can of soda after freezing. The casings change shape during the buildup of motor pressure. This bulging has an effect on the joint. As in the case of the frozen soda can, the wall of the casing near the joint is no longer vertical, or perpendicular to the bottom, but angles out to meet the larger diameter in the center of the casing. NASA calls this change in angle at the joint "joint rotation."

This joint rotation is a component of an overall spacing problem that includes: changes caused by casing wear and tear experienced during refurbishment; case growth (swelling) from pressurizing the casings; distortion that occurs during shipment of the loaded casings; and the physical handling of the casings during stacking operations.

The joint rotation problem was aggravated when the steel casings were made thinner to achieve a reduction in weight and thus an increase in payload. The rotation problem was further aggravated by changing the design of the propellant geometry to achieve greater thrust. This increased the pressure within the casings and thereby increased the "gap opening."[17] These changes compromised the integrity of the joint seals because joint rotation increases the spacing (gap) between the tang and the O-ring grooves in the clevis.

When the increase in the gap occurs, it can open the O-ring seal, leaving the path from the propellant combustion chamber open to the outside of the casing, except for any blockage by the putty. But, as noted above, the putty frequently has holes blown through it. If there were blowholes in the putty, and the original spacing between the metal parts of the joint was such that the joint rotation left open spaces between the O-rings and the tang, then the joint would fail and burning gases would escape to the outside.

(d) Joint Behavior

In a memo from John Miller to Mr. Eudy of NASA on June 16, 1980, the following statement was made:

> STA-1 test data shows that the secondary O-ring can become unseated from the tang due to joint rotation at approximately 40 percent of MEOP [Mean Effective Operating Pressure], and therefore, is not likely to assume a sealing position should the above primary seal failure occur. The SRM has never been tested to evaluate the above failure condition, nor has credibility of such a failure been officially declared.[18]

In March of 1984 Thiokol had completed its SRM O-ring assembly test plan, which was to confirm the O-ring erosion scenario, provide data for heat transfer predictions and establish the effec-

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[17] The Light Weight Casings, first used on STS-6, had thinner casing walls than the standard steel casings. Light weight casings permitted flight with heavier payloads. On STS-8, NASA began using the High Performance Motor (HPM) which developed higher internal pressures while using the light weight casings. The purpose of the HPM was to further increase payload capacity.

[18] NASA, "Evaluation of TWR-12690 CD, Test Plan for Space Shuttle SRM Lightweight Cylinder Segment Joint Verification, dated June 10, 1980", EP 25 (80-70), June 16, 1980, p. 2.

tiveness of the vacuum putty. The introduction to that plan included the statement:

> O-ring seals in rocket motors in general and the Space Shuttle SRMs in particular, can suffer thermal degradation because of exposure to the high temperature motor chamber gases. Although none of the SRM primary O-rings to date have failed to perform their design function, there is some concern because of isolated events which show localized erosion as high as 0.053 inches. The postulated scenario for this thermal degradation effect is a short-time duration impingement of a high energy jet which is induced during ignition pressurization by a combination of voids in the protective vacuum putty and the filling of available free volumes created by the tolerances of mating parts and the O-ring slots. Unfortunately, the overall assembly and the vacuum putty layup does not lend itself to a well-defined geometry for predicting the hot gas flow and associated heat transfer to the O-rings.[19]

A subsequent report, dated May 7, 1984, contained a statement:

> Symptom of failure: a vaccum putty exhibited gas paths located at 319 deg., 338 deg., and 347 deg. Erosion of the primary O-ring occurred at 319 deg. only. The damaged region was approximately 5.6 inches long with a .034 inch maximum depth and involved 136 deg. of the O-ring cross section diameter.[20]

In a memo from Larry Mulloy to Bob Lindstrom, Director, MSFC Shuttle Projects Office, in November of 1984 it was noted:

> . . . it was determined that shims could be used to make the case joint sufficiently concentric to consistantly achieve a 7.54 percent minimum O-ring squeeze. Therefore the 7.54 percent has been established as the minimum acceptable requirement for both case and nozzle O-ring joints and verified by subscale testing and full scale experience.[21]

On a 0.280 inch diameter O-ring a 7.54 percent squeeze would be equal to a compression distance of 0.021 inches.[22]

On July 17, 1985, Irv Davids, Manager of the Solid Rocket Booster Program at NASA Headquarters, sent a memo to the Associate Administrator for Space Flight, the subject of which was case-to-

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[19] Thiokol, Philip Shadlesky, "Performance Characteristics of the SRM O-ring Assembly Test Plan", TWR-14336, dated March 1984, p. 1.

[20] Thiokol, S. Rodgers, "Significant Problem Report DR4-5/35: 5 Day Report; O-ring Erosion at Nozzle/Aft Segment Joint of SRM 11A (STS 41-B/Mission 41-C)", TWR-14370-1, May 7, 1984, pp. 1-2.

[21] O-ring squeeze is the distance, in fractions of an inch, that an O-ring is compressed from its normally round shape. This dimension can also be expressed as a percentage of the total diameter before compression. In 1984 NASA was using a term "minimum O-ring squeeze." During an SRM design analysis of the case and nozzle O-ring joints it was concluded that the 146 inch diameter case cylinders would not meet the design standard of 15 percent minimum O-ring squeeze at zero pressure. The various problems that prevented this included flaws in the O-ring grooves and sealing surfaces and differences in the spacing between tang and clevis on various casings.

[22] NASA, Larry Mulloy, "ECP SRM 1197, Nozzle Nose Inlet Housing O-ring Squeeze," SA 42-562-84, November 20, 1984, p. 1.

case and nozzle-to-case O-ring seal erosion problems.[23] Davids sent copies to Messrs. Weeks, Hamby, Herrington and Winterhalter.[23a] In the memo it was noted that there has been twelve instances of primary O-ring erosion during Shuttle flights. In addition, in one specific case there had also been erosion of the secondary O-ring seal. There were also two primary O-ring seals that were heat affected without erosion and two cases in which soot blewby the primary seals. In this memo it was noted that the prime suspect for the cause of erosion on the primary O-ring seals was the type of putty being used. It was Thiokol's position that during assembly leak check, or ignition, a hole could be formed through the putty which then initiated O-ring erosion due to a "jetting effect." It was even mentioned in this memo that Thiokol was seriously considering the deletion of putty on the QM-5 nozzle/case joint since they believed the putty was the prime cause of the erosion. Davids, however, had reservations about deleting the putty because he recognized the significance of the QM-5 firing in qualifying the FWC (Filament Wound Case) for flight.

In the matter of case-to-case O-ring erosion the memo noted that there had been five occurrences during flight where there was primary field joint O-ring erosion. There was also one case where the secondary O-ring was heat damaged with no erosion. The memo stated:

> The erosion with the field joint primary O-ring is considered by some to be more critical than the nozzle joint due to the fact that during the pressure build up on the primary O-ring the unpressurized field joint secondary seal unseats due to joint rotation.[24]

The memo continued:

> The problem with the unseating of the secondary O-ring during joint rotation has been known for quite some time. In order to eliminate this problem on the FWC field joints a capture feature was designed which prevents the secondary seal from lifting off.[25]

Lastly the memo noted:

> The present consensus is that if the primary O-ring seats during ignition, and subsequently fails, the unseated secondary O-ring will not serve its intended purpose as a redundant seal. However, redundancy does exist during the ignition cycle, which is the most critical time.[26] (See Appendices VII-B and VII-C.)

On August 2, 1985, Larry Wear, MSFC's SRM Element Manager, sent a letter to Joseph Kilminster, Thiokol's Vice President for Space Booster Programs, on the subject of SRM field joint second-

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[23] NASA, Irving Davids, "Case to Case and Nozzle to Case 'O' Ring Seal Erosion Problems," July 11, 1985.

[23a] Mr. Weeks, Dep. Assoc. Administrator for Space Flight (Technical); Mr. Hamby, Dep. Dir., STS Program Integration; Mr. Herrington, Dep. Dir. of Launch & Landing Operations; and Mr. Winterhalter, Acting Dir., Shuttle Propulsion Div.

[24] Ibid., p. 2.

[25] Ibid.

[26] Ibid.

ary O-ring lift-off during pressurization. The letter concerned the situation wherein one O-ring might not seal subsequent to joint rotation. The letter stated:

> Because of recent experiences of flight and ground test motors having increasing incidences of putty blow-holes and the associated burning of primary O-ring, it would seem prudent for us to attempt to assure that the secondary O-ring is capable of sealing during the entire SRM burn.[27]

The letter requested an assessment of the possibility of lift-off of the secondary O-ring.

In August of 1985 Jim Thomas, MSFC's Deputy SRM Element Manager, wrote a memo for Mr. Mulloy to Mr. Hamby at NASA Headquarters, which was apparently never signed or sent. The subject of the memo was SRM Joint/O-ring Erosion. The memo stated:

> On July 11, 1985, you and Irv Davids were briefed by Jim Thomas of my office on the history of the effort underway to resolve the issues and concerns of the above subject.

The memo than went on to discuss a number of questions.

1. What would happen if the secondary seal lifted off the mating surface during motor pressurization, and, also, how long it would take for the seal to return to a position where contact was made? The answer to that question stated that bench test data indicated that the O-ring resiliency, that is, its capability to fill the gap between the tang and the clevis, was a function of temperature and the rate at which the gap opened.

The memo stated, "at 100 deg. F the O-ring maintained contact. At 75 deg. F the O-ring lost contact for 2.4 seconds. At 50 deg. F the O-ring did not reestablish contact in 10 minutes at which time the test was terminated." The memo then stated, "the conclusion is that secondary sealing capability in the SRM field joint cannot be guaranteed." [28]

2. Another question concerned whether or not the secondary O-ring would seal in sufficient time to prevent joint leakage if the primary O-ring had not sealed. The answer to that question was as follows:

> MTI has no reason to suspect that the primary seal would ever fail after pressure equilibrium is reached, i.e., after the ignition transient. If the primary O-ring were to fail from 0 to 170 milliseconds, there is a very high probability that the secondary O-ring would hold pressure since the case has not expanded appreciable at this point. If the primary seal were to fail from 170 to 330 milliseconds, the probability of the secondary seal holding is reduced. From 330 to 600 milliseconds the chance of the secondary seal

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[27] NASA, Larry Wear, "SRM Field Joint Secondary O-ring Lift-Off During Pressurization," SA 41-326-85, August 2, 1985.

[28] Engineering consultants to the Committee have serious questions as to how this test relates to actual O-ring performance in flight hardware.

holding is small. This is direct result of the O-ring's slow response compared to the metal-case segments as the joint rotates.

3. The third question indicated that NASA Headquarters was not aware that the secondary O-ring may not seat due to joint rotation, and they wanted to know when this data was incorporated into the FMEA/CIL? The answer noted that Thiokol had submitted a TWR-13520 to MSFC in December of 1982. This was approved by NASA Level III on January 21, 1983. NASA Level II authorized a change request March 2, 1983 and Level II issued a PRCBD to implement approved Level I change request on May 2, 1983.[29]

Thiokol completed their engineering study of O-ring compression set and dated the report October 2, 1985.[30] (Compression set relates to the ability of a material, in this case, O-rings, to rebound to its original dimensions after having being subjected to compression for various periods of time and or at various temperatures.) That report contained the following information. There was a concern of the ability of the O-ring to rebound to or near its original dimensions after having been subjected to compression for various periods of time and at various temperatures. The Parker Seal Company of Culver City, California, tested several O-rings to determine the properties of the material. Two compression set tests in accordance with ASTM (American Society for Testing and Materials) D-395 method B were performed.[31] The first test was conducted at a constant temperature of 75 deg. and the time that the ring was in compression was varied. In the second test the temperature was varied and the compression was held constant. A small O-ring of 0.139 inch diameter was used for test purposes. The test showed that the percentage of compression increased with an increase of temperature. However, these tests were not conducted at low temperatures. Rather, they were conducted at temperatures of 212 deg. F and above and therefore, they have little relevance to ambient conditions.

A status report from Thiokol's SRM O-ring Task Force, presented on November 20, 1985, recommended that a slightly larger Viton O-ring of 0.292 inch diameter, along with thicker shims, be used as a short-term solution. The current O-rings were 0.280 inches. Thiokol pointed out that there would be more erosion margin due to greater material thickness at the sealing surface. They noted that the thicker shims would reduce the initial and absolute final gap opening dimension, resulting in more O-ring "squeeze" initially. Thiokol stated that the greater initial squeeze would be better for compression set and resiliency, and would give a higher probability of maintaining a secondary seal longer into the ignition transient. Thiokol also noted that various tests were conducted on the Randolph putty using hot five-inch char motors.[32] Two tests were conducted, which determined that the

[29] NASA, Larry Mulloy, "SRM Joint/O-ring Erosion," SA 42-349-85, August 1985, pp. 1-2.
[30] Thickol, B.L. Orme, "Enginering Study of O-ring Compression Set," TWR-15218, October 2, 1985.
[31] Refer to Appendix VII-A for ASTM specification.
[32] Small scale test motors.

putty erosion could take place at a rate between 5.5 and 13.0 mils per second. Two other tests noted that the erosion on GS-43 [33] putty was ten times higher than that on the Randolph.[34]

Primary concerns drawn from the charts provided by Thiokol on January 27, 1986, centered around the following items. During the ignition transient, 0 to 170 milliseconds, there is a high probability of a reliable secondary seal. Between 170 and 330 milliseconds there is a reduced probability of a reliable secondary seal and between 330 and 600 milliseconds there is a high probability of no secondary seal capability. Under steady state conditions, between 600 milliseconds and two minutes, the notes states "if erosion penetrates primary O-ring seal—high probability of no secondary seal capability." [35]

A. Bench testing showed O-ring not capable of maintaining contact with metal parts gap opening rate to MEOP.

B. Bench testing showed capability to maintain O-ring contact during initial phase (0 to 170 ms) of transient.[36]

What follows is taken from Chart 2-2:

1. A temperature lower than current data base results in changing primary O-ring sealing timing function.

2. SRM 15-A 80 deg. arc black grease between O-rings. SRM 15-B 110 deg. arc black grease between O-rings.

3. Lower O-ring squeeze due to lower temperature.

4. Higher O-ring Shore hardness.

5. Thicker grease viscosity.

6. Higher O-ring pressure activation time.

7. Activation time increases, threshold of secondary seal pressurization capability is approached.

8. If threshold is reached then secondary seal may not be capable of being pressurized.[37]

The presentation went on to included the following blow-by history:

SRM 15 worst blow-by.

A. Two case joints (80 deg.), (110 deg.) arc.

B. Much worse visually than SRM 22. SRM blow-by.[38]

The presentation then included a chart titled "O-ring (Viton) Shore Hardness vs. Temperature." [39]

Degree F	Shore Hardness
70 degrees	77 hardness
60 degrees	81 hardness
50 degrees	84 hardness
40 degrees	88 hardness
30 degrees	92 hardness
20 degrees	94 hardness
10 degrees	96 hardness

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[33] A type of putty made by another company that also was considered for use in the SRM.

[34] Thiokol, "SRM O-ring Task Force Status and QM-5 Recommendations," TWR-15349, November 20, 1985.

[35] Thiokol, "Temperature Concern on SRM Joints," January 27, 1986, chart 2-1.

[36] Ibid.

[37] Ibid., Chart 2 2.

[38] Ibid., Chart 3-1.

[39] Ibid., Chart 4-1.

The term Shore Hardness refers to a method of identifying the hardness of materials, and a higher number means a harder material. Regardless, though, it is seen from the above table that the hardness increases as the temperature decreases. Engineers presented a chart titled Secondary O-ring Resiliency, listing the following temperatures.[40]

Temperature degree F	Time to recover (seconds)
50 degree	600 recover
75 degree	2.4 recover
100 degree	*did not separate

The conclusions presented at the end of the teleconference were:

1. Temperature of O-ring is not only parameter controlling blow-by. SRM 15 with blow-by at an O-ring temperature at 53 deg. F. SRM 22 with blow-by at an O-ring temperature at 75 deg. F. Four development motors with no blow-by were tested at O-ring temperature of 47 deg. to 52 deg. F. Development motors had putty packing which resulted in better performance.
2. At about 50 deg. F blow-by could be experienced in case joints.
3. Temperature for SRM 25 on 1/28/86 will be 29 deg. F 9:00 a.m., 32 deg. F. 2:00 p.m.
4. Have no data that would indicate SRM 25 is different than SRM 15 other than temperature.[41]

Recommendations

1. O-ring temperature must be greater than or equal to 53 deg. F at launch. Development motors at 47 deg. to 52 deg. F with putty packing had no blow-by. SRM 15 (the best simulation) worked at 53 deg. F.
2. Project ambient conditons (temperature and wind) to determine launch time.[42]

The effect of Thiokol's recommendations would be that the Shuttle should not be launched unless the O-ring seal temperature was at least 53°F.

(e) Loads Acting on the Joint

There are other loads on the joint in addition to those caused by the pressures of the burning propellant. The following table identifies those loads relative to time.[43]

Time	Activity	Source of load	Static or dynamic	Impact on joint
Days before launch	Mating of casing	Weight of upper casing contacting lower casing.	Static plus impact	Physical contact between tang and clevis.

[40] Ibid., Chart 4-2.
[41] Ibid., Chart "Conclusions."
[42] Ibid., Chart "Recommendations."
[43] This Chart was prepared by the Committee and is based on information obtained by Committee staff during meetings at MSFC on June 30, 1986.

Time	Activity	Source of load	Static or dynamic	Impact on joint
Days before launch	Solid rocket motor assembly (Stacking) at KSC.	Weight of SRB components.	Static	Compressive; shear on pins at faces between tang and each clevis leg.
Days before launch	Joining of the external tank, and orbiter to SRB's.	Additional weight of tank and orbiter.	Static	Compressive; additional shear on pins which connect tang to clevis.
Days before launch	Transport to pad on crawler.	Movement of Transporter	Static and dynamic	Compressive; slight shear changes on pins.
Days before launch	Addition of payloads	Added weight	Static	Additional compressive and shear loads.
Within 24 hours of launch.	Loading of fuel	Weight of liquid hydrogen and liquid oxygen.	Static	Additional compressive and shear loads.
Within 24 hours of launch.	Loading of fuel	External tank contracts in diameter due to reduction in temperature.	Static	Lateral tensile force applied by aft attachment structure between external tank and solid rocket motor casing.
6 seconds to launch	Firing of main engines (SSME's).	Thrust of engines	Static and dynamic	Further moments compressive and vibratory (25 to 30 Hz).
At start of launch	Solid rocket motor ignition before lift-off.	Combustion pressures	Basically static	Bending (Joint rotation) lateral forces perpendicular to casing walls.
At start of launch	Solid rocket motor ignition before liftoff.	Engine thrust	Static and dynamic	Instant load reversal from compressive to tensile in joints and instant shear reversal in pins.
At start of launch	Release of hold down bolts.	Instant release of lateral force at aft External Tank attachment structure.	Dynamic	Instant change in stress, vibratory at 3 Hz.
At liftoff	Launch maneuvering	Thrust, plus nozzle vector forces.	Static and dynamic	Combination: Vibration, tensile, shear, lateral (via attachment structure).
Launch phase	In-flight maneuvering	Thrust plus gimbaling, applied loads at attachments.	Static and dynamic	Combination vibration, tensile, shear, lateral.
Launch phase	Turbulence—wind gust loads.	Impact, thrust nozzle gimbaling (changes in applied loads).	Static and dynamic	Impact loads transmitted to joints.
Launch phase	Reduction of main engine power and Solid Rocket Motor thrust at Max q (maximum dynamic pressure).	Decrease in thrust	Static and dynamic	Changes in bending and stress in joint, changes in frequency of vibration.
Launch phase	Increases in thrust thrust of main engines and solid rocket motors.	Increase in thrust	Static and dynamic	Changes in bending and stress in joint and in vibration.
Separation phase	Burning out of solid propellants and explosive forces at attachment points.	Release of thrust, impact forces at attachment points.	Static and dynamic	Reduction of tensile forces and shear on pins.

Time	Activity	Source of load	Static or dynamic	Impact on joint
Ocean impact and retrieval.	Weight of SRB impacting ocean at about 60 mph (vertical) 25 mph (horizontal).	Impact on joint	Dynamic	Variation in stresses at joints.

How these loads are accommodated by the joint is critical to the seal. In Thiokol's analytical evaluation report (TWR-12019, dated October 6, 1978), S. Stein of the Structures Section included the statement, "except in local area of pin, all stress levels are considerably below yield." [43a] As a result of this information, the Committee will explore this condition as part of its normal oversight work to determine the long-term effect on structural integrity of the casings.[44] Stein also wrote, "at MEOP [maximum expected operating pressure] the primary 'O' ring gap increases 0.052 and the secondary 0.038"." [45] It should be noted however, the analysis was made for no thrust, i.e. internal pressure only.[46] As noted on the forgoing chart, loads on the joint do work in combination and so the analysis should also provide for the combined effect of all loads at the time they occur.

On page 55 of the Rogers Commission Report there is a chart which shows a series of curves which relate maximum aerodynamic force to Mach Number. As a result of a discussion with Dr. Richard Feynman, Department of Physics, California Institute of Technology, and a member of the Rogers Commission, the Committee will review these curves after the completion of this report in an effort to ascertain their validity. There is reason to suspect that the "flight envelope" as repressented in the chart is inaccurate.[47]

As stated previously, the proper choice of materials is critical to attaining performance objectives. The steel casings are designed to withstand the propellant pressures and loads incurred in flight. Secondly, they must accommodate these forces over and over as the casings are reused. Consequently, the choices of the type of steel selected was important.

The steel used to make the casings and the joint is a D-6A. D-6A is a low-alloy steel for aircraft and missile structural applications. It is designed primarily for use at room-temperature tensile strengths of 260 to 290 ksi.[48] D-6A maintains a very high ratio of yield structure to tensile strength up to a tensile strength of 280 ksi, combined with good ductility.

Typical mechanical properties of D-6A steel: [49]

[43a] Thiokol, S. Stein, "Analytical Evaluation of the Space Shuttle Solid Rocket Motor Tang/Clevis Joint Behavior", TWR-12019, October 6, 1978, p. 1.

[44] Thiokol, S. Stein, "Analytical Evaluation of Space Shuttle SRM Tang/Clevis Joint Behavior", TWR-12019, October 6, 1978.

[45] Ibid.

[46] Ibid.

[47] Discussion with Dr. Richard Feynman, California Institute of Technology, Pasadena, California, September 3, 1986.

[48] 1000 pounds per square inch equals 1 ksi.

[49] American Society for Metals, Handbook Edited by H.E. Boyer and T.L. Gall, November, 1984.

Tempering temperature		Tensile strength ksi	Yield strength ksi	Elongation in 50 mm or 2 in., percent	Reduction in area, percent	V=notch impact energy ft-lb
°C	°F					
150	300	299	211	8.5	19.0	10
205	400	290	235	8.9	25.7	11
315	600	267	247	8.1	30.0	12
425	800	236	228	9.6	36.8	12
540	1,000	210	204	13.0	45.5	19
650	1,200	150	141	18.4	60.8	30

Normalized at 900°C (1650°F) and tempered at various temperatures.

In addition to the steel, other principal materials in the joint design that were to seal in the propellant gases were the zinc chromate putty and the O-rings.

On April 12, 1984, John Miller, Chief of the Solid Motor Branch of NASA, wrote a memo to Mr. Horton, Chief Engineer, SRB Engineering Office, MSFC which referred to concerns with putty made by Randolph. The Randolph putty was selected on the basis that it had several desirable performance characteristics. The change in putty was made after Fuller-O'Brien discontinued making putty because their product contained asbestos. Mr. Miller noted, "Stacking difficulties and observed O-ring anomalies appear to be more frequent with Randolph putty than with the previously used Fuller-O'Brien putty." [50] Miller requested that Thiokol expedite development and qualification of a putty with properties similar to those of Fuller-O'Brien.

On June 18, 1984, Miller wrote Horton again, mentioning erosion/heat exposure O-ring experience on QM-4, STS-2, STS-6, STS-11, and STS-13 and citing Deficiency Reports which violated specifications.[51]

By June 29, 1984, 5 inch motor tests has been completed. These tests substantiated the concept of hot gas jet impingement against O-rings. Interestingly, a simulation of "no putty" yielded no O-ring damage. This information was conveyed to NASA via telecon from Thiokol, which also stated that there was no second source for the Randolph putty. Thiokol had abandoned their program to mix the putty themselves. Measures taken to correct the putty problems included changes in the putty layup to reduce air entrapment, use of a porous sacrificial heat barrier such as carborundum fiberfrax or removing the putty and reducing joint gaps were introduced.[52]

A new joint design was forwarded to NASA by Thiokol on July 19, 1984, which included a fill capture feature. This feature looked similar to the "capture feature" proposed for future Shuttle flights. The fill capture feature, however, was to be filled with grease. A thermal analysis had shown that "severe heat effects would result if the cavity were not filled."

As stated previously, the putty was to insulate the O-ring seals from the hot propellant gases. It was also to remain flexible enough to move outward under the pressure of the burning propellant, thereby compressing the gas in the joint which, in turn, was

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[50] NASA, John Miller, "Concerns with Randolph Vacuum Putty," EP-25 (84-35), April 12, 1984.

[51] NASA, John Miller, "Zinc Chromate Putty Installation in Nozzle to Case Joint Discrepancy," EP-25 (84-53), June 18, 1984.

[52] Thiokol, "Vacuum Putty Telecon," June 29, 1984.

to seat the primary O-ring. O-rings require some pressure from the working fluid, in the case of the SRM, this was gas, in order to seat properly and provide a effective seal. In practice, this design philosophy did not prove to be correct because the putty frequently held the pressure off of the O-rings, or if it did not, the putty had blowholes in it. It was then postulated that these holes might actually benefit the seating of the O-ring by allowing more pressure to reach it sooner. It was even suggested that holes might be deliberately made through the putty. However, it was then learned that blowholes served to concentrate propellant gas on small segments of the primary O-ring and caused the ring to erode.

The unacceptable heat erosion damage to both primary and secondary O-rings on SRM-16A resulted in an evaluation of the putty produced by Randolph Products. In July 1985, L. Thompson of MSFC made a presentation which noted that five different types of putty from four companies were under study in an effort to solve the putty performance problem. As late as 1985 twelve different types of tests had been performed and six more were in progress. The only putty to survive the water tests was General Sealants No. 43, which was a non-asbestos formulation. The Randolph putty had disintegrated in all three water tests. However, in comparing dynamic viscosity to temperature, the General Sealants product, at 25,000 poise,[53] was not viscous above 125 deg C. It was slightly better than the Randolph product and another product made by Inmont. The previously used Fuller-O'Brien product, however, increased in dynamic viscosity with an increase in temperature. It was 100,000 poise at 250 deg C, while it was less than 50,000 at 50 deg C.[54] Consequently, no product met all the design requirements as well as the Fuller-O'Brien did.

The Randolph putty is hydroscopic and its behavior is unsuited to use in the dry climate of Utah, as well as the humid climate of the Florida coast. In one case the putty was too stiff and in the other, too sticky. Since both factory and field joints required the use of the putty, a product with consistent performance in both climates was required.

The materials used in the manufacture of the O-rings was also critical to the safe operation of the Shuttle system. The O-rings had to be serviceable at the high temperatures in the joint which would result from heat transfer from the rocket combustion chamber. However, the use of NBR insulation around the propellant, and the use of putty, was to protect the steel casings and the O-rings from the direct heat of the propellant gases. This protection was not always successful when blowholes in the putty occurred, however, and the O-rings would frequently be damaged by heat. The lower temperatures that occur in Florida during the winter months was not covered by NASA's specifications. While elastomers are known to become brittle at low temperatures, a product specification sheet on Viton Fluroelastomer claimed, "Cold-VITON is generally serviceable in dynamic applications down to −18 to −23 deg C (0 to 10 deg F)."[55] The sheet added: "The brittle point of Viton at a thick-

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[53] Poise is a measure of viscosity or resistance to flow.
[54] NASA, L.M. Thompson, "SRM/SRB Putty Evaluation," July, 1985.
[55] 3M-Chemical Division, "Viton Fluoroelastomer," Undated, p. 1.

ness of 0.075 inches is in the neighborhood of 50 deg F. Yet, as with other elastomers, thickness has a marked effect upon low temperature flexibility. Thinner cross-sections are more flexible than thicker ones at every temperature." The thickness of the O-rings on the Shuttle is 0.280 inches, thicker than the 0.075 inch article with a brittle point of 50 deg F noted above.[56] Consequently, the brittle point of Viton was misleading since the O-rings were much larger the the test specimen.

Military Specification MIL-R-83248A, 17 Feb. 84, "Rubber, Fluorocarbon Elastomer, High Temperature, Fluid, and Compression Set Resistant" set the specification for the O-rings that Thiokol had to meet.[57] They included:

> Type I-O-rings and compression seals Class I-75 +/−5 Hardness[58]

This specification then included other specifications issued by the Society of Automotive Engineers and the American Society for Testing Materials. One of the ASTM Specifications listed was ASTM 1329, "Evaluating Rubber Property, Retraction at Low Temperatures."[59] It was these referenced specifications which defined the significant characteristics required.

On February 6, 1979, Mr. William Ray of NASA's Marshall Space Flight Center, sent a memo to Messrs Hardy, Rice, Eudy, and McCool (See Appendix V-I). That memo was essentially a trip report of Mr. Ray's visits to the Precision Rubber Products Company and the Parker Seal Company, in search of information on the performance of O-rings. Some of the points covered in the memo were:

> The purpose of the visits was to present the O-ring seal manufacturers with data concerning the large O-ring extrusion gaps being experienced on the Space Shuttle Solid Rocket Motor clevis joints and to seek opinions regarding the potential risks involved.[60]
>
> With regard to the visit with company officials at Precision Rubber Products, "they voiced concern for the design, stating that the SRM O-ring extrusion gap was larger than that covered by their experience." [61]
>
> In response to the data presented to Parker Seal Company officials by Mr. Ray, Parker officials "also expressed surprise that the seal had performed so well in the present application." [62]
>
> Regarding the visit with the Parker officials, the memo stated, "their first thought was the O-ring was being asked to perform beyond its intended design and that a different type of seal should be considered." [63]

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[56] Ibid.

[57] Department of Defense, "Military Specification: Rubber, Fluorocarbon Elastomer, High Temperature, Fluid, and Compression Set Resistant", MIL-R-83248A, February 17, 1984.

[58] Ibid, p. 1.

[59] Ibid., p. 3.

[60] NASA memorandum, William Ray, "Visit to Precision Rubber Products Corporation and Parker Seal Company", EP 25 (19-23), February 6, 1979, p. 1.

[61] Ibid.

[62] Ibid., p. 2.

[63] Ibid.

> The need for additional testing of the present design was also discussed and it was agreed that tests which more closely simulate actual conditions should be done.[64]

As a result of the foregoing data, the Committee has arrived at the specific Findings and Recommendations contained in Chapter V.

[64] Ibid.

VIII. LAUNCH OPERATIONS

INTRODUCTION

The purpose of this section is to document the series of decisions that culminated in the launch of STS 51-L on January 28, 1986. In Section A, the discussion details the Flight Readiness Reviews used to assess the mission's readiness, and also describes the teleconference on the night of January 27 when Thiokol engineers attempted to delay the launch. Also discussed are the circumstances surrounding the uncertainty represented by ice covering the launch pad's gantry.

Section B describes a specific example where the launch crew in the Firing Room waived a launch commit criterion. The discussions that took place on the subject indicate that the alternate procedure used as a justification for the waiver should not have been allowed, since the environmental conditions on the morning of January 28 were outside the limits specified for the alternate procedure.

A. THE STS 51-L LAUNCH DECISION

Discussion

Before each flight of the Space Shuttle, the ground support team carries out a series of meetings that are collectively known as the Flight Readiness Review. Policy guidance for this procedure is supplied by NASA Program Directive 710.5A, which states:

> It is the policy of the Associate Administrator for Space Flight (AA-SF) to make an assessment of mission readiness prior to each flight. This will be accomplished by a consolidated Flight Readiness Review (FRR) of all activities/elements necessary for safe and successful conduct of the launch, flight, and post-landing operations. . . . The FRR will be preceded by detailed readiness reviews (pre-FRR's) on individual elements, including cargo, under the cognizance of the responsible Managers.[1]

The FRR policy directive offers the following guidance to project and program managers regarding the expected content of their presentations:

> The Project/Element Managers will conduct pre-FRR's to develop their readiness assessment and are responsible for the FRR briefing content in their particular area.[2]

As for the agenda at these reviews, the directive has this to say:

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[1] James Abrahamson, NASA, Headquarters, "Space Shuttle Flight Readiness Reviews," SFO-PD 710.5A, September 26, 1983, p. 1. See Appendix VIII-A.

[2] Ibid., pp. 1-2.

The presentation of agenda items will normally include a brief status summary with appropriate supporting detail on significant items and conclude with a readiness assessment. The presentation topics and scope should be developed from the pre-FRR's and should:

(1) be that required to provide the AA-SF with the information needed to make a judgment as to flight readiness;

(2) review recent significant resolved problems and prior flight anomalies when necessary to establish confidence;

(3) cover all problems, open items and constraints remaining to be resolved before the mission;

(4) establish the mission baseline configuration in terms of all significant changes since the last STS mission (changes to be considered include hardware, software, vehicle servicing/checkout, launch commit criteria, flight plans, flight rules and crew procedures);

Within the above guidelines, the scope of the review should cover status and issues in areas such as: vehicle checkout, shortages and open work, unexplained anomalies, hardware failures, prior flight anomalies, certification/verification, as-built hardware configuration versus certified hardware list, Critical Items List (CIL), development, qualification and reliability testing, waviers and deviations, limited life components, launch critical spares, sneak circuits, system safety/hazards and flight margins. . . .[3]

In the case of STS 51-L, no deviation from normal procedure apparently occurred. This means that the Solid Rocket Motor, containing the seal that apparently failed, proceeded through the usual eight levels of review at Thiokol's Wasatch Division, Marshall Space Flight Center, the STS Program Office at Johnson Space Flight Center, and the Associate Administrator's review at NASA Headquarters.[4]

The Flight Readiness Review process for STS 51-L began on December 11, 1985, at Thiokol's Utah plant. No information is presented in the briefing charts used that day regarding the continuing failure of the SRM joint seals. The chart entitled "STS-61C (STS-32) (SRM-24) Performance "has only one entry: "TBD [to be determined]."[5]

Post-flight disassembly of STS 61-C SRB hardware following its launch on January 12 revealed that erosion of the primary O-ring had occurred in the aft field joint of the left motor. Hot gas had also bypassed the primary seal in the left nozzle joint. Erosion of the primary seal had also occurred in the nozzle joint of the right motor.[6]

Under the terms of the FRR Policy Directive, such damage would appear to require discussion: "the scope of the review should

[3] Ibid., pp. 2-3.
[4] Table I (page 44) indicates the date and scope for each of these eight reviews.
[5] Thiokol, "STS-51L (STS-33) Solid Rocket Motor (SRM-25) Flight Readiness Review," TWR-15380, December 11, 1985, chart 1-1. See Appendix VIII-B.
[6] Rogers Commission Report, Volume II, p. H-3.

cover status and issues in areas such as . . . prior flight anomalies. . . ." [7]

However, according to Mr. McDonald and Mr. Kennedy, Thiokol normally took about one week to prepare a discussion of problems noted in the initial inspection following SRB hardware disassembly.[8]

It would seem logical, when faced with the lack of data from the previous hardware set, to expand the search to other previously-flown hardware. On 61-A, hot gas had bypassed the primary seals in both the center and aft field joints of the left motor.[9]

The right motor suffered erosion of the primary O-ring in the nozzle joint.[10]

The SRBs from 61-B suffered erosion of the seals in both nozzle joints, with gas bypassing the primary seal of the left nozzle.[11]

The Associate Administrator's policy directive is not alone in stressing that any available information capable of assisting with an assessment of flight readiness should be presented at a readiness review. Marshall's Shuttle Projects Office policy guidance uses virtually identical language. Under "Shuttle Policy Guidance," it states, "*Review Concept*: The Shuttle Projects FRR will employ a delta review concept from prior reviews and previous STS missions." [12]

In his letter announcing the STS 51-L Marshall Center FRR, Dr. Lucas wrote:

> Each project manager must certify the flight readiness of his hardware and present supporting rationale and data so the Board can independently assess the flight readiness . . . Emphasis will be placed on safety of flight and mission success, including potential impact of prior flight anomalies.[13]

Apparent in the STS 51-L process, however, is that the continuing SRM seal problem did not receive such treatment. The "delta review concept" referred to above, according to Mr. McDonald, meant that the contractor was obligated to step back only to the previous mission for comparison.[14]

For 51-L, there was no previous mission to compare data with, since 61-C had not yet flown. Anomalies on STS 61-A and 61-B were not discussed, Mr. McDonald said, because they had already been dispositioned in the FRR's for 61-B and 61-C.[15]

The Marshall Space Flight Center FRR conducted by Dr. Lucas occurred only one day after the 61-C launch. Mulloy's presentation

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[7] SFO-PD 710.5A, p. 3.

[8] Discussions with Allan McDonald and Carver Kennedy, Thiokol (Wasatch Operations), Brigham City, Utah, September 4, 1986.

[9] Rogers Commission Report, *loc. cit.*

[10] Ibid.

[11] Ibid.

[12] Robert Lindstrom, NASA, Marshall Space Flight Center, "Shuttle Project Flight Readiness Review," SOP 8000.1, December 29, 1983, p. 2. See Appendix VIII-C.

[13] William Lucas, NASA, Marshall Space Flight Center, "MSFC Flight Readiness Review (FRR) Board for MSFC Elements for Mission 51-L," January 7, 1986, pp. 1-2. See Appendix VIII-D.

[14] Discussion with Mr. McDonald, September 4, 1986.

[15] Ibid.

under STS 61-C performance noted that "all SRB systems functioned normally."[16]

Under "ascent," the chart shows "no anomalies."[17]

There is no indication in the documentation for this FRR that the continuing problem with the SRM seals was raised. The parachute recovery system was discussed at some length. Much of the presentation appears to be drawn from the booster assembly presentation made at Mulloy's Level III FRR on January 3. The only relevant item that might refer to the O-rings appears under "Certification/Verification Status," where Mulloy stated that there were "no findings from continuing analyses that changes previously established rationale for flight."[18]

Mr. Mulloy's presentation at the January 15, 1986, Level 1 FRR does not indicate any serious problems with the SRB's. Documentation under "Problems/Anomalies" lists "[n]o 61-C flight anomalies."[19]

Again, the focus of his presentation involved the changes made in the parachute recovery system. The SRM booster nozzle on STS 51-L would be separated at the apogee of the SRB flight path (following separation of the boosters from the Shuttle vehicle) to protect the drogue parachute from debris, and the main parachutes were to be separated at water impact to reduce risks to the divers that assisted with recovery.[20]

Mulloy's presentation to the Associate Administrator was not noticeably different from the presentation be made to Mr. Aldrich, the Shuttle Program Manager, at the Level II readiness review the day before. In fact, the briefing charts are identical.[21]

SRM seal erosion was ultimately raised during the STS-51L Flight Readiness Review cycle. Mr. McDonald stated that at the L-1 FRR Mr. Mulloy informed the Mission Management Team of the erosion damage seen on STS 61-C, characterizing it as "within the experience base."[22]

This evaluation of the seal erosion problem does not indicate the seriousness of the issue, and would not lead senior managers to a conclusion that the seal problem was a threat to the safety of the Shuttle. Given the historical treatment of the SRM seal problem in this process, however, it is not surprising that the STS 51-L reviews did not raise any new concerns about the integrity of the joint.

This point is readily apparent in the Commission's report. There is no implication that a serious problem exists, if Mr. Mulloy's presentations are examined. The presentation made to Level 1 during the STS 41-C FRR indicated that erosion was "acceptable," and offered a rationale for accepting the possibility that the phenomenon would recur.[23]

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[16] Larry Mulloy, NASA, Marshall Space Fight Center, "Center Board: STS-51L Flight Readiness Review Solid Rocket Booster," January 13, 1986, Chart SRB-3.

[17] Ibid.

[18] Ibid, Chart SRB-28.

[19] Larry Mulloy, NASA, Marshall Space Flight Center, "STS-51L Level I Flight Readiness Review," January 15, 1986, Chart SRB-3. See Appendix VIII-E.

[20] Ibid., Chart SRB-4.

[21] Larry Mulloy, NASA, Marshall Space Flight Center, "STS-51L Level II Flight Readiness Review," January 14, 1986. See Appendix VIII-F.

[22] Discussion with Allan McDonald, September 4, 1986.

[23] Rogers Commission Report, Volume II. See Chart 15 (p. H-10) and Chart 19 (p. H-12).

Indeed, Level 1 displayed more concern about the problem than did Marshall, since Dr. Hans Mark, then Deputy Administrator, directed Marshall to prepare a review of the seal erosion problems.[24]

For mission 41-G, Mulloy argued that "test shows maximum erosion possible less than erosion allowable."[25]

In his presentation to the STS 51-E Level 1 FRR, Mulloy is seen on a videotape stating that:

> The rationale that was developed after observing this erosion on STS-11 [41-B] was that it was a limited duration that was self-limiting in that as soon as the pressure in the cavity between the putty and the primary O-ring after the primary O-ring seats, or the pressure between the primary and the secondary O-ring equals the motor pressure, the flow stops and the erosion stops. The maximum erosion that we have seen previously is 53 thousandths [of an inch]—that was back on STS-2. The erosion that we saw on 51-C was 10 thousandths of an inch on one O-ring and 38 thousandths on the other, so we believe that because of the limited exposure and the fact that the leak check assures that the secondary O-ring is properly sealing against motor pressure and the fact that the duration is limited, and that we can take 95 thousandths erosion on a primary O-ring and seal against 3,000 psi which is three times the motor pressure, that this represents an acceptable risk.[26]

Mulloy's confidence that the SRM seal could take "95 thousandths erosion on a primary O-ring and seal against 3,000 psi which is three times the motor pressure, . . ." is based on computer modelling of the joint performance. Dr. Feynman's analysis of the model, however, questions its use as the basis for declaring the seal problem "an acceptable risk."

> . . . This was a model based not on physical understanding but on empirical curve fitting. To be more detailed, it was supposed a stream of hot gas impinged on the O-ring material, and the heat was determind at the point of stagnation (so far, with reasonable physical thermodynamic laws). But to determine how much rubber eroded it was assumed this depended only on this heat by a formula suggested by data on a similar material. A logarithmic plot suggested a straight line, so it was supposed that the erosion varied as the .58 power of the heat, the .58 being determined by a nearest fit. At any rate, adjusting some other numbers, it was determined that the model agreed with erosion (to depth of one-third the radius of the ring). There is nothing much so wrong with this as believing the answer! Uncertainties appear everywhere. How strong the gas stream might be was unpredictable, it depended on holes formed in the putty. Blow-by showed that the ring might fail even though not, or only partially eroded

[24] Ibid., p. H-13.
[25] Ibid., Chart 30 (p. H-18).
[26] Ibid., p. H-42.

> through. The empirical formula was known to be uncertain, for it did not go directly through the very data points by which it was determined. There were a cloud of points some twice above, and some twice below the fitted curve, so erosions twice predicted were reasonable from that cause alone. Similar uncertainties surrounded the other constants in the formula, etc., etc. When using a mathematical model careful attention must be given to uncertainties in the model.[27]

Mr. Mulloy's analysis of erosion also notes that it was a "self-limiting" phenomenon, assuming that the damage ceased after the pressure built up against the seal. His analysis demonstrates that either the primary or the secondary seal would serve the purpose.

On December 17, 1982, an amended version of the SRB Critical Items List was approved. It stated, "Leakage of the primary O-ring seal is classified as a single failure point due to possibility of loss of sealing at the secondary O-ring because of joint rotation *after motor pressurization.* [emphasis added]"[28]

In the "Rationale for Retention," the document states, "Full redundancy exists at the moment of initial pressurization."[29]

Mr. Mulloy read this to indicate that during the ignition transient, the seal was a Critically 1R system, a redundant seal existed, and the secondary O-ring could be relied upon. After completion of the ignition transient (approximately 600 milliseconds), the joint became a Critically 1 system.[30]

Congressman Roe, however, said,

> We don't buy the point of view, do we measure other criticality points in degrees? My father taught me . . . [i]t is or it isn't . . . you took it from a R1 position and made it a number one position. You didn't qualify that, there is nothing in the record that qualifies it as half an R1 or three-quarters of an R1 in terms of temperature. . . . We didn't say we put them in there in number of degrees. We either did or we didn't.[31]

Implied in the presentation by Mr. Mulloy is that the Marshall and Thiokol engineers understood the joint's performance during the ignition transient. But, as Congressman Volkmer noted,

> Mr. VOLKMER. ". . . Mr. Mulloy, it says on page 148 [of the Commission's report] that prior to the accident neither NASA or Thiokol clearly understood the mechanism by which the joint sealing took place. Do you agree or disagree with that?"
>
> Mr. MULLOY. "I totally agree, sir."[32]

Also notable by its absence in Mulloy's presentation is the fact that STS 51-L had demonstrated an extreme example of blowby in the nozzle joint. It was this case that led Thiokol engineers to the

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[27] Rogers Commission Report, Volume II, p. F-2.
[28] NASA, "SRB Critical Items List," Page A-6A, December 17, 1982, Sheet 1.
[29] Ibid.
[30] Rogers Commission Report, Volume V, p. 834.
[31] Cmte Hgs., Transcript, June 17, 1986, p. 185.
[32] Ibid., p. 291.

conclusion that temperature was a contributing factor to joint damage. Mr. Boisjoly explained to General Kutyna that, based on photographs of the joints on 51–C (launched at a seal temperature of 53°F) and 61–A (launched at a seal temperature of 75°F), he "concluded, and so presented on the night before the launch . . . that it was telling us that temperature was indeed a discriminator . . ."[33]

The appearance of the material that had bypassed the joint, according to Boisjoly, was significantly worse for 51C in appearance and extent.

For mission 51–F, even after the failure of the primary seal in the 51–B nozzle joint, Mulloy's presentation to Level 1 listed the problem as "closed."[34]

Chairman Roe, questioning witnesses from NASA on 17 June, learned that managers at Johnson and at Headquarters had not necessarily perceived the seriousness of the situation represented by the seal problem.

> Mr. Roe. "I would like to get Mr. Mulloy to answer the question—would you repeat the nine flights and tell the Committee at what level the O-ring problem was discussed and who was at that level? . . . You mentioned again, you listed the whole nine, and tell the Committee at what level the O-ring problem was discussed. We have been going on this for seven years and then who was at that meeting?"
>
> Mr. Mulloy. "Yes sir. I can answer part of your question. . . . I am reading from what was provided to me. It looks like it fits within the erosion. STS–11 [41–B], 41–C, 41–G, 51–E . . ., 51–F . . . , 51–I, 51–J, 61–A . . ., and 61 Bravo."
>
> Mr. Roe. "These were a problem with the O-rings and they were discussed at Level 1?"
>
> Mr. Mulloy. "Level 1 and Level 2."
>
> Mr. Roe. "Therefore it is inconceivable that Level 1, which is top management, would not have understood the issue?"
>
> Mr. Mulloy. "That is right, and I believe that has been acknowledged. . . ."
>
> Dr. Graham. "We are in fact, reviewing the records to see who was at the various Flight Readiness Reviews that occurred when the O-ring data was mentioned, and we have not yet been able to pull that together. . . ."
>
> Mr. Roe. "So what you are basically saying is that Washington level knew of part of the problems; is that a fair comment?"
>
> Dr. Graham. "There are two pieces to this: one, what was transmitted; and what was understood. I believe what Mr. Mulloy and Dr. Lucas are addressing is what was transmitted. I don't know that they are the most appropriate people to express what was understood. That was a Headquarters issue and, in some cases, a Johnson Space

[33] Rogers Commission Report, Volume V, p. 784.
[34] Ibid., Chart 130 (p. H–66).

> Center issue. It is clear the issue was not perceived at the seriousness with which it actually affected the system. However, the information was transmitted to these agencies." [35]

Dr. Graham's statement is important when discussing the August 19, 1985, briefing on the joint seal problem. Thiokol and Marshall personnel did not communicate that the situation required a halt in operations until the problem of seal erosion had been solved. It should also be noted that Mr. Moore was then occupied with the failure of the SSME temperature sensors (a failure which had led to premature shutdown of a main engine on flight STS 51-F and caused the first abort-to-orbit in the program's history), and so the briefing was attended by Mr. L. Michael Weeks, Deputy Associate Administrator (Technical) for Space Flight. A more complete analysis of his descripton of the situation to Mr. Moore is discussed in the section on Technical Expertise.[36]

Mr. Aldrich was not made aware of the briefing at all, removing Level 2 from the information flow.[37]

Levels 1 and 2 were not alone in their misapprehensions. the lack of understanding of the seal problem also appears in the presentations to Dr. Lucas and Mr. Reinartz at Marshall made by Mr. Mulloy. In the STS 61-C FRR cycle, immediately preceding 51-L, Mr. Mulloy was given an extensive discussion of the information obtained from STS 61-B, describing the damage to the seals, at the Level 3 SRB Project Office briefing he chaired.[38]

Mulloy's presentation to the Shuttle Projects Board then noted "SRM joint O-ring performance within experience base."[39]

In his presentation to the Level 1 FRR, however, Mulloy stated there were "no 61-B flight anomalies."[40]

In hindsight, a fundamental error that pervades the history of the seal erosion problem is this reliance on the "experience base" argument. Unwarranted confidence existed in the analysis of the joint seal erosion problem developed by Thiokol engineers and agreed to by Marshall's program office. There is a vital lesson to be learned in this episode, and it is best expressed by Henry Petroski, from the School of Engineering at Duke University.

> *Dismissing the single structural failure as an anomaly is never a wise course* (emphasis added). The failure of any engineering structure is cause for concern, for a single incident can indicate a material flaw or design error that renders myriad structural successes irrelevant. . . . In en-

[35] Cmte Hgs, Transcript, June 17, 1986, pp. 203-206.

[36] Section VI.B.1.c. of this report.

[37] "The second breakdown in communications. . . ," Mr. Aldrich testified before the Commission, "is the situation of the variety of reviews that were conducted last summer between the NASA Headquarters Organization and the Marshall Organization on the [joint seal problem] and the fact that that was not brought through my office in either direction—that is, it was not worked through by the NASA Headquarters Organization nor when the Marshall Organization brought these concerns to be reported were we involved. And I believe that is a critical breakdown in process and I think it is also against the documented reporting channels that the program is supposed to operate to." Rogers Commission Report, Volume V, p. 1490.

[38] Larry Wear, NASA, Marshall Space Flight Center, "Flight Readiness Review SRM-24 (STS 61-C)," December 2, 1985, Charts 3-2; 3-2B. See Appendix VIII-G.

[39] Rogers Commission Report, Volume II, p. H-3.

[40] Ibid.

gineering, numbers are means, not ends, and it ought rightly to have taken only the failure of a single bridge to bring into question the integrity of every other span. . . . The common expectation of the engineer and the layman is that the road will not lead to bridges that collapse.[41]

Clearly evident is the fact that the Flight Readiness Review procedure cannot compensate for poor engineering analysis. The FRR is similar to a checklist and will not necessarily discover problems not on the list. The technical rationale presented by Level 3 managers is assumed to reflect the best engineering judgment available. Relying on this expertise, managers in more senior positions at NASA were misinformed regarding the severity of the problem of seal erosion and its critical importance to flight safety.

The Committee is also concerned about the so-called "launch constraint" imposed on the Shuttle system following STS 51-B, and the role this constraint was expected to play in the Flight Readiness Review process. The term "launch constraint" would seem to indicate that the Shuttle should not be launched until the problem giving rise to the constraint was solved. This is apparently not the case, according to Mr. Mulloy:

> The problem assessment system is in place at the Marshall Center as a tool to assure—it is a tool used by our quality and a reliability assurance organization to assure that problems that occur in flights and in ground test, in development, our qualification motor tests that would have a bearing on the flight or the upcoming flights, that that is documented and tracked. That problem assessment system shows in the case of the O-ring erosion, it shows essentially the same information, in many cases identical information to what is in the Flight Readiness Reviews. It is the basis for continuing to fly given the observations we are seeing.[42]

Testifying before the Commission, Mulloy had also made this distinction.

> Chairman ROGERS. "Let's go back just a bit, because I think it is helpful to me if you—you use words that I understand a little bit. What caused the constraint to be put on in the first place?"
>
> Mr. MULLOY. "The constraint was put on after we saw the secondary O-ring erosion on the nozzle, I believe."
>
> Chairman ROGERS. "Who decided that?"
>
> Mr. MULLOY. "I decided that, that that [the joint seal erosion] would be addressed, until that problem was resolved, it would be considered a launch constraint, and addressed at Flight Readiness Reviews to assure that we were staying within our flight experience base. . . ."
>
> Dr. RIDE. "Why didn't you put a launch constraint on the field joint at the same time?"

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[41] Henry Petroski, *To Engineer is Human: The Role of Failure in Successful Design* (New York: St. Martin's Press, 1985), pp. 69–73.
[42] Cmte Hgs, Transcript, June 17, 1986, p. 205.

Mr. Mulloy. "I think at that point . . . the logic was that we had been discussing the field joint, the field and nozzle joint primary O-ring erosion. This erosion of [STS 51–B's] secondary O-ring was a new and significant event, very new and significant event that we certainly did not understand. Everything up to that point had been that the primary O-ring, even though it does experience some erosion, does seal. What we had evidence of was that here was a case where the primary O-ring was violated and the secondary O-ring was eroded, and that was considered to be a more serious observation than previously observed." [43]

The Marshall Space Flight System Problem Assessment System (PAS) was tracking the problem of nozzle joint primary O-ring erosion in Record A09288, "O-Ring Erosion in the Case to Nozzle Joint."[44] The record was apparently opened on July 10, 1985, some two months following the launch of STS 51–B on April 29, 1985. The last entry in this record is dated January 23, 1986, and begins "Resolution." It continues with a rationale for closing out the tracking record. Part of the rationale is quoted here:

> Analytical studies based on both impingement erosion and blowby erosion show that this phenomenon has an acceptable ceiling since implementing the above changes [in performance of the seal leak check and in stacking procedures]. Recent experience has been within the program data base. The seal improvement program plan will continue until the problem has been isolated and damage eliminated to the SRM seals.[45]

An identical entry appears in PAS Record A07934, "Segment Joint Primary O-ring Charred." Though tracking problems with the field joint seals, work on the nozzle joint problem was included "as they are the same generic problem."[46]

The logic behind this "resolution" of the O-ring problem is not readily apparent. As the field joint tracking report notes, "The O-rings in the SRM segment ass[embl]y joints are designed as press[ure] seals & are not intended to be exposed to hot gases."[47]

Yet, in the rationale for closing out these tracking reports, it is stated that "[p]rimary O-ring erosion is expected to continue since no corrective action has been established that will prevent hot gases from reaching the primary O-ring cavity."[48]

The rationale ad described also appears to violate the directive, issued in 1980, that addresses the question of launch constraints. "All open problems coded criticality, 1, 1R, 2 or 2R," it stated:

> will be considered launch constraints until resolved (recurrence control established and its implementation effectivity determined) or sufficient rationale, i.e., different con-

[43] Rogers Commission Report, Volume V, p. 1510.

[44] NASA, Marshall Space Flight Center, "O-Ring Erosion in the Case to Nozzle Joint," Problem Assessment System Record Number A09288, February 26, 1986. See Appendix VIII–H.

[45] Ibid., p. 2.

[46] NASA, Marshall Space Flight Center, "Segment Joint Primary O-ring Charred," Problem Assessment System Record Number A07934, March 7, 1986, p. 4. See Appendix VIII–I.

[47] Ibid., p. 1.

[48] Ibid., p. 5.

figuration, etc., exists to conclude that this problem will not occur on the flight vehicle during prelaunch, launch or flight.[49]

The Committee attributes this situation to the concept of "acceptable erosion," which is more fully discussed in Chapter VII of this report.

According to testimony before the Commission, these tracking records for O-ring erosion were closed out upon receipt of a letter from Mr. McDonald dated December 10, 1985.[51]

This was apparently a mistake. As Mr. Mulloy and Mr. Wear explained to the Commission,

> Mr. MULLOY. ". . . Now, the entry that is shown in there that the problem was closed prior to 51-L is in error. What happened there was, one of your documents here which we did not discuss is the letter from Mr. McDonald to Mr. Wear which proposed that this problem be dropped from the problem assessment system and no longer be trapped [tracked] for the reasons stated in Mr. McDonald's letter. That letter was in the review cycle . . . . After Mr. Wear brought this letter to my attention, my reaction was, 'we are not going to drop this from the problem assessment system because the problem is not resolved and it has to be dealt with on a flight-by-flight basis.' Since that was going through the review cycle, the people who run this problem assessment system erroneously entered a closure for the problem on the basis of this submittal from Thiokol. Having done that, then for the 51-L review, this did not come up in the Flight Readiness Review as an open launch constraint, so you won't find a project signature because the PAS system showed the problem was closed, and that was an error."
>
> Chairman ROGERS. "Who made the error? Do you know?"
>
> Mr. MULLOY. "The people who do the problem assessment system."
>
> Mr. WEAR. "Mr. Fletcher, and he reports within our quality organization at the Flight Readiness Review, at the incremental Flight Readiness Reviews . . . . At my review and at Larry's review, there is a heads up given to the quality representative at that board for what problems the system has open, and they cross-check to make sure that we address that problem in the readiness review. On this particular occasion, there was no heads up given because their PAS system considered that action closed. That is unfortunate." [52]

Mr. Mulloy's discussion with Chairman Roe, and his description provided to the Commission, indicate that the NASA Safety, Reliability and Quality Assurance (SR&QA) organization should play a

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[49] Robert Lindstrom, NASA, Marshall Space Flight Center, "Assigning Launch Constraints on Open Problems Submitted to MSFC PAS," September 15, 1980, p. 1. See Appendix VIII-J.
[51] Rogers Commission Report, Volume V, p. 1509.
[52] Ibid.

64-420 0 - 86 - 8

significant role in the Flight Readiness Review process. The information available on this topic suggests that this was not the case. The Committee is concerned by the invisibility of SR&QA in this area.[53]

First, the number of people operating in the SR&QA organization has apparently been declining since the Apollo program. The decline at Marshall is among the most severe, according to NASA's internal estimate.[54]

The obvious conclusion is that SR&QA has fewer people to oversee the myriad details involved in preparing for Flight Readiness Reviews, in addition to their other duties.

Second, there is nothing to show what evaluation SR&QA personnel had made of the joint seal erosion problem. The recurring nature of this problem, and the Criticality-1 status of the joint, argue that SR&QA should have paid close attention to this situation, including the execution of an independent test program and presentations of their evaluation at Level 3 Flight Readiness Reviews.

Third, though management of the PAS system is apparently the responsibility of the SR&QA organization, they appear to exercise little control. The system is operated under contract by Rockwell, data is entered by hardware manufacturers, and the only technical analysis that appears in the reports on joint seal erosion was developed by Mr. Wear or his deputy, James Thomas, in the SRM program office. There is no input, either concurrence or dispute, registered by SR&QA personnel. Even more important, as illustrated by Mr. Mulloy's testimony, problem reports can too easily be removed from the system.

Fourth, the PAS tracking records do not support the testimony of Mr. Mulloy and Mr. Wear before the Commission. The entry entitled "Resolution" is dated January 23, 1986, while the FRR in Mr. Wear's SRM Project Office occurred on December 17, 1985, and the FRR in Mr. Mulloy's SRB Project Office took place on January 3, 1986. It is also interesting to note that the PAS Record, dated February 26, 1986, shows "Status Open."[55]

If Mr. Mulloy's and Mr. Wear's testimony is accurate, SR&QA should still have raised the issue of seal erosion as a concern at their reviews. No evidence exists that this occurred.

The Committee, however, is concerned not only about the SR&QA organization at Marshall. The Commission noted in its report that "[t]he Problem Reporting and Corrective Action document (JSC 08126A, paragraph 3.2d) requires project offices to inform Level II of launch constraints. That requirement was not met. Neither Level II nor Level I was informed."[56]

Testifying before the Committee, however, Mr. Mulloy argued that both levels were informed.

> Mr. VOLKMER. "Even though . . . you continued to see erosion of the O-ring, you continued to waive the launch constraint?"

[53] See also "Safety, Reliability and Quality Assurance," Section VI.B.2.c(2) of this report.
[54] This is documented in Section VI.B.2.c(1) of this report.
[55] PAS Record A09288, p. 1.
[56] Rogers Commission Report, Volume I, p. 159.

Mr. MULLOY. "That is correct, sir, on the basis of the rationale or the explanation as to why that was an acceptable risk that was presented to me by Morton Thiokol, reviewed and approved by my management. . . ."

Mr. VOLKMER. "Was JSC [the Johnson Space Center] informed of the problem?"

Mr. MULLOY. "Through the Flight Readiness Review and through the submission of this problem to the problem tracking system at JSC. I do not know what distribution was made at JSC when it goes down there. The report also goes to the Chief Engineer's office at Headquarters."

Mr. VOLKMER. "It is my understanding that we had some testimony earlier from Mr. Aldrich that he wasn't knowledgeable that there was a launch constraint."

Mr. MULLOY. "That is entirely possible, sir, I don't know what distribution was made, and I have testified, and I think—I have testified that it wasn't briefed in the Levels 2 and the Level 1. When I went—"

Mr. VOLKMER. "That is right."

Mr. MULLOY. "—that 'we have a problem, the concern is flight safety, the rationale for continuing to fly is this.' That was not briefed in the context of 'this is a launch constraint in the problem assessment system,' and it is entirely possible that if that report, whatever distribution is made of that report at Houston, that he might not have seen that."[57]

It is quite likely that neither Mr. Moore nor Mr. Aldrich was made aware of the launch constraint on the SRM. SR&QA personnel at Johnson and at NASA Headquarters should have received these reports described by Mr. Mulloy and tracked them as they did similar launch constraints on other Shuttle hardware. What steps they took to assure that these constraints were raised at the Flight Readiness Review for these management levels is less clear. The Committee's review of the FRR's for the six missions subject to the launch constraint on SRM nozzle joint seals does not indicate a greater level of discussion took place because the constraint was in force, except for the STS 51-F FRR where an explanation of the 51-B failure was required.

The rationale for closing out these problems on the problem assessment system stated that "status will continue to be provided in the Flight Readiness Reviews and in formal technical reviews at Thiokol and MSFC." [58]

Without a change in the prevailing technical evaluation of the problem, however, it is unlikely that proper action to correct the overall problem would have been undertaken.

On the eve of the launch, Thiokol engineers attempted to change this prevailing technical evaluation. In a teleconference on the night of January 27, they presented data demonstrating that the low temperatures in the area would impair the function of the O-ring seals inside the joint. After NASA managers expressed con-

[57] Cmte Hgs Transcript, June 17, 1986, pp. 284-85.
[58] PAS Record A09288, p. 3.

cern about the delay such constraints would have on the flight schedules, Thiokol's management withdrew and met with their engineers again; only this time, the managers would not listen to further argument about aborting the flight. Thiokol then recommended that the launch be allowed to proceed.

Launch operations were terminated at 12:36 p.m. Eastern Standard Time (EST) on January 27 because of high crosswinds at the launch site. At 2:00 p.m. EST, the Mission Management Team met and decided to attempt a launch at 9:38 a.m. EST January 28. The weather was expected to be clear but cold with temperatures in the low twenties. There were concerns about the facilities and various water drains but no concerns were expressed about the O-ring and the Solid Rocket Boosters. All members of the team were asked to review the situation and call if any problems arose.

At Thiokol's Wasatch Division in Utah, Mr. Robert Ebeling [59] met with Mr. Boisjoly at about 2:30 p.m. Mountain Standard Time (MST) on the afternoon of January 27. They were joind by other Thiokol engineers. Mr. Ebeling was concerned about predicted cold temperatures at the Kennedy Space Center. When he was questioned by the Rogers Commission he responded:

> . . . The meeting lasted one hour, but the conclusion of that meeting was engineering, epecially Arnie [Thompson], Roger Boisjoly, Brian Russell, myself, Jerry Burns, they come to mind, were very adamant about their concerns on this lower temperature, because we were way below our data base and we were way below what we qualified for.[60]

Later Mr. Ebeling called Mr. McDonald at the Kennedy Space Center. Mr. McDonald remembered the call, saying:

> He called me and said they had just received some word earlier that the weatherman was projecting temperatures as low as 18 degrees F [Fahrenheit] sometime in the early morning hours of the 28th and that they had some meeting with some of the engineering people and had some concerns about the O-rings getting to those kinds of temperatures.[61]

Mr. Ebeling wanted Mr. McDonald to get some accurate predicted temperatures for the Cape so he could make some calculations to determine what could be expected of the O-rings. McDonald told him he would get the temperature data for him and call him back. Mr. Carver Kennedy, Vice President of Space Services for Thiokol, working at the Kennedy Space Center, obtained the information. Mr. McDonald then relayed the information to Mr. Ebeling in Utah. The information indicated that the temperature was to get as low as 22° in the early morning hours, probably around 6:00 a.m., and that they were predicting a temperature of about 26° at the intended time of launch, 9:38 a.m. on the 28th.[62]

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[59] Solid Rocket Motor Igniter and Final Assembly Manager, Thiokol.

[60] Thiokol, Robert Ebeling, Interview before the Presidential Commission on the Space Shuttle Challenger Accident, March 19, 1986, p. 13.

[61] Rogers Commission Report, Volume IV, p. 715.

[62] Ibid.

Mr. McDonald then called Mr. Cecil Houston, the Resident Manager for the Marshall office at Kennedy Space Center, and told him of Thiokol's concerns with the low temperature and potential problems with the O-rings. Mr. Houston said he would set up a teleconference including Marshall and personnel at Thiokol in Utah.[62a]

[62a] Table II lists the principal participants in the teleconference on January 27, 1986.

TABLE II

PRINCIPAL PARTICIPANTS IN THE TELECONFERENCE ON JANUARY 27, 1986

At Kennedy Space Center, Florida

Thiokol	Allan McDonald	Director, Solid Rocket Motor Program Office
Thiokol	Jack Buchanan	Manager, KSC Resident Office
MSFC	Lawrence Mulloy	Manager, Solid Rocket Booster Project Office
MSFC	Stanley Reinartz	Manager, Shuttle Projects Office
MSFC	Judson Lovingood	Deputy Manager, Shuttle Projects Office

At Marshall Space Flight Center, Alabama

MSFC	George Hardy	Deputy Director for Science and Engineering

At Thiokol Wasatch Operations, Utah

Thiokol	Jerald Mason	Senior Vice President, Wasatch Operations
Thiokol	C.G. Wiggins	Vice President and General Manager, Space Division
Thiokol	Robert Lund	Vice President for Engineering
Thiokol	Joseph Kilminster	Vice President, Space Booster Programs
Thiokol	Roger Boisjoly	Staff Engineer, Applied Mechanics
Thiokol	Robert Ebeling	Manager, SRM Ignition System, Final Assembly, Special Projects and Test
Thiokol	Arnold Thompson	Supervisor, Structures Design

Mr. Houston then called Dr. Judson Lovingood, Deputy Shuttle Project Manager at Marshall, to inform him of the concerns about the O-rings. Mr. Houston asked Dr. Lovingood to set up a teleconference with senior project management personnel, including Mr. George Hardy, Deputy Director of Science and Engineering at Marshall, and with Thiokol personnel. Dr. Lovingood called Mr. Stanley Reinartz, Marshall's Shuttle Project Office Manager, a few minutes later and informed him of the planned teleconference.

The first phase of the teleconference began at 5:54 p.m. EST and included Messrs. Reinartz, Lovingood, Hardy, and others at Kennedy, Marshall and Thiokol's Wasatch plant. Concerns about the effect of low temperature of the O-rings and the joint seal were presented by Thiokol personnel, along with an opinion that launch should be delayed.

A recommendation was also made that Arnold Aldrich, the Space Transportation System Program Manager, be told of the upcoming telecon and that the fact that Thiokol had expressed some concerns. Mr. Reinartz testified before the Commission that "we did not have a full understanding of the situation as I understood it at that time, and felt that it was appropriate to do before we involved the Level II into the system." [63]

Testifying before the Rogers Commission, Dr. Lovingood was asked whether the possibility of a launch delay had been mentioned in this telcon on January 27. Dr. Lovingood replied:

> That is the way I heard it, and they were talking about the 51-C experience and the fact that they had experienced the worst case blow-by as far as arc and the soot and so forth. And also, they talked about the resiliency data that they had.
>
> So it appeared to me—and we didn't have all the people there. That was another aspect of this. It appeared to me we had better sit down and get the data so that we could understand exactly what they were talking about and assess that data.
>
> And that is why I suggested that we go ahead and have a telecon within the center, so that we can review that.[64]

Dr. Keel, the Staff Director for the Rogers Commission, asked,

> Dr. KEEL. "So as early as after that first afternoon conference at 5:45, it appeared that Thiokol was basically saying delay. Is that right?"
>
> Dr. LOVINGOOD. "That is the way it came across to me. I don't know how other people perceive it, but that's the way it came across to me."
>
> Dr. KEEL. "Mr. Reinartz, how did you perceive it?"
>
> Mr. REINARTZ. "I did not perceive it that way. I perceived that they were raising some questions and issues which required looking into by all the parties, but I did not perceive it as a recommendation to delay."
>
> Dr. KEEL. "Some prospect for delay?"

[63] Rogers Commission Report, Volume V, p. 919.
[64] Ibid., p. 923.

Mr. REINARTZ. "Yes, sir, that possibility is always there."

Dr. KEEL. "Did you convey that to Mr. Mulloy and Mr. Hardy before the 8:15 teleconference?"

Mr. REINARTZ. "Yes I did. And as a matter of fact, we had a discussion. Mr. Mulloy was just out of communication for about an hour, and then after that I got in contact with him, and we both had a short discussion relating to the general nature of the concerns with Dr. Lucas and Mr. Kingsbury at the motel before we both departed for the telecon that we had set up out at the Cape."[65]

At approximately 8:45 p.m. EST, the second phase of the teleconference commenced, Thiokol's charts and written data having arrived at the Kennedy Space Center by telefax. The charts presented a history of the O-ring erosion and blow-by in the Solid Rocket Booster joints of previous flights, presented the results of subscale testing at Thiokol and the results of static tests of Solid Rocket Motors.

Mr. Boisjoly testified:

> I expressed deep concern about launching at low temperature. I presented Chart 2-1 with emphasis—now, 2-1, if you want to see it, I have it, but basically that was the chart that summarized the primary concerns, and that was the chart that I pulled right out of the [August 19] Washington presentation without changing one word of it because it was still applicable, and it addresses the highest concern of the field joint in both the ignition transient condition and the steady state condition, and it really sets down the rationale for why we were continuing to fly. Basically, if erosion penetrates the primary O-ring seal, there is a higher probability of no secondary seal capability in the steady state condition. And I had two sub-bullets under that which stated bench testing showed O-ring not capable of maintaining contact with metal parts gap opening rate to maximum operating pressure. I had another bullet which stated bench testing showed capability to maintain O-ring contact during initial phase (O to 170 milliseconds of transient). That was my comfort basis of continuing to fly under normal circumstances, normal being within the data base we had.
>
> I emphasized, when I presented that chart about the changing of the timing function of the O-ring as it attempted to seal. I was concerned that we may go from that first beginning region into that intermediate region, from 0 to 170 being the first region, and 170 to 330 being the intermediate region where we didn't have a high probability of sealing or seating.[66]

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[65] Ibid.

[66] Ibid., Volume IV, p. 790.

Mr. Boisjoly then presented Chart 2-2 with his added concerns related to the timing function. He mentioned in his testimony to the Rogers Commission:

> We would have low O-ring squeeze due to low temperature which I calculated earlier in the day. We should have higher O-ring Shore hardness. . . . Now, that would be harder. And what that material really is, it would be likened to trying to shove a brick into a crack versus a sponge. That is a good analogy for purposes of this discussion. I also mentioned that thicker grease, as a result of lower temperatures, would have higher viscosity. It wouldn't be as slick and slippery as it would be at room temperature. And so it would be a little bit more difficult to move across it.
>
> We would have higher O-ring pressure actuation time, in my opinion, and that is what I presented. . . . These are the sum and substance of what I just presented. If action time increases, then the threshold of secondary seal pressurization capability is approached. That was my fear. If the threshold is reached, then secondary seal may not be capable of being pressurized, and that was the bottom line of everything that had been presented up to that point.[67]

Asked by Chairman Rogers, "Did anybody take issue with you?." Mr. Boisjoly responded:

> Well, I am coming to that. I also showed a chart of the joint with an exaggerated cross section to show the seal lifted off, which has been shown to everybody. I was asked, yes, at that point in time I was asked to quantify my concerns, and I said I couldn't. I couldn't quantify it. I had no data to quantify it, but I did say I knew that it was away from goodness in the current data base. Someone on the net commented that we had soot blow-by on SRM-22 [Flight 61-A, October, 1985] which was launched at 75 degrees, I don't remember who made the comment, but that is where the first comment came in about the disparity between my conclusion and the observed data because SRM-22 had blow-by at essentially a room temperature launch.
>
> I then said that SRM-15 [Flight 51-C, January, 1985] had much more blow-by indication and that is was indeed telling use that lower temperature was a factor. This was supported by inspection of flown hardware by myself. I was asked again for data to support my claim, and I said I have none other than what is being presented, and I had been trying to get resilience data, Arnie [Thompson] and I both, since last October, and that statement was mentioned on the net.[68]

This second phasse of the telecon on the evening of January 27 concluded with statements from Robert Lund, Thiokol's Vice President of Engineering. His conclusion at that time was that the Shut-

[67] Ibid., p. 791.
[68] Ibid.

tle should not fly outside Thiokol's database; that, is, that the O-ring seals should be above 53 degrees Fahrenheit before lift-off.

NASA participants in the telecon were not pleased with these conclusions and recommendations, according to Mr. Boisjoly and Mr. McDonald. Mr. Hardy, when asked what he thought about Thiokol's recommendation, was quoted to the effect that he was "appalled" at Mr. Lund's decision.[69]

Boisjoly also testified that Mr. Hardy said, "No, not if the contractor recommended not launching, he would not go against the contractor and launch."[70]

Shortly thereafter, Mr. Joseph Kilminster, Thiokol's Vice President for Space Booster Programs, was asked by NASA if he would launch and "he said no because the engineering recommendation was not to launch."[71]

Then, according, to Mr. Boisjoly, someone in Thiokol management asked for a five-minute caucus, and at that point Thiokol cut their speakerphone off.

> Chairman ROGERS. "Mr. Boisjoly, at the time that you made the—that Thiokol made the recommendation not to launch, was that the unanimous recommendation as far as you knew?"
>
> Mr. BOISJOLY. "Yes. I have to make something clear. I have been distressed by the things that have been appearing in the paper and things that have been said in general, and there was never one positive, pro-launch statement ever made by anybody. There have been some feelings since then that folks have expressed that they would support the decision, but there was not one positive statement for launch ever made in that room."[72]

Asked for his recollection of these incidents, Mr. McDonald commented,

> . . . And the bottom line was that the engineering people would not recommend a launch below 53 degrees F. The basis for that recommendation was primarily our concern with the launch that had occurred about a year earlier, in January of 1985, I believe it was 51–C.[73]

Mr. Mulloy testified:

> The bottom line of that, though, initially was that Thiokol engineering, Bob Lund, who is the Vice President and Director of Engineering, who is here today, recommended that 51–L not be launched if the O-ring temperature predicted at launch time would be lower than any previous launch, and that was 53 degrees. . . .[74]

At 10:30 p.m. EST, the teleconference between NASA and Thiokol was recessed. The off-net caucus of Thiokol personnel lasted ap-

[69] Ibid.
[70] Ibid.
[71] Ibid.
[72] Ibid.
[73] Ibid., Volume IV, p. 717.
[74] Ibid., Volume IV, p. 604.

proximately thirty minutes at the Wasatch office. Jerald Mason, Senior Vice President for Wasatch Operations, remembered that the conversation during the caucus centered around O-rings and the history of eroosion of the O-rings. Mr. Mason testified:

> Now, in the caucus we revisited all of our previous discussions, and the important things that came out of that was, as we had recognized, we did have the possibility that the primary O-ring might be slower to move into the seating position and that was our concern, and that is what we had focused on originally. . . . The fact that we couldn't show direct correlation with the O-ring temperature was discussed, but we still felt that there was some concern about it being colder.[75]

Ten engineers participated in the caucus, along with Mr. Mason, Mr. Kilminster, Mr. Lund and Mr. C.G. Wiggins (Vice President and General Manager for Thiokol's Space Division). Arnold Thompson [76] and Mr. Boisjoly voiced very strong objections to launch, and the suggestion in thieir testimony was that Lund was also reluctant to launch.

Mr. Boisjoly, in testifying before the Rogers Commission, stated:

> Okay, the caucus was started by Mr. Mason stating that a management decision was necessary. Those of us who opposed the launch continued to speak out, and I am specifically speaking of Mr. Thompson and myself because in my recollection he and I were the only ones that vigorously continued to oppose the launch. And we were attempting to go back and rereview and try to make clear what we were trying to get across, and we couldn't understand why it was going to be reversed. So we spoke out and tried to explain once again the effects of low temperature. Arnie actually got up from his position which was down the table, and walked up the table and put a quarter pad down in front of the table, in front of management folks, and tried to sketch out once again what his concern was with the joint, and when he realized he wasn't getting through, he just stopped.
>
> I tried one more time with the photos. I grabbed the photos, and I went up and discussed the photos once again and tried to make the point that it was my opinion from actual observation that temperature was indeed a discriminator and we should not ignore the physical evidence that we had observed.
>
> And again, I brought up the point that SRM-15 [Flight 51-C, January, 1985] had a 110 degree arc of black grease while SRM-22 [Flight 61-A, October, 1985] had a relatively different amount, which was less and wasn't quite as black. I also stopped when it was apparent that I couldn't get anybody to listen.[77]

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[75] Ibid., p. 759.
[76] Supervisor of Structures Design, Thiokol.
[77] Rogers Commission Report, Volume IV, pp. 792-93.

Commissioner Walker asked, "At this point did anyone else speak up in favor of the launch?" Mr. Boisjoly replied:

> No, sir. No one said anything, in my recollection, nobody said a word. It was then being discussed amongst the management folks. After Arnie and I had our last say, Mr. Mason said we have to make a management decision. He turned to Bub Lund and asked him to take off his engineering hat and put on his management hat. From this point on, management formulated the points to base their decision on. There was never one comment in favor, as I have said, of launching by any engineer or other nonmanagement person in the room before or after the caucus. I was not even asked to participate in giving any input to the final decision charts.
>
> I went back on the net with the final charts or final chart, which was the rationale for launching, and that was presented by Mr. Kilminster. It was hand written on a note pad, and he read from the notepad. I did not agree with some of the statements that were being made to support the decision. I was never asked or polled, and it was clearly a management decision from that point. . . .
>
> I left the room feeling badly defeated, but I felt I really did all I could to stop the launch.[78]

In testimony before the Committee on June 18, 1986, concerning the caucus and the decision to overrule engineering recommendations, Boisjoly said:

> When we went off the line and caucused—one of the first statements that was made was that we would have to make a management decision by management people. And we continued very strongly to oppose that and we argued as vigorously as we could argue, and when you look up into people's eyes you know you have gone about as far as you can go.
>
> And so both Mr. Thompson and I just plain frankly backed off. You had to be there and you had to see the looks and feel the experience that it didn't really make any difference what further you were going to say, you were just not going to be heard.[79]

At approximately 11 p.m. EST, the Thiokol/NASA teleconference resumed, with Mr. Kilminster stating that they had reassessed the problem, that the temperature effects were a concern, but that the data were admittedly inconclusive. He read the rationale recommending launch and stated that to be Thiokol's recommendation. Mr. Hardy of NASA requested that it be sent in writing by telefax both to Kennedy and to Marshall, and it was.[80]

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[78] Ibid.

[79] Cmte Hgs, Transcript, June 18, 1986, p. 85.

[80] The Committee has learned that (apparently due to an error in duplicating the relevant chart) the copy of this telefax that was sent to the Rogers Commission did not contain the standard caveat that was printed below the company logo on the original telefax. The caveat reads, "Information on this page was prepared to support an oral presentation and cannot be considered complete without the oral discussion." At the Committee hearings on June 18, 1986, Mr. U.

Continued

Having heard the debate and this decision, Mr. Reinartz accepted the conclusion and ended the teleconference. He asked whether anyone had any further concerns. None were expressed. Commissioners then asked what steps he had taken after the decision was made.

> Chairman ROGERS. "I guess the question that still lingers in my mind is, in the Navy we used to have an expression about going by the book, and I gather you were going by the book. But doesn't the process require some judgment? Don't you have to use common sense? Wouldn't common sense require that you tell the decisionmakers about this serious problem that was different from anything in the past?
>
> Mr. REINARTZ. "In looking at that one, Mr. Chairman, together with Mr. Mulloy when we looked at were there any launch commits, any Level II, as I perceived during the telecon, I got no disagreement concerning the Thiokol launch between any of the Level III elements, the contractor, with Mr. McDonald there. I felt that the Thiokol and Marshall people had fully examined that concern, and that it had been satisfactorily dispositioned based upon the evidence and the data that was supplied to that decision process on that evening, from that material, and not extraneous to what else may have been going on within Thiokol that I had no knowledge of."
>
> Chairman ROGERS. "Okay. Thank you. I'm sorry for the long interruption."
>
> Mr. REINARTZ. "Based upon—and as we skipped over it is only a point to illustrate, Mr. Chairman, that in our discussion about the parachute with KSC and Mr. Aldrich, was to indicate that there was a clear area there where we had a very direct responsibility to inform them of the situation, which Mr. Mulloy did. And after a discussion of that issue, Mr. Aldrich concluded that the launch should proceed in that nature. Based on the results of the meeting and the conclusions out of the meeting, Mr. Mulloy and I informed the Director of Marshall, Dr. Lucas, and the Director of Science and Engineering, Mr. Kingsbury, on the 28th of January—about 5:00—of the initial Thiokol concerns and engineering recommendations, the final Thiokol launch recommendation, that I felt had led to a successful resolution of this concern."
>
> General KUTYNA. "Could I interrupt for a minute? You informed Dr. Lucas. He is not in the reporting chain?"
>
> Mr. REINARTZ. "No, sir."
>
> General KUTYNA. "If I could use an analogy, if you want to report a fire you don't go to the mayor. In his position as center director, Dr. Lucas was cut out of the reporting chain, much like a mayor. If it was important enough to

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Edwin Garrison, President of the Aerospace Group at Thiokol, testified that the caveat at the bottom of the paper in no way". . . insinuates. . . that the document doesn't mean what it says." (Cmte Hgs, Transcript, June 18, 1986, p. 43.) After further investigation of this deletion, the Committee has concluded that it was not significant.

report to him, why didn't you go through the fire department and go up your decision chain?"

Mr. REINARTZ. "That, General Kutyna, is a normal course of our operating mode within the center, that I keep Dr. Lucas informed of my activities, be they this type of thing or other."

General KUTYNA. "But you did that at 5 o'clock in the morning. That's kind of early. It would seem that's important. Why didn't you go up the chain?"

Mr. REINARTZ. "No, sir. That is the time when we go in, basically go into the launch, and so it was not waking him up to tell him that information. It was when we go into the launch in the morning. And based upon my assessment of the situation as dispositioned that evening, for better or worse, I did not perceive and clear requirement for interaction with Level II, as the concern was worked any dispositioned with full agreement among all reasonable parties as to that agreement."

Chairman ROGERS. "Did I understand what you just said, that you told Dr. Lucas that all the engineers at Thiokol were in accord?

Mr. REINARTZ. "No, sir. What I told him was of the initial Thiokol concerns that we had and the initial recommendation and the final Thiokol recommendation and the rationale associated with that recommendation, and the fact that we had the full support of the senior Marshall engineering [81] and, as George has testified, to the extensiveness of the group of people we had involved in that telecon with the various disciplines, that those three elements made up the final recommendation."

Mr. HOTZ. "Mr. Reinartz, are you telling us that you in fact are the person who made the decision not to escalate this to a Level II item?"

Mr. REINARTZ. "That is correct, sir." [82]

According to NASA's Program Directive SFO-PD 710.5A, Mr. Reinartz may have been required to report the matter to the Associate Administrator for Space Flight. A portion of that directive reads as follows:

> Significant items occurring subsequent to the FRR will also be reported to the AA-SF. Actions that can be easily accomplished without safety, mission, or launch impact and do not violate flight vehicle or launch complex configuration integrity or cause basic changes to launch commit criteria, flight rules, flight plan, or abort and alternate mission plans, need not be reported.[83]

Was this telecon, and the decision reached, "significant?" NASA's request that the Thiokol decision be put in writing indi-

[81] Staff review of teleconference materials used by Marshall engineers Wilbur Riehl (Chief, Nonmetallic Materials Division) and John Miller (Technical Assistant to the SRM Manager) indicates that some of the Marshall engineering staff shared the concerns expressed by Thiokol engineers.

[82] Rogers Commission Report, Volume V, pp. 917–18.

[83] SFO-PD 710.5A, p. 3.

cates that MSFC personnel felt the situation was significant, since in effect Thiokol was reconfirming the flight readiness of the SRM.

It is also interesting to note, in light of the directive, that Mr. McDonald testified before the Commission to the effect that Mr. Mulloy had made some "fairly strong comments . . . about trying to institute new launch commit criteria." [84]

Mr. Mulloy responded to this by explaining what he had attempted to say.

> Mr. MULLOY. "The total context, I think, in which those words may have been used is, there are currently no Launch Commit Criteria for joint temperature. What you are proposing to do is to generate a new Launch Commit Criteria on the eve of launch, after we have successfully flown with the existing Launch Commit Criteria 24 previous times. With this LCC, i.e., do not launch with a temperature greater than [*sic*] 53 degrees, we may not be able to launch until next April. We need to consider this carefully before we jump to any conclusions. It is all in the context, again, with challenging your interpretation of the data, what does it mean and is it logical, is it truly logical that we really have a system that has to be 53 degrees to fly? . . ."
>
> General KUTYNA. "Mr. Mulloy, if in fact the criteria were 53 degrees, it would have an impact not only on this launch, but on the shuttle program. . . . It is a fairly important decision to say you can't launch below 53 degrees, isn't it?
>
> Mr. MULLOY. "Yes, sir, I agree with that. I cannot describe the impacts, but, as I say, based upon our previous experience and our actions in flying subsequent vehicles after 51C, I found that to be a surprising conclusion. . . ."
>
> Mr. SUTTER. ". . . [I]nstead of saying you have to wait until next April to launch, the thing that you do is you go and there were three different levels of improvements that were discussed. The thing to do then was to put those improvements in the program, not infer that these engineers are saying, we're throwing a ringer at you that says don't launch until next April. I think that is putting those engineers into a little bit of a hot seat. And if they're trying to do their job and say, hey, we ought to do something about this, there ought to have been more attention paid." [85]

The Rogers Commission report included the statement, "It is clear that crucial information about the O-ring damage in prior flights and about the Thiokol engineers' arguments with the NASA telecon participants never reached Jesse Moore or Arnold Aldrich, the Level I and II program officials, or J. A. Thomas, the Launch Director for 51-L." [86]

Based on the available evidence, the Committee is unable to conclude whether or not Mr. Moore or Mr. Aldrich, with the informa-

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[84] Rogers Commission Report, Volume IV. p. 721.
[85] Ibid., Volume V. pp. 843-45.
[86] Ibid., Volume I, p. 101.

tion available to them that day, would have reached a decision to stop the launch had they been informed of this meeting. It does appear, however, that a three-hour telecon, in which arguments are raised about launch commit criteria and the contractor is asked to reconfirm the flight readiness of his hardware fall under the definition of the policy directive. At the very least, the STS Program Manager should have been presented the new declaration of flight readiness with an explanation of why it had been developed. This should have been a necessary addition to the Certificate of Flight Readiness prepared after the Flight Readiness Review.

The Committee also reviewed tapes and transcripts of conversations that took place in the Firing Room on January 28th involving discussions of the threat posed by ice on the Fixed Service Structure. Kennedy Space Center managers, in response to the predicted low temperatures that would be seen in the hours before the launch of STS-51L, took action to protect Launch Complex 39B and the Shuttle from freezing and ice buildup. This involved implementing the "freeze protection plan" for launch pad facilities. According to a post-accident report:

> Two actions within the PLAN were intended to limit the ICE DEBRIS which potentially could cause damage to the Shuttle Vehicle during launch. The first action involved adding approximately fourteen hundred gallons of antifreeze into the overpressure water troughs. The water troughs in both SRB exhaust holes have a total capacity of 6,580 gallons. The resultant antifreeze to water ratio was calculated to be 21.3%. According to the manufacturer's specifications, solution protected against freezing down to an ambient temperature of 16 degrees F. The second action involved the draining, where practical, of all water systems. Several systems, such as Firex [fire extinguishing], Deluge, and emergency shower and eyewash, were not drained. These systems were opened slightly and allowed to trickle into drains. The trickling water was found to cause drain overflows. High wind gusts then spread the water over large areas and it then froze.[87]

Soon after the call-to-stations on 28 January, at approximately midnight, cameras on the pad allowed engineers in the Firing Room to see that the gantry was heavily encrusted with ice. Over the Engineering Support Room communications loops, the following conversation took place: [88]

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[87] NASA, Kennedy Space Center, "STS-33 (51-L) Ice/Frost Team Evaluation Report: ESS/RSS/MLP Deck/Pad Apron Icing," January 30, 1986, p. 1.

[88] Conversations were recorded from the Kennedy Space Center Operational Intercommunication System (OIS), which permits members of the launch crew to discuss problems that occur during the countdown, and permits them to contact various mission support facilities around the country. The transcripts provided to the Committee do not indicate the exact times at which the referenced conversations occurred, and so the flow of conversations has been reconstructed in an attempt to provide logical consistency.

Most of the transcripts in this section are drawn from OIS Channel 245, identified as the coordination channel for JSC/MSFC/KSC Engineering personnel in the Engineering Support Area at Kennedy Space Center. Conversations among Rockwell International personnel were obtained from OIS Channel 216, described as the coordination channel for JSC personnel at Johnson and Kennedy centers and Rockwell engineers in Downey, California. Transcript page numbers are those supplied by NASA.

Continued

BL. This is Bob on 245.

DIRECTOR. Go ahead, Bob.

BL. Did you ever find out any more about that water?

DIRECTOR. No, no I haven't. [89]

BL. Okay, I guess the thing, they don't know yet where that water's coming from and the NTD's [NASA Test Director] got some folks looking into it. I guess the thing you guys need to do is see if you can get certain cameras to look at the vehicle and to determine if there's any water getting on the vehicle that might freeze and cause problems.

DIRECTOR. I think what we need to do, Bob, we need to decide do we feel comfortable enough to let it keep running and forming ice up there, or we ought to stop and send somebody in there and try to shut it off.

BL. Yeah, okay.

DIRECTOR. Do you guys think we could form enough ice there to cause us any problem on liftoff or anything?

BL. I think you've already done that, Horace.

DIRECTOR. Well, then, we ought to stop and go out there and get the water shut off.

BL. Yeah, we're worried about an icicle up high—well, see, camera 108's on the 155 foot. . . .

ITL. 155? Yeah. So you're already getting high up, you know. And if the wind's going to be out of the north-northwest. . . .

DIRECTOR. All right, let's stop them and send the people out there and see if they can shut the water off.

ITL. I think what it is, is the fire hose—if you look right over, if you go to 108 and go like you're going to the elevator, you'll see a fire hose, looks like a fire hose draped across there...

DIRECTOR. Yeah.

ITL. . . . And I think they take the fire hose and carry it over to the shower, the eye shower. And evidently the drains that they're draining into is frozen off, or either the hose has fallen off the drain, one or the other.

DIRECTOR. Okay, we gotta work this in. We're going to tell them to go out there and shut the water off.

ITL. Okay.[90]

The ice/frost team was dispatched to the pad and arrived at approximately 1:45 a.m. What they found during their inspection of the Fixed Service Structure was not very encouraging to the team leader. He reported back to the Firing Room:

ITL. Horace, this is Charlie on 245.

BL. Hello, this is Bob. I think he may still be in that HIM (Hardware Interface Module] meeting [on the fire detector problem]. What do you see out there?

ITL. Okay, starting on about the 235 foot level where the top hose is, the fire hose that was draining into the shower, the hose is not really, the drain is in the shower, the hose is not really draining into the little bowl on the shower and it was spilling over. So we have a lot of hard solid ice from the 235 feet down to 195 feet where I am now. Most of it's on the west side and the north side, and about halfway in-between, the floor is one solid sheet of ice about an inch and a half thick. And down on 195 foot level, the water's on the pipe and plumbing and structure and beams all the way over to the Orbiter Access Arm [OAA]. That's as far down as we got so far.

BL. Copy.

ITL. We have some icicles about 18 inches long.[91]

The Ice Team leader later reported that ice was covering part of the floor on the 195 foot platform where the crew would enter the Orbiter. Part of this discussion follows:

Significant participants in these conversations include:
DIRECTOR: Director of Engineering, Kennedy Space Center.
BL: Chief, Mechanical Systems Division, Shuttle Engineering Directorate.
LD: Launch Director, STS 51-L.
ITL: Ice Team Leader.

[89] OIS Channel 245, p. 570.

[90] Ibid., pp. 570–72.

[91] Ibid., p. 573.

BL. Charlie, is that ice going to be any kind of impediment to the crew?
ITL. Uh, not to the crew. I don't see any out here where the crew walks.
BL. How 'bout to the baskets [the emergency escape system]?
ITL. . . . Up to the what?
BL. If they had to go to the baskets for any reason, do they have a clear path through that ice?
ITL. Oh, no. When you get over that way, we got ice.
BL. So they had to get out in a hurry in order to get to the baskets, they've got to go over a sheet of ice.
ITL. Oh, yeah. On the north side, that's all one hard sheet of ice. Now, they could get to the two baskets on the south side, probably three baskets on the south side. Hold on just a minute, I'll go take a look. . . .[92]
ITL.Okay, Bob, I'm back.
BL. Okay. What's it look like over there?
ITL. Okay—some right at the elevator, right where the camera is, going back toward the baskets, we got ice on the floor. And the ice goes all the way across the west side of the facility, all the way over to the north corner on the floor. So it is slippery. Once you get past the west-most part of the FSS, the ice on the floor ceases, and you got a clear walkway, so all five baskets are, uh, six baskets do have a clear walkway right around the baskets. But to get between the elevator and the camera where you're looking at, there's some ice on the floor.
BL. Okay.
ITL. And including the handrails that they would be holding on to. But out here from the Orbiter Access Arm over to the camera is clear.
BL. What's your Safety guy there think about that? You got a Safety guy with you, don't you?
ITL. Yes, he's concerned. Matter of fact, there's some ice right under my feet now that I look.
BL. Charlie, Horace is back with us now. Why don't you start up your review from the 235 foot level on down again.
ITL. Okay. From the 235 foot level is where we had these litle Firex systems—the hose, the rubber hose which we ran over to the shower back on the northwest corner of the FSS, so we ran it on to the eyewash shower the level below, the 235 foot level, and it was running out of the drain, you know, the little basin—it was overflowing the little basin. So we have over in that area, down to the 195 foot level on the north side of the FSS, icicles that are about 18 inches or so long, about one inch in diameter or more at their maximum diameter. This floor, the grating, over on the north side paralleling the showerway and the elevator in some places are frozen solid about two inches thick. You know, the area is like 10 [by] 10 or more.
DIRECTOR. On the floor, which floors, Charlie?
ITL: The floor of the grating, on like the 215 and there's a level between 215 and right under, between 215 and 195; there's a half-level that you go out to the hatch on the north side.
DIRECTOR. Okay.
ITL. And there's a lot of icicles hanging, you know, under the floor. As far as the east side goes, on the 195 foot level where we are now, we have about one-quarter inch or one-half inch ice. On the beam structures themselves, they go all the way over to the—right at the hinge where the Orbiter Access Arm goes out. I don't see any on the Orbiter Access Arm itself, but there is some here where I'm standing and a little bit on the floor. Just before you came on, we were talking about the slidewire—the baskets. The floor from the Orbiter Access Arm over to, back where the camera is, is fairly clean. And from the camera back to the west side, is some ice on the floor to the west edge of the FSS.
DIRECTOR. Okay.
ITL. So the crew would have to walk across one slick spot. Around the baskets themselves, it's fairly clean. But to the northwest corner of the FSS where the baskets are, there's heavy concentration of ice on the floor.[93]

The discussion concluded:

DIRECTOR. Do you see anything out there that makes it unsafe for the crew?
ITL. At this time, I'd say from the elevator to the Orbiter Access Arm would be fairly good; the floor's in good shape. The elevator's got a little bit of—the doors are real hard to work but everything seems to work in that neighborhood. If they had to

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[92] Ibid., p. 574.
[93] Ibid, pp. 574-76.

go to the slidewire, it'd be very slippery from the camera that you're looking at to the slidewire itself. There's an area about ten feet long where the handrails have ice on them, as well as the floor.[94]

This discussion is of concern to the Committee. The slidewire referred to is the means by which the Shuttle crew would evacuate the launch pad if an emergency were to occur that required a rapid exit from the pad area. Crew safety concerns should dictate that the ice situation described by the Ice Team leader is unacceptable, and that some effort to remove the ice from the floor and handrails should be made. There is no indication that this was done. If the ice could not be removed, the mission should have been delayed until the danger represented by the ice could be eliminated.

This situation, admittedly, had nothing to do with the accident that destroyed STS-51-L. However, had this been the one time that use of the pad escape system was required, the crew would very likely have been impeded in their attempt to reach the escape baskets, and the lost time might have proven fatal. This system must be operated with the expectation that it will be used, and the countdown procedure should require that no barriers to its use be present before launch.

As the Ice Team continued its inspection, the discussion in the Firing Room involved the possible threat posed by the ice problem.

BL. We're just going to ask your opinion on the debris concern. If Charlie thinks we have a concern with debris, and I guess I would find it hard to believe that we'd be concerned about it from the FSS, but if we do have a concern, can we go out there and try to clean it up a little bit?

DIRECTOR. Yeah, I think we could. I think when he gets through here, if we think there's some areas that we need to clean up a little bit, we probably could.

BL. I think Safety would probably have to make a call, myself, on the floor if they think it's, that's a concern, but. . . .

DIRECTOR. Okay, he's [ITL] on 108 now.

BL. I see him on camera 108, next to the 155 foot level.

LD. Hey, what kind of debris are you guys talking about?

BL. The icicles on the FSS.

LD. Yeah, and how is that going to hurt you?

BL. Well, that's what I'm saying. I don't think it—personally, I don't think it would, but I just wanted to. . . .

DIRECTOR. [garbled] by there, you're not gonna hit the Orbiter, but Charlie's worried about it, Gene—the acoustics releasing it and it being free when the Orbiter comes by.

LD. Boy, he's really stretching it.

DIRECTOR. Oh no, I don't know whether that's stretching it too much or not.

LD. Well, I mean if we can ignore it, we need to feel comfortable about it.

P. All right, Gene, remember the wind is coming from the northwest.

LD. We need to all know if we don't get back into tanking as soon as possible, we could possibly blow it just for that.

DIRECTOR. Yeah, we understand, Gene. . . .[95]

The Ice Team leader's next report was no more encouraging. Water had spilled over the platform as the drains were unable to cope with the volume of water they were asked to manage. Icicles were found on the platform handrails that could easily be knocked off.[96]

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[94] Ibid., p. 576.
[95] Ibid., pp. 577-78.
[96] Ibid., p. 578.

As the team leader was explaining that the water could not be completely cut off for fear of making the situation worse.[97] the following conversations occurred:

LD: Hey, we gotta come out of there when you guys telling us you're pretty sure that water system's gonna work.

DIRECTOR. Yeah, you feel comfortable with what you see out there, Charlie, now?

ITL. We have a lot of ice, if that's what you mean. I don't feel comfortable with what's on the FSS.

DIRECTOR. Then what choices we got?

ITL. Well, I'd say that only choice you got today is not to go. We're just taking a chance of hitting the vehicle.

LD. You see that much ice?

ITL. Well, the problem we have is we hve a lot of icicles hanging, you know, even on the west side of the FSS here, which is only 60 feet or more from the Orbiter wing. And I'm sure that stuff is going to fall off as soon as the acoustics get to it. And you got a northwest wind, so you know, somebody will have to make that assessment. If we're worried about that little bit of ice that comes off the hydrogen vent arm, and the GOX [gaseous oxygen] vent arm, what we have over here is considerably more than that, you know—it's a hundred-fold.

DIRECTOR. You got enough ice that's over there that's big enough and got enough density to it that if it hits the Orbiter it could do some significant damage?

ITL. Yes, we do. . . . It's on the east side of the FSS. On the northeast corner of the FSS, which puts you about 65 feet or so from the vehicle. But it comes right to about where this camera is, it's right on the center thin line of the FSS, it comes that far over.

BL. Charlie, I would doubt the wind could blow that over. Are you concerned about during—after engine start, that things should kinda blow around?

ITL. Uh, yes. And the problem is it's so high, too. You know, it's way up to the top. If it were all the way down here to the bottom, it probably wouldn't be any problem.

BL. Can we go along the east side handrails and knock it off now? Isn't that the biggest concern—the east side?

ITL. Well, it's on the handrails and its on the floor underneath, too. You know, I guess it could probably be done but it'd be a job.

BL. It would take a long time, wouldn't it?

ITL. All the FACS pipes, and all the conduit and all the cable trays and then hanging down underneath the floor, you know, everywhere, on all the pieces of grating, you got little icicles hanging down.

DIRECTOR. Who do you have out there with you?

ITL. B.K., as far as my group goes.

DIRECTOR. Okay, why don't you guys go ahead and walk everything down and quick as you get—come on back, let's get with Gene and we'll sit down and talk about what we got.[98]

LD. Okay, why don't you do like Horace said, come on back and we'll go ahead and tank and we'll have you look at it when you go back.

ITL. Okay, and I think—you know, the Rockwell people have a program which says it probably would be all right, so contact them and let them put it in the machine and see what they get.[99]

As the Ice Team returned from its initial inspection, the launch director spoke to KSC's Director of Engineering about the ice situation:

LD. Yeah, we really need to do some head scratching on this ice thing and what we gonna do once we get back in. We've just about used all our hold.

DIRECTOR: Okay.

LD. What we're gonna do—we're not gonna opt to have a hold, we'll let the ice team go in just like normal, but we gonna keep counting the clock, and if we have to get ourselves into eating some of our launch window, we'll do it late.

DIRECTOR. Okay. We can do the ice inspection parallel with counting.

LD. Okay, but we need to have Rockwell in there where we need to ready to talk.

DIRECTOR. We can get Don in and we'll do that.

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[97] Ibid.

[98] Ibid. pp. 579–81.

[99] Ibid., pp. 581–82.

LD. I don't see how in the world—we need to worry about that ice. I don't know what's going to knock it off. It's going to freeze and stay there.

DIRECTOR: I don't think it will hit the Orbiter, Gene. It'll probably come off just from the acoustics, but. . . .

LD. Yeah, as you go by, I would think . . . I would think after we start the engines, if it starts breaking loose then, I don't know how it would travel that far until, you know, the six seconds or so we are on the ground.

DIRECTOR. Yeah. We'll get everybody here; we'll talking before—while we're tanking before they go out, then we'll be ready to help make an assessment.[100]

The Ice/Frost Team returned to the launch pad during the scheduled countdown hold at T-3 hours. The crew was somewhat larger than usual, since their primary objective was to clear ice from the water troughs on the launch pad. The team would also be making a follow-up assessment of ice on the FSS.

According to the post-accident report, "the team arrived at Pad B at 0654 hours and departed at 0844." A summary of their activity during this time stated:

> Ice in the troughs had thickened and was found to be solid. All secondary troughs except the northern most one in each hole now had ice. The two inboard primary hole troughs were also forming ice. . . . The "shrimp net" was employed to break up the ice and remove it. Approximately 95% of the ice was removed. The ice and unfrozen antifreeze solution was measured using an infra-red pyrometer and found to have a temperature between 8 and 10°F. . . . The pyrometer measured the MLP deck surface temperature as 12°F. On the FSS the quantity of ice had increased but the overall extent of icing was generally the same. In most cases, sheet ice was firmly adhered to the structure. Icicles could very easily be "snapped" off. Water continued to trickle down the facility—including the RSS [Rotating Service Structure].[101]

NASA launch team members were continuing their debate over the risk represented by the ice at this time.

LD. What do you think about the ice now?

DIRECTOR. Well, I don't know, Gene. I keep thinking there is an answer if we can find it, but we got people out talking it and I think we can make a decision on the SRB things [troughs]. I think we have the data there that we can make the right decision on that. The tower is what's going to be the one that is going to be hard to come to a decision on.

LD. But do we have any data that shows a mechanism for moving that ice across there?

DIRECTOR. That's what we're trying to see if we can come up with some kind of rationale why it won't. Charlie says we've got some data that we have moved some pieces across from basically the tower to the vehicle but we're—[Marty Ciofoletti, Vice President for Space Transportation System Integration, Rockwell International] and the guys are working with Downey to see how they feel if they come up with anything on the acoustics and things like that.

LD. Okay.[102]

KSC's Engineering Director then called the Rockwell liaison.[102a]

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[100] Ibid., pp. 583–84.
[101] "Ice/Frost Team Evaluation Report," p. 4.
[102] OIS Channel 245, p. 595.
[102a] Rockwell personnel appearing in House transcripts are identified as follows:
RI: Kennedy Space Center liaison.
RTI: Director of Technical Integration.

Continued

DIRECTOR. John, on 245.
RI. Go ahead, Horace.
DIRECTOR. You guys still talking to Downey?
RI. Yeah, we haven't go anything new since I last talked to you. Larry Williams just called in. He wants to talk to Bill Hovath [?] so they are getting ready to make a call to his now.
DIRECTOR. Okay.
RI. But we haven't got anybody. You know Jack McTimmons [McClymonds]—I talked to him and all he had to offer was his old program data of how far ice would go at various wind speeds, you know, from the FSS that's not our primary concern, obviously, right now.
DIRECTOR. No, that's right.[103]

In the Mission Support Room in Downey, California, Rockwell support personnel were expressing the same concerns as the Ice Team Leader, and were not confident their computer model could remove the uncertainty presented by the ice. Rockwell's Kennedy Space Center liaison was asked for information.

RTI. KSC, MSR [Mission Support Room]
RI. Go.
RTI. Good morning, John. Uh . . .
RI. It's been a busy morning!
RTI. I bet—looks bad, eh?
RI. Ice does look bad, yeah. The situation we've got right now is that they're working the bags in the SRB hole; they reported slush in those bags and we were watching on TV and some of that slush was pretty big and pretty heavy. But I think we can take care of that part—I think they're gonna get that cleared up. There's a crew out there working on those right now. One of the concerns [Richard] Colonna [Orbiter Project Manager, JSC] had was reflected pressure wave problems if there was a film of ice across those bags, but it looks like they're breaking that up. The big concern is gonna be the mass of ice that is on the FSS, from the 235 foot level all the way down to the MLP [mobile launcher platform]. Every platform had had water running on it all night and they're just a—some of the closeups of the stairwells looks like, uh, something out of Dr. Zhivago. There's sheets of icicles hanging everywhere. We've had reports, back on the northwest corner, of ice, icicles—this is a couple hours ago, the crew are up there walking it down right now, so we'll probably get some updates here shortly—but the initial walkdown said icicles up to two feet long by an inch in diameter. On the northwest corner, kind of graduating down to about three inches by one-quarter inch diameter on the east side, with periodic one-foot icicles on the east side on some of the cross beams.
RTI. Sounds grim.
RI. The big concern is that nobody knows what the hell is going to happen when that thing lights off and all that ice gets shook loose and come tumbling down and—what does it do then? Does it ricochet, does it get into some turbulent condition that throws it against the vehicle? Our general input to date has been basically that there's vehicle jeopardy that we've not prepared to sign up to. . . .
RTI. Okay. We didn't see this when we had icing conditions before?
RI. No, and they didn't run the showers all damn night before. They ran the showers this time and ran'em, pretty heavily by the look of it, the drains froze up and they all overflowed.
RTI. Oh. . . .
RI. And I guess nobody watched it all night or, if they did, they didn't say anything. But, uh—is John [Peller, Rockwell Vice President for Engineering] in yet?
RTI. No.
RI. Okay. We need to—you know, somebody at his level needs to get in and try to get up to speed as fast as they can. They're going to be looking for a final position from Rockwell here very shortly. We got—Bill Frohoff is right now talking to Larry Williams of JSC. I've got Colonna and Bobola sitting here with Al Martin and myself and we're probably going to be the forcing factor on this decision. Until

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RSD: Site Director for Launch Support Operations.
RTP: Thermal Protection Project Manager.
RDE: Vice-President for Engineering.
RVP: Vice President and Program Manager for Orbiter Operations Support.
RSR: Senior Representative, Mission Evaluation Room, Johnson Space Center.
[103] Ibid., pp. 596–97.

somebody can come up and tell us that the potential flow path is to the objects on the FSS at liftoff—you know, we're going to have to assume the worst case—but I don't think anybody is going to have that sort of data.

RTI. This is going to be a tough one.[104]

The Ice Team Leader reported that efforts to clear out the water troughs under the SRBs were meeting with success, and that the team was managing to clear the ice that had formed on the left SRB aft skirt.[105]

As this occurred, Rockwell's Site Director for Launch Support Operations was reporting from KSC to Downey. He said,

RSD. . . . [T]he situation here is that—very quickly—when Charlie gives his report, then they are gonna want to reconvene a top level management meeting here, so whatever we want to say in that meeting we're gonna have to come up with it here and now in order to be ready to say it and I guess the situation is that there are icicles all over the stand, that's the fixed service structure, all up and down it, various levels—some of the icicles are two feet long, an inch or two at the base, there are lots of small icicles hanging all over the place. What they say is that when they touch them gently that they break off and for that reason I don't think there is any doubt about the fact in my mind that when we start the SSMEs a lot of these icicles are going to break off and they're going to—and when they do break off, then what's going to happen is that they're gonna come tumbling down, they can ricochet off of the service structure and they can—then some of them wind up on top of the MLP.[106]

The discussion was interrupted at this point by another report from the ice team, indicating that the lower levels of the FSS had ice coverage equal to the levels already discussed. The decision was made to bring the team leader back for a report to managers, in order to decide whether the threat was sufficient to stop the launch.[107]

The discussion at Rockwell then resumed:

RSD: Okay. He was just reporting on one of the levels. As he, as Charlie Stevenson, of NASA, moves up and down with the ice team, they're reporting on each level and on that particular level he was reporting a signficant amount of ice as the result of the overflow from the shower. You know, they left the water running in order to keep the pipes from freezing and then, I guess, some of the drains have frozen so then the water's overflowing and that's what's creating a lot of the big icicles. But at any rate, what I was getting around to, it just appears to me that when these icicles break off when they start the SSMEs some of them are very likely—in fact, I'll tell you, almost for sure—are gonna wind up on top of the MLP and then when we launch it seems to me it would be very difficult for anybody to predict where that debris would go and it appears to me that there would be a possibility of some of that debris impacting the Orbiter tiles and I don't know how our aerodynamists or analysts or anybody you know could really say that that wouldn't happen. They can predict what happens when you drop a piece of ice in the wind. They can also predict what happens due to aspiration when you start the Solid Rocket Motors and SSMEs. The real question is how do you predict what happens to ice chunks that are on top of the MLP at launch and where they go. So, at any rate, that's how we see it here at launch.

RTI. Well, one thing I guess we can see, from the view that we have, is ice on top of the MLP right now.

RSD. Yes, there is ice on top of the MLP right now.

RTI. That's unacceptable. Anything in the trough is unaçcepatable and any ice that would impact the vehicle during ascent is unacceptable and we can't predict what's going to happen to all that massive ice on the towers, so I think we're in a critical situation. . . .

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[104] OIS Channel 216, pp. 340–41.
[105] OIS Channel 245, pp. 600–601.
[106] OIS Channel 216, p. 345.
[107] OIS Channel 245, pp. 603–604.

RTP. Most of the ice on the tower is going to end up on an MLP, probably right before SRB ignition anyway—right, Al? You think?

RTI. Well, it's going to end up looking like snow, though, isn't it?

RTP. No, this is hard ice.

RTI. Once it hits that tower it's not going to be hard ice. What we're worried about is the aspiration effects of the motion of the ice into the vehicle.

RTP. You're still going to have large chunks of ice, ice cubes. Like an ice cube.

RTI. That's unacc . . . question is, how high is, is the highest elevation of ice, what was the. . . .

RTP. I think they're saying it's all the way to the top of the tower, like the 235 foot level has icicles forming and all the way down from there.

RTI. Okay.

RSD. Bob, would you say again what you were saying about the ice on the tower and the concern about that?

RTI. We really don't have a data base to know what's going to happen to the ice. We do have some information that we can get horizontal movement of the ice into the vehicle. Obviously, since it's very tenuous, it's going to be bouncing all over the place. It'll be bouncing off the J-boxes and everything else. So you're going to have some horizontal velocity of ice.

RTP. Hey, Bob, you're breaking up again.

RTI. Okay. Let me try once more. Our data base does not allow us to scientifically tell you what's going to happen to the ice. Therefore, we feel we're in a no-go situation right now.

RSD. Okay. That, Bob, is a consensus down here, too—that there's no way, that is for Rockwell, the consensus down here for Bill Frohoff and ourselves, that there's no way to predict what's going to happen, and I think that when we get into this next meeting, we need to state that as Rockwell's position and I think that's going to come up fairly soon, here. Now, I have told Dick Colonna that I suspect that's going to be the Rockwell position, I haven't told them officially. I've also told Horace that, but I haven't told him that officially, and I guess, or, uh, do you think we're ready now to, uh, for Rockwell to state that position and do you want to go back to the MER with that or how do you want to handle it?

RTI. Well, what I would like to do is get ahold of [Bob] Glaysher—we're not supposed to overrule him—and talk to him about it. Is he there?

RSD. We woke him up at 4:00 this morning. He called in about an hour ago. We understood he was on the way and he's not here yet.

RTI. Okay, I'd like to stonewall it until he gets here.[108]

At about this time, Mr. Bill Fleming, the senior representative of Rockwell International, reported from the Mission Evaluation Room at JSC that "ice on MLP, tower and trough not acceptable to MSR [Rockwell's Downey facility]."[109]

Just prior to attending the meeting called by Arnold Aldrich, the Vice President and Program Manager for Orbiter Operations Support at Rockwell held the following teleconference with their Chief Engineer at Downey.

RDE. Hey, we've gone over this again. Colonna called me and wanted to see if there is a way we could give it a go. But, when all the experts have looked at it, we still have concerns with three mechanisms. One, direct transport of falling ice into the vehicle at SSME ignition and the wind is adequate to make that happen. The ten-knot wind can move it laterally like twenty feet and a fifteen-knot gust could take it laterally forty feet. So even though you might be able to placard it, it's very close with the wind you've got. Secondly, you've got a rebound mechanism, where ice falls down into the lower part of the platform and goes out. Some pretty sizeable chunks and sometimes all it does is break an icicle in two, that's clearly enough to cause significant tile damage. And, finally, the ice ends up on the MLP and in the trough is all potential debris sources at SRB ignition and liftoff and the trajectory those things take are highly unpredictable and we just note in films tended to go in different directions. So we are not in the position to, uh. . . . So we've been through the three mechanisms, none of which we can completely clear. Dr. Petrone's here; we've discussed it with him. We still are of the position that it's still a bit of Rus-

[108] OIS Channel 216, pp. 345–48.

[109] NASA, Johnson Space Flight Center, "Presidential Commission Action Item (A-301) Response," DDATF-86-36, April 7, 1986, p. 1.

sian roulette; you'll probably make it. Five out of six times you do playing Russian roulette. But, there's a lot of debris. They could hit direct, they could be kicked up later by the SRBs, and we just don't know how to clear that.

RVP. Okay. Our position fundamentally hasn't changed. We'll just go in now, we got a 9:00, we'll go in and express it. I'll let you know what happens.

RDE. And obviously, uh, you know, it's their vehicle and they can take the risk, but our position is as stated.

RVP. Okay, you got it.[110]

No recording exists of the meeting at 9:00 a.m. on January 28. In testimony before the Commission, Mr. Robert Glaysher stated that he told Mr. Aldrich that "Rockwell cannot assure that it is safe to fly."[111]

Mr. Al Martin testified that:

> I also added that we do not have the data base from which to draw any conclusions for this particular situation with the icicles on the tower, and also, we had no real analytical techniques to predict where the icicles might go at lift-off. The other thing that I did was review the fact that prior to each launch there is great care taken to assure that there is no debris out on the launch pad. A day or two before launch a crew goes out and they walk down the entire tower and walk down the mobile launcher surface, and also the concrete apron around the launch pad for the purpose of removing any debris such as nuts, bolts, rocks or anything else that might be there. . . . So I was drawing a corollary between the care that is normally taken for debris and painting a picture that the icicles appeared to me to be in that same category.[112]

Mr. Marty Cioffoletti testified that "I felt that by telling them we did not have a sufficient data base and could not analyze the trajectory of the ice, I felt he understood that Rockwell was not giving a positive indication that we were for the launch."[113]

Mr. Aldrich, conversely, told the Commission that:

> Glaysher's statement to me as best as I can reconstruct it to report it to you at this time was that, while he did not disagree with the analysis that JSC and KSC had reported, that they would not give an unqualified go for launch as ice on the launch complex was a condition which had not previously been experienced, and thus posed a small additional, though unquantifiable, risk.[114]

Aldrich concluded the meeting by deciding to recommend that the countdown continue until the ice team could return to the pad just prior to launch and make a final assessment. Aldrich testified that he told Jesse Moore about Rockwell's reservations, explained his decision, and recommended that the launch proceed unless the ice team discovered that the situation had badly deteriorated.[115]

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[110] OIS Channel 216, p. 353.
[111] Rogers Commission Report, Volume V, p. 1013.
[112] Ibid.
[113] Ibid., p. 1014.
[114] Ibid., p. 1025.
[115] Ibid.

The results of the meeting were reported back to Rockwell over the communications system:

RSR: MSR, this is MER.
UKN: Go ahead.
RSR: We just got a report in from Arnie and they're going to go ahead and go into the count. They're going to go out, sweep down the pad as best they can and remove as much ice as they can and go for the launch today.
UKN. We copied that.[116]

In their final report, the Ice Team found that ice on the MLP in direct sunlight had begun melting. They also found that icicles had begun to fall from the upper FSS levels. Ice cube sized pieces of these icicles were found within 10 feet of the left-hand SRB hole. The west MLP deck was swept clean of ice/icicles. The water troughs were checked and found to be forming ice, which was again removed using the "shrimp net."[117]

In analyzing launch films after the accident, NASA found that, contrary to expectations and analysis, ice from the Fixed Service Structure did reach and impact the Shuttle vehicle during liftoff. The report stated:

> Numerous launch films were viewed regarding FSS and RSS ice debris. A film (E-43) [Engineering Camera 43] looking directly in at the vehicle and FSS shows some ice falling straight down in the period between SSME ignition and vehicle ascent through approximately 20 feet. It shows that very many particles fell at approximately a 45° angle during the vehicle rise through 20 to 40 feet. This ice included sheet ice particles up to 6 in. x 6 in. and flowed down into the plumes at a point directly below the engine nozzles. Some of this struck the LH SRB. One downward looking camera (E-36) on the FSS clearly showed that a small amount of FSS ice debris reached the area of the LH SRB exhaust hole. Particles numbered 50-100 and were approximately ice-cube size. None of these or any other debris was observed to be ejected upward toward the Orbiter. Another film (E-18) looks upward from the SSME pit. This shows that after a vehicle rise of 10 ft. hundreds of ice particles flowed in below the main engine at a 45° angle. No Orbiter impacts are observed. Camera E-26 . . . reveals many small pieces of falling ice striking the LH2 TSM [liquid hydrogen tail service mast] in the period between SSME ignition and vehicle rise through approximately 25 feet. Due to aspiration, 50-100 small ice particles flowed into the LH SRB plume directly below the SRB nozzle as the vehicle rose through 4 to 25 ft. These films and others show fairly clearly that there was little or no debris damage to the oribiter [*sic*] during liftoff due to FSS/RSS icing for the conditions observed.[118]

In the summary and conclusions section of this report, the following statements appear:

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[116] OIS Channel 216, p. 355.
[117] "Ice/Frost Team Evaluation Report," p. 5.
[118] NASA, "Hazard to Orbiter Tiles Posed by the Vertical Structure Ice: Mission 51-L," January 30, 1986, p. 3.

> On STS-33 (51L) the actual FSS/RSS ice movement, as proven by the photographic documentation did not conform to the predictions in two important respects:
>
> 1. The ice generally did not release until after SSME ignition.
>
> 2. The ice translated several times farther toward the vehicle than predicted.
>
> To do meaningful predictions of ice movement, the effects of aspiration must be considered. Similarly, the release time of the ice must be known.
>
> Until the above capability is available, it should be assumed that FSS/RSS ice would be released early and pulled by aspiration into contact with the vehicle. FSS/RSS ice thereby could be judged as a potential high risk to flight safety.[119]

The Committee has proceeded at some length to develop the conversations regarding ice that occurred on the morning of 28 January because they illuminate tendencies that are at variance with the careful attention to safety the Nation has come to expect from NASA. It is the Committee's view that the information developed by the discussions between members of the ice team and those that took place between Rockwell personnel on this subject should have led to the conclusion that "FSS/RSS ice . . . could be judged a potential high risk to flight safety."

The Committee also notes that, in his presentation to the STS Program Manager at the 9:00 a.m. Mission Management Team meeting, the ice team leader apparently did not inform Mr. Aldrich that he had earlier recommended that the launch be held due to the ice in the pad area. There is no indication in testimony to the Commission that Mr. Aldrich knew of the team leader's comment, "Well, I'd say the only choice you got today is not to go." Had it been presented to him in those terms, the later reluctance of Rockwell to recommend a launch might have been sufficient to cause Mr. Aldrich to recommend a launch scrub. In any event, the uncertainly present in connection with this discussion should have been sufficient to cause a delay in the lauch until the ice melted off the gantry. The unknown risk represented by the ice would then have been removed.

These conversations also indicate that the launch director was not operating in a manner the Committee would expect. Given his position as the senior official responsible for the preparation of the Shuttle for launch, the Committee would expect a healthy skepticism to underlie discussions he had with members of the launch crew. In contrast to these expectations these tapes demonstrate that the director was often reminding the engineering team about time, and spent much time questioning the ice team leader's analysis of the ice on the pad. There is also no indication that he took steps to see that the pad escape system was ready for use by the flight crew, if necessary.

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[119] Ibid., p. 4.

Finally, Congressman Ron Packard discussed with witnesses at the Committee's hearings on July 25, 1986, ways to improve contractor participation in preparing the Shuttle for launch.

Mr. PACKARD. "Mr. Davis, you spoke regarding the companies having a voice in the decisionmaking, I presume, after the FRRs—that two week interim between launch and the readiness review system. Do you believe that the companies should have more voice, less voice, or have they had any voice in whether its a go or no-go?"

Mr. DAVIS.[120] "Well, I can tell you how it runs now. Up to and including the L-minus-one day review, there's no doubt that every company has a very strong voice; and, as a matter of fact, at the L-minus-one review, they are required to stand up and commit their hardware as go or no-go. And those are very unequivocal commitments, also. After that time, then the reviews are more mission management meetings that are held, and as you get down into the countdown, it turns into more of a real time polling of the people that are actually controlling the launch. In those latter meetings, we are not, I would say, formally involved in those unless there is some problem with the hardware itself, the External Tank hardware. We are in Firing Room 2 in a very significant presence; we are aware of what is happening in some of the consoles. We sit behind them; we do not operate them. We are polled by the Director of Engineering prior to the launch actually proceeding, so we are sort of polled in an informal manner. We are not asked at any time after the L-minus-one for a formal go or no-go. I believe it would probably be appropriate, in terms of the Commission's desires, that indeed we be more formally involved in the mission management meetings, and that at some appropriate late time in the launch count—and I would leave that to NASA to decide—that indeed the companies be asked to declare go or no-go."

Mr. PACKARD. "A quick answer, Mr. Murphy. Do you agree?"

Mr. MURPHY.[121] "Yes, I agree with what Rick has said. I think that we have found out that we commit ourselves, I guess, at 20 minutes and 9 minutes by the people who are manning the consoles, but it does not rise to the management level which it should, in accordance with what Mr. Davis has stated. We would like that opportunity also."

Mr. DAVIS. "I'd like to make one other comment on that. I have never felt that if I needed to stop a launch, I could not stop it. While I have not been asked for a positive go or no-go, the ability is always there if I decide no, to stop the launch."

Mr. PACKARD. "Mr. Jeffs, do you feel the same?"

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[120] Mr. Richard Davis, President, Martin Marietta Michoud Aerospace.

[121] Mr. George Murphy, Executive Vice President and General Manager, United Technologies Booster Production Company.

Mr. JEFFS.[122] "Yes, I think the system should be formalized more. We have great visibility as to the problems and real times, being on the net and having CRTs [console displays] and people that are involved in depth, both at Downey and at Houston, who support it, even though it's at the Cape. But especially, when you have holds or delays and what have you, it needs to be—again—upgraded in real time with, I believe, the contractor's participation with NASA management right up to the launch decision point, and a little more formal process involved in the polling of the contractors."

Mr. PACKARD. "Mr. Murphy, if you'd had that system set up prior to the accident, would the flight—would it have still gone?"

Mr. MURPHY. "It would not have influenced our position at all. Our hardware—we had stipulations on what we required on the hardware during the whole period. They were met, and so we were in a 'go' posture as far as we were concerned. It would not have affected our position."[123]

The Committee believes that had the hardware contractors been required by NASA to formally declare their flight readiness, it would have removed the ambiguity in Rockwell's recommendations involving the ice on the Fixed Service Structure.

B. LAUNCH REDLINES

Section VI.B.1.b. of this report describes the rationale for development of certain criteria that serve to indicate when the Shuttle system is experiencing problems during the countdown. On the morning of January 27 and 28, during the countdown for STS 51-L, the launch crew in the Firing Room wrote waivers to certain of these criteria in order to permit launch of the Shuttle. Tapes and transcripts from the Operational Intercommunication System demonstrate that, at least in one instance, the technical analysis authorizing the use of a backup procedure did not account for ambient temperatures below the limits specified for this procedure. Thus, a waiver should not have been granted.

Revision C, Amendment 18 of the Launch Commit Criteria specified 45 degrees Fahrenheit as the minimum redline temperature for the External Tank nose cone.[124]

But the ambient temperatures during the countdown were well below that. On January 27, while the Shuttle waited for liftoff, conversations indicated that the nose cone heaters were not able to maintain proper temperatures. Excepts from the transcript of this discussion follow.

CF. Okay, go ahead, Fred.

FH. Okay, we may have a problem with propellant temperatures at that low level. We're about three degrees away from red line and losing ground right now.

CF. Because of the amount of heat that the ground system's able to put in there?

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[122] Mr. George Jeffs, President, North American Space Operations, Rockwell International.

[123] Cmte. Hgs., Transcript, July 25, 1986, pp. 71-74.

[124] NASA, "Launch Commit Criteria and Background," Revision C, Amendment 18, JSC-16007, December 1, 1982, p. 5.1-4. See Appendix VIII-K.

FH. That's right, they're giving us all they can right now. . . .[125]

DIRECTOR. . . . What do you guys feel about all the temperatures we saw today, like nose cone, and all those? Think about that so we can talk about that a little bit.

M. Yeah, that's another thing we're not happy about. We think we could probably still get by with it, but we're marginal.

UKN. You know, nose cone temp. Horace, is probably gonna be down in the low 20's, maybe even below 20.

M. I think the intertank we ought to be able to keep it up high enough. . . .[126]

DIRECTOR. . . . Okay. You guys think that nose cone heater is putting out all we gonna be able to get out of it?

UKN. Yep. You got it full blast. You're gonna be down 18-20° tomorrow on the nose cone. And the waiver we wrote today said we're only good down to 28. That was today's. . . .

UKN. And Horace, the [intertank] heater was running full bore for quite awhile. And we were running at least 10° below the set point temperature.

DIRECTOR. What about Fred Heinrich? He on 245?

FH. I'm here, Horace.

DIRECTOR. Fred, what do you think about the flow rate, I mean the RCS temps?

FH. Okay, Grady said they can crank that thing up locally and get outside the OMRSD [Operations Maintenance Requirements and Specifications document] limit, which may be enough, but we need to get started as soon as we get in there. Otherwise, we can't get the tank warm enough. We're going to lose ground all throughut the cryo load.

DIRECTOR. Okay.

FH. We're about 3° away from red line right now. We lost some during this cryo load with full bore on the heaters.

DIRECTOR. Okay.

R. Horace, this is Robinson, We'll have to check with JSC about the upper limit on this temperature.

DIRECTOR. What do you mean?

R. —right now OMRSD is not in—it may be something else other than Fred Heinrich's temperatures.

DIRECTOR. Okay. Then you don't have no problem picking it up, though, that's the only requirement we got, right?

R. That's affirmative.[127]

The waiver referred to in this conversation offered the following technical rationale: "No visible ice buildup on the nose cap fairing exit area. Temperature is 12 deg F below redline."[128]

The waiver also read, "For STS-33 [51-L] Min LCC acceptable is 28 deg F (was 45 deg F). Ullage transducers are acceptable down to 28 deg F (was 40 deg F). Refer Note A, LCC 5.1-4." [129]

Note A read:

> "The following purge temps are backup measurements.
> GLOT 4104A PRI Nose Cone Heated Purge Temp.
> GLOT 4604A SEC Nose Cone Heated Purge Temp." [130]

These refer to telemetry data channels. As the temperatures are received via telemetry, they are to be interpreted by means of a curve shown on page 5.1-4B.[131]

It is important to note that Note A applies only if the ambient temperature is between 40 and 99 degrees F. Otherwise, the redundant procedure is invalid.

As for the effect of exceeding this redline, the launch commit criteria reads:

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[125] OIS Channel 245, p. 217.
[126] Ibid., p. 218.
[127] OIS Channel 245, pp. 219-20.
[128] See Appendix VIII-L.
[129] Ibid.
[130] "Launch Commit Criteria and Background," p. 5.1-4.
[131] Ibid., p. 5.1-4B. See Appendix VIII-K.

"The minimum redline was established for two reasons.

A. L02 ullage pressure transducers calibrated to 40 deg F.

B. Avoid ice buildup at nose cone fairing exit."

"Consequences of exceeding redline;

1. Ice build up and possible impact to Orbiter.

2. Inaccurate ullage pressure readings." [132]

Engineering support communications continued with the following:

M. Horace, 245.

DIRECTOR. Yeah, go ahead.

M. Okay, one other thing's been brought to my attention. The, the LOX, the LOX ullage pressure transducers are calibrated to a minimum of 40 degrees and maximum of 140, which is what sets our minimum in the nose cone. Below that, we may get some variations in reading.

FH. No, that's not true, Mark.

M. Well, okay.

FH. (garbled) see I got some measure readings here from Mark was. . . .

M. That's what's on the LCC backup page.

UKN. Read the LCC backup page on the lower limit.

M. It says 40 degrees.

FH. It says for reasons of ice and frost at that the exit on the fairing.

M. Yeah.

MAC. Test on 245. (garbled) . . . 245.

DIRECTOR. Go ahead, Mac.

MAC. Hey, are you guys reading this LCC that the consequences of exceeding the nose cone temp redline? The sheet we have over here says that we will get inaccurate ullage pressure readings.

DIRECTOR. Okay, Mac, we understand, thank you.[133]

On the morning of January 28, the following discussions between engineering support personnel were also directed toward waiving the launch commit criteria on the External Tank nose cone.

UKN. Dave, they were, of course, expecting to violate those ET nose cone purge temps LCCs again. It's 20 degrees colder today than it was yesterday.

D. Well, I guess we'll be going down the line producing a waiver to the same effect that we produced yesterday.

UKN. How did we just define it yesterday for 51L? Or for this attempt for 51L?

D. Copy your whole question. Say again?

UNK. I thought we just annotated it yesterday as for 51L only.

D. That may be it, I'll check the waiver log. . . .

UNK. Dave? 161.

D. Go ahead.

UNK. Yeah, it is effectively 51L, I think we are in good shape.

D. OK.[134]

Later on the morning of the 28th, the following discussion occurred:

D. FR[Firing Room] 2, this is FR[Firing Room] 1.

UKN. Yeah Chris.

D. This is Dan. We need to send a waiver over for signatures. We're right now showing nose cone gas temps that we were discussing yesterday are down in the 12–16 degree range and the waiver that we wrote yesterday for 51L only gives us allowance down to 28 degrees F.

UKN. Yeah, we'll have to rewrite that waiver.

D. OK. You don't think we'll have any trouble getting that signed?

UKN. No, as long as our pressure transducers are OK.

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[132] Ibid., p. 5.1–4.

[133] OIS Channel 245, pp. 220–21.

[134] OIS Channel 161, p. 289. Channel 161 is identified as an Engineering channel used for troubleshooting and systems integration.

D. OK. What number would you like to use on that Horace? [135]

Shortly after this conversation concludes, the discussion on preparing the ET nose cone temperature waiver continued. Told "[W]e're right now sitting at 10 [degrees]" KSC's Director of Engineering said:

DIRECTOR. Let's hold on a minute and then we'll write the waiver, we'll probably wanna go below 10.

UKN. Yeah, we may just want to say no low and put a note on there that based on the pressure transducers.

Director. OK, so you want us to stand by and wait on that?

UNK. Yeah.

DIRECTOR. OK. We'll be waiting.[136]

If engineers intended to apply the same rationale for waivers as that used on Janury 27, the rationale is invalid. Ambient temperatures were well below the 40 degree limit necessary for a valid backup procedure. Therefore, the backup procedure should not have been employed.

Later, in a discussion between the Launch Director and the Director of Engineering regarding countdown problems, the following discussion about the temperature waiver took place.

LD. OK. I understand we're in the process of writing a new waiver with a lower limit of 10?

DIRECTOR. We're still looking. We'll give you a low limit.

LD. What are they running today?

DIRECTOR. It's been down as low as 10 basically.

LD. Wow! [137]

It was in a subsequent discussion between the same principals that the new limit was established. Their conversation, however, does not reflect that the limit was chosen by rigorous technical analysis.

LD. We have nothing else, Horace, not unless you guys are working something.

DIRECTOR. No, just the ice.

LD. OK. The only outstanding item we have right now is the one waiver on the cone temps.

DIRECTOR. OK. It looks like we probably could say about 10° and be OK on that one.

LD. OK. We'll use 10° then.

DIRECTOR. OK.[138]

Completing preparations of the waiver, the Director had the following conversation with one of the technicians.

DIRECTOR. OK, Jackie, in writing this waiver for nose cone temps we want to put the words on here saying that we can use the ullage transducer as an alternative way of determine redline, but we believe that yesterday Warren Wiley was saying that they had looked at those and they're good down to approximately 11 degrees and we wanted to verify that.

J. I think they did to 10 degrees.

DIRECTOR. 10?

J. Yeah.

DIRECTOR. The ullage transducers.

J. OK. Thank you, Horace.[139]

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[135] Ibid., p. 296.
[136] Ibid., p. 297.
[137] Ibid., p. 299.
[138] Ibid., p. 302.
[139] Ibid., p. 305.

There is no documentation to describe the method by which the ullage pressure transducers were qualified to "10 degrees." No alternative analysis is described on the launch commit criteria to support use of this low temperature as a rationale for approving a waiver.

This example indicates that NASA personnel do not necessarily employ a sufficiently rigorous engineering analysis to the waiver of launch commit criteria during countdown. There also appears to have been some confusion as to the effect of exceeding the redline temperature. Ullage pressure readings may be critical parameters if fed to the Main Engine controllers during flight. According to NASA:

> Following engine ignition at about T-4 seconds, the ullage pressure is supplemented using propellant gases vaporized in the engine heat exchangers and routed to the two ET propellant tanks. The tank pressure is maintained based on data inputs from ullage pressure sensors in each tank to control valves in the Orbiter. A combination of ullage and propellant pressure provides the necessary net positive suction pressure to start the engines. The net positive suction is the pressure needed at the main engine pump inlets to cause the pumps to work properly. The pumps, in turn, supply high-pressure liquid oxygen and liquid hydrogen to the thrust chamber. Acceleration pressure is added for operation. Fuel is forced to the engines primarily by tank pressures and, to a lesser degree, by gravity.[140]

Inaccurate readings from these sensors might cause the engines to operate improperly.

Also, according to the launch commit criteria, violations of launch redlines may also have occurred on the Auxiliary Power Unit (APU) gearbox lube oil (minimum redline temperature 42 deg F), and the fuel test lines (minimum temperature 41 deg F), if in fact the actual temperature were lower than minimum.[141]

There was also no mention of the 34.2 deg F minimum redline temperature for the SRB recovery batteries or what the temperature of the batteries was at launch.[142]

Under "Remarks," this criterion states, "Violation of this redline shall require an assessment to determine if a hazard exists which jeopardizes the Shuttle. . . ."[143]

Mr. Mulloy testified before the Commission that:

> Mr. MULLOY. "I had a discussion on my SRB loop with the SRB people dealing with the question of a 24-hour turnaround to attempt to launch again at 9:38 on the 28th and the effect that the predicted cold temperatures for the night of the 27th might have on that.
>
> The input was received back both to Mr. Reinartz and myself that we were looking at the Launch Commit Crite-

[140] NASA, "Space Shuttle News Reference," 1981, p. 2–40.
[141] "Lauch Commit Criteria and Background," Amendment 20, p. G–23.
[142] Ibid., p. G–1
[143] Ibid.

64-420 O - 86 - 9

ria relative to temperatures. It was felt there was a need to look at the recovery battery temperatures that are in the forward skirt of the SRB and the fuel service module temperatures that are in the fuel service modules for the thrust vector control system in the aft skirt of the solid rocket booster.

The input received back by me was that they did not feel that would be of any concern. They were going to continue to look at it, and if any concern arose they would let me know.

I went to the 2:00 Mission Management Team and reported that there were no constraints to the solid rocket booster for a 24-hour turnaround, that we had taken a look at the recovery battery temperatures and the fuel service module. We did not feel at this time that there would be any Launch Commit Criteria for the low temperature limits that were established for those systems, but that we were continuing to assess that; should anything change in that regard, I would so report that."

Chairman ROGERS. "You referred to the Launch Commit Criteria. What were they as far as you knew in terms of weather conditions? Any?"

Mr. MULLOY. "In terms of weather conditions, yes, sir, I'm aware that there is a Launch Commit Criteria for the system for weather. There are a number of factors in that Launch Commit Criteria. One of them is the ambient temperature, which is established at 31 degrees.

Another is the sea state and winds in the SRB recovery area. Another is the cross-winds at the return to landing site runway at Kennedy Space Center. Another is the trans-Atlantic landing site weather, and another is severe weather, which is related to lightning and thunderstorms in the area."

Chairman ROGERS. "And when you say there were no constraints in the 2:00 meeting, does that mean that as far as you could see there were no problems in those areas?"

Mr. MULLOY. "No, sir, I did not evaluate those areas of the Launch Commit Criteria. What I was looking at was the specific Launch Commit Criteria items that are on the solid rocket booster and the effect that the low temperatures would have on that.

I would expect Mr. Aldrich would normally make the judgements on, and his people at the Johnson Space Center, would make the judgements on crosswinds and trans-Atlantic weather and the general ambient environment for launch." [144]

While there is no reason to believe that these waivers directly contributed to the cause of the accident, the low temperatures during the night of the 27th and morning of the 28th most probably did. Committee staff learned in discussions with Thiokol per-

[144] Rogers Commission Report, Volume V, pp. 827-28.

sonnel[145] that liquid hydrogen apparently remained in the External Tank throughout the night of January 27. This most likely played a role in the joint seal failure, since it permitted heat transfer through the ET/SRB aft attachment strut throughout the night. Of equal interest, however, is the fact that ET requires an eight-hour recovery period between tanking cycles, measured from the time the hydrogen tank low-level sensors are dry.[146]

If the hydrogen tank was never emptied during the turnaround procedure, this would represent a violation of those criteria. Had the criteria been observed, STS-51L would have required an afternoon window on January 28, or it might have been necessary to attempt the launch on January 29. This has not been independently confirmed, however.

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[145] Discussion with Carver Kennedy, Thiokol Wasatch Operations, Brigham City, Utah, September 4, 1986. General Kutyna also noted this in the Commission's hearing on February 14, 1986 (Rogers Commission Report, Volume IV, p. 660).

[146] OIS Channel 245, p. 218.

IX. DEFINITIONS OF TERMS AND ACRONYMS

AA-SF—Associate Administrator for Space Flight.
AFPRO—Air Force Plant Representative Office.
APU—Auxiliary Power Unit.
BOC—Base Operations Contractor.
BTU—British Thermal Unit.
CAR—Configuration Acceptance Review.
CDR—Commander.
CDR—Critical Design Review.
CIL—Critical Items List.
CoFR—Certification of Flight Readiness.
CPIF—Cost-plus, Incentive-fee.
CTS—Call-to-stations.
DAR—Deviation Approval Request.
DCAS—Defense Contract Administration Service.
DCR—Design Certification Review.
DFRF—Dryden Flight Research Facility.
DR—Discrepancy Report.
EG&G—Edgerton, Germeshausen and Grier.
ESMC—Eastern Space and Missile Center.
EST—Eastern Standard Time.
ET—External Tank.
FEAT—Flight Element Assignment Table.
FDO—Flight Dynamics Officer.
FMEA—Failure Modes and Effects Analyses.
FRR—Flight Readiness Review.
FSS—Fixed Service Structure.
GOX—Gaseous oxygen.
GSE—Ground Support Equipment.
HA—Hazard Analyses.
HDP—Holddown Posts.
ILL—Impact Limit Line.
IPR—Interim Problem Report.
IR—Infra-red.
IUS—Inertial Upper Stage.
JSC—Johnson Space Center.
KSC—Kennedy Space Center.
ksi—thousands of pounds per square inch.
LCC—Launch Control Center.
LFC—Left Forward Center.
L/H—Left Hand.
LOS—Loss of signal.
LOX—Liquid oxygen.
LPS—Launch Processing System.
LRU—Line Replaceable Unit.
MEOP—Maximum Expected Operating Pressure.
MER—Mission Evaluation Room.

MLP—Mobile Launch Platform.
MMT—Mission Management Team.
MRB—Material Review Board.
ms—millisecond.
MSFC—Marshall Space Flight Center.
MST—Mountain Standard Time.
NASA—National Aeronautics and Space Administration.
NRC—National Research Council.
NRP—National Resource Protection.
NSTS—National Space Transportation System.
NTD—NASA Test Director.
OASCB—Orbiter Avionics Software Control Board.
OIS—Operational Intercom System.
OM—Operations Manual.
OMI—Operations Maintenance Instruction.
OMP—Operations and Maintenance Plan.
OMRSD—Operations Maintenance Requirements Specification Document.
OMS—Orbiter Maneuvering System.
OPF—Orbiter Processing Facility.
PAC—Problem Assessment Center.
PAS—Problem Assessment System.
PDR—Preliminary Design Review.
PGHM—Payload Ground Handling Mechanism.
PR—Problem Report.
PRCBD—Program Requirement Change Board Directive.
psig—pounds per square inch gage.
PSP—Processing Support Plan.
QC—Quality Control.
RCS—Reaction Control System.
R.F.—Radio Frequency.
RPSF—Rotation, Processing and Surge Facility.
RSO—Range Safety Officer.
RSS—Range Safety System.
RSS—Rotating Service Structure.
RTLS—Return to Launch Site.
SCA—Shuttle Carrier Aircraft.
SPC—Shuttle Processing Contractor.
SRB—Solid Rocket Booster.
SRM—Solid Rocket Motor.
SR&QA—Safety, Reliability and Quality Assurance.
SSME—Space Shuttle Main Engine.
STS—Space Transportation System.
TBD—to be determined.
TDRS—Tracking and Data Relay Satellite.
TM—Telemetry.
TSR—Technical Status Review.
TVC—Thrust Vector Control.
VAB—Vehicle Assembly Building.
VPF—Vertical Processing Facility.
WAD—Work Authorization Document.

X. APPENDICES

V-A

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
WASHINGTON, D.C. 20546

REPLY TO ATTN OF: HO-1

MEMORANDUM

TO: Marshall Space Flight Center
Attn: Mr. Garland G. Buckner
~~Acting~~ Procurement Officer

THRU: Marshall Space Flight Center
Mr. Roy E. Godfrey
SEB Chairman

FROM: HO-1/Chief, Operations and
Review Division

SUBJECT: Selection of Contractor for Space Shuttle Program
Solid Rocket Motors

Subject statement, signed by the Administrator, is enclosed for inclusion in the official contract file. Also enclosed is a copy of the Administrator's Statement for the SEB Chairman.

William E. Stuckmeyer

Enclosure

Copy to Roy Godfrey

* * * * * * *

THIOKOL CHEMICAL CORPORATION

Thiokol presented an approach to the SRM Program which clearly focused on maximum utilization of existing facilities and low early year funding. In-house production effort would be accomplished in the Wasatch Division, Utah facility. Increment III production would be accomplished by acquisition of portion of the adjacent Air Force Plant 78 as Air Force requirements phased out. AP requirements would be met by increasing the capability of existing facilities in nearby Henderson, Nevada. Use of an existing, skilled, stable work force in a low labor rate area would minimize new hires and provide low labor costs. Thiokol's decision to fabricate nozzles in-house provided cost savings and good control over this extremely critical component; however, the Board concluded that this introduced some early risk because of lack of experience in fabricating nozzles of this size. Facility location resulted in high transportation cost of the SRM's; however, these costs were more than offset by low facility investments. The Thiokol proposal received the second highest overall Mission Suitability score by the SEB, being tied with UTC. The SEB ranked Thiokol fourth under the Design, Development and Verification Factor, second under the Manufacturing, Refurbishment and Product Support Factor and first under the Management Factor.

Design, Development and Verification

The Thiokol case design met the general SRM requirements; however, the cylindrical segment was close to the upper limits of size capability of the case fabricator. The nozzle design included ablative materials not currently developed or characterized. This offered potential savings in program cost, but with attendant technical and program risk. An expanded characterization and development program would be required. The thickness of the nozzle material was insufficient to meet required safety factors and thus degraded reliability. The amount of material required to correct the deficiency was substantial and the deficiency could require a redesign of the metal portions as well as the ablative portions. The design was complex and would contribute to difficulty in manufacturing. The Thiokol motor case joints utilized dual O-rings and test ports between seals, enabling a simple leak check without pressurizing the entire motor. This innovative design feature increased reliability and decreased operations at the launch site, indicating good attention to low cost DDT&E and production. The thickness of the internal insulation in the case aft dome was marginal and created a technical risk.

Thiokol provided comprehensive test plans and development verification objectives; however, they proposed to verify propellant burning characteristics by testing four to six full scale mixes which was excessive, and could be reduced by establishing correlations with smaller mix size data during DDT&E. Also, Thiokol proposed to hydroburst two motor case assembly specimens, whereas one test would be sufficient.

Manufacturing, Refurbishment and Product Support

Thiokol had extensive processing experience with their proposed propellant formulation, having processed over 150 million pounds of this general type of propellant. Thiokol's major weakness in this area of evaluation was in the area of case fabrication. The segment fabricator would be unable to fabricate the case segments strengthened with stiffening rings as proposed by Thiokol for alternate water entry load conditions, if required. This would probably require a case and grain redesign. Thiokol's manufacturing approach provided a good mechanized method of installing insulation, coupled with an innovative method of preparing the insulation surface for the liner by peeling off a dacron cloth from the inner surface of the insulation. A minor weakness in the manufacturing approach was the decision to fabricate nozzles in-house due to Thiokol's lack of experience in fabricating nozzles of this size.

Thiokol proposed to utilize existing facilities which, with minor modifications, were totally adequate for all three increments. The one exception to this was a failure to meet Quantity Distance safety requirements between casting pits for Increment III, however, there are ways to adequately cure this problem. Thiokol maximized the refurbishment of components and the potential cost savings provided by refurbishment. Another less significant strength was the enhancement of segment assembly provided by three alignment pins thereby reducing the assembly hours on the launch pad. Thiokol failed to provide enough new cases and nozzles to meet the launch schedule. Eight additional cases and nozzles would be required to provide assurance that launch dates could be met.

Management

Thiokol structured the development program so that all major costs were deferred to the latest practicable date. This resulted in low early year funding, which is a key program objective. The availability of an operating plant, with ample experienced personnel and a proven organization which could be phased to the SRM effort with minimum modification added considerable maturity and confidence and proved to be cost effective. The Board considered this to be a major strength for all three increments. A strong matrix management was evident and key line organization supervisors were experienced and had worked together as a team on many successful development and production programs such as Minuteman and Poseidon. Strong management participation and visibility in variance analysis was another strong feature as was the approach to corrective actions and their effect on estimate-to-complete. Procurement Management was thorough and well planned. SRM commodity purchases would be consolidated with that of other programs at Wasatch, which should result in lower cost. The Procurement of major items was well matched to overall SRM schedule requirements. Thiokol proposed a strong Configuration Management System which included thorough identification and traceability during DDT&E, production and refurbishment. The tentative decision to make the molded and tape wrapped nozzle in-house was considered a strength in this area. It would contribute to the low cost-per-flight goal by using available resources, avoiding subcontract fees, lowering overhead rates, and taking advantage of lower cost labor. The inherent risk management aspects also were considered.

In the area of Key Personnel, the proposed Program Director was considered exceptionally strong and had successfully performed as a Project Manager on other major programs. He is widely known for his excellent performance. The proposed Deputy Program Director would also be the Chief Project Engineer. He had important and successful engineering management roles in previous major motor programs and has an excellent reputation in the trade.

Although adequately qualified for their proposed assignments, the proposed Functional Managers and their Team Members in the Project Organization did not reflect the depth of experience available in the Functional Departments of the Thiokol matrix type organization and had not previously performed as a team. This was not considered a significant weakness by the Board because of the strong experienced matrix organization at Thiokol.

*   *   *   *   *   *   *

V-B

TWR- 14359

PROGRAM PLAN

PROTECTION OF SPACE SHUTTLE SRM PRIMARY MOTOR SEALS

4 MAY 1984

DR NO. 5-6

PREPARED FOR

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
GEORGE C. MARSHALL SPACE FLIGHT CENTER
MARSHALL SPACE FLIGHT CENTER, ALABAMA 35812

CONTRACT NAS8-30490

WBS 1.1.1.2.3

Thiokol / WASATCH DIVISION
A DIVISION OF THIOKOL CORPORATION
P O Box 524, Brigham City, Utah 84302 801/863-3511

FORM TC 4677 (REV 3-79)

PROGRAM PLAN

PROTECTION OF SPACE SHUTTLE SRM PRIMARY MOTOR SEALS

Prepared by:

Brian Russell

B. G. Russell, Manager
SRM Ignition System

Approved by:

Bob Ebeling

R. V. Ebeling, Manager
Igniter, Final Assy,
& Static Test

A. J. McDonald,
Director
Solid Rocket Motor

Released by:

J. G. Tibbitts

J. G. Tibbitts
Data Management

TWR-14359

Thiokol Corporation, a Subsidiary of
MORTON THIOKOL, INC.
Wasatch Division

PROGRAM PLAN

PROTECTION OF SPACE SHUTTLE SRM
PRIMARY MOTOR SEALS

1.0 INTRODUCTION

There have been incidents on SRM flight and static test motors where a primary o-ring has been slightly charred by hot gases which penetrated through the vacuum putty barrier. Motors affected thus far are STS-2A aft field joint, QM-4 nozzle joint, STS-11A forward field joint, STS-11B nozzle joint, and STS-13A nozzle joint. This program plan will result in defining the solution to this o-ring char.

2.0 OBJECTIVE

The program objectives are to systematically isolate the problem and to eliminate damage to SRM seals.

3.0 APPROACH

The program approach will consist of analysis, subscale (hot) tests, full-scale joint tests, and final verification in motor static testing.

The analysis will attempt to identify the cause of o-ring erosion, its acceptability, and justification. It will identify specific design or process changes which will eliminate further o-ring charring. Studies will be performed showing the effects of material variation characteristics, putty layup configurations, and fresh materials versus environmentally exposed putty. In conjunction with these analyses, a thorough study will be performed on material from alternate sources.

Testing will include laboratory material characterization, small motor hot tests simulating effects on cavity volume variations, flow patterns, exposure of o-ring and lubrication effects. The burn time of small hot motors will be in the range of 3 - 30 seconds depending on the results of previous small scale motor test results. Morton Thiokol also recommends that actual full-scale segment joint tests be used to evaluate pressurization effects on putty layup arrangements and flow changes due to final assembly of the joints. It is further suggested that a group of experienced people from MSFC and MTI be selected to witness the entire joint preparation, assembly, leak testing, and postfire teardown at KSC and MTI/Clearfield facilities. This team will also review all analyses, laboratory tests, subscale hot test, and support team reviews.

3.1 Analysis

The following tests shall be performed on the vacuum putty as a minimum. If further testing is required, it shall be performed and documented. Some of the tests are currently being performed on the existing putty, but are listed to assure that presently available data are summarized in the ensuing report. These tests will have to be repeated on any potential new putty. (See section 3.1.5)

3.1.1 Chemical Composition

An analysis will be performed on the putty to determine solids content, asbestos fiber (or other filler) content, chromate content, binder makeup, and all other applicable tests described in STW4-2847, Putty, Vacuum Seal.

3.1.2 Physical Properties

Tests shall be developed and conducted to determine adhesive strength of the putty (tackiness), strain capability, compressibility, and resistance to heat, erosion, and pressure shock (at SRM ignition).

3.1.2.1 Environmental Effects

The putty will be conditioned in controlled temperature and humidity environments, including ambient conditions at Utah and Florida, then tested for all physical properties and appropriate chemical properties such as non-volatile content and water solubility.

3.1.3 Aging

An aging program will be conducted on putty in Utah and Florida. The program should run for five years with particular emphasis during the first year at 3, 6, 9, and 12 months. Chemical and physical tests shall be performed at each stage of aging. The putty will also be checked for shrinkage and silicone migration from the paper backing used as a separator in the roll form.

3.1.4 Compatibility of Putty With Other Materials

The putty will be tested to determine effects of its mixing with Conoco HD-2 grease, cured NBR rubber, and both fresh and saltwater. If the materials react, properties of the resultant material will be established. Tests shall also determine whether the resultant material is corrosive to the D6AC case.

3.1.5 Second Source for Putty

A second (or third) source of putty is desirable to prevent further supply problems, which could seriously impact the Space Shuttle program. A development program will be implemented to test alternate putty candidates per the preceding requirements as a minimum. Subscale firing tests shall use the alternate putty to establish confidence to install the new putty in FWC-SRM static firing DM-6 for putty qualification.

There are three alternate source candidates at this time: Plastic Sealer 579.6 from Inmont Corporation of Georgetown, Ontario, with asbestos and Plastic Sealer 579.6 from Inmont Corporation, St. Louis, without asbestos. General Sealants is developing a high temperature putty that will also be screened.

3.1.6 Viton Characteristics

To aid the accuracy of the hot gas jet analysis, tests will be run to determine the erosion rate of Viton. These data, along with results from other tests described in this plan, will be used when the analysis is redone.

3.2 Subscale Firing Tests

3.2.1 Five Inch CP Motor

To verify that hot gas jets through the putty openings is correct, tests will be conducted which induce a gas jet impingement on an o-ring using five inch CP motors as test beds. Under tightly controlled conditions (environmental and mechanical), this data will be assessed to more fully understand what is happening in the SRM applications.

If a meaningful subscale joint test can be devised with putty in it, it will also be performed using the five inch CP as the hot gas source.

3.2.2 40 Lb. Char Motor

Depending on the results of the five inch CP hot testing, it may be desirable to include larger scale test motors having putty installed. Morton Thiokol is investigating the 40 lb. char motors. If required, such tests shall be performed to further verify the change in putty layup, type, and/or other filler materials.

3.3 Full-Scale Joint Tests

Tests shall be performed using a full-scale SRM field joint to verify the subscale results of the candidate putty layup configurations as affected by the actual joint assembly and leak test procedures. The "short stack" hardware is preferred for use instead of SRM case segments for ease of assembly and inspection. The following questions shall be answered as minimum requirements of this test sequence:

a. What is post assembly pressure in the cavity between the putty and the primary o-ring?

b. What is the minimum pressure required to blow through the putty?

c. What is the maximum acceptable leakage rate at 200 psig which meets the 50 psig leak test criterion?

d. How is o-ring seating affected by pressure and time?

e. What is the proper amount of HD-2 grease to be applied to the joint metal surfaces to minimize the free volume between the vacuum putty and the primary o-ring?

f. What is the effect of case eccentricity during segment mating on the flow of vacuum putty in the joint?

g. What dimension and weight controls are required to assure the vacuum putty layup is consistent and adequate?

Potential fixes will be investigated such as inducing paths through the vacuum putty at regular circumferential intervals to prevent localized o-ring damage caused by small, supersonic gas jets. The concept of a soft rubber barrier between the putty and primary o-ring will also be investigated. In addition, leak check procedures, particularly those employing the use of a flow meter, will be examined with acceptable and non-acceptable (leaking) o-rings. In all instances the behavior of the putty shall be closely monitored.

Results of the above described tests will be extrapolated to the nozzle to case joint and tests using a full-scale nozzle fixed housing and aft dome will be conducted, if necessary, to verify the adequacy of any change resulting from the field joint tests.

A plan, TWR-13983, has been prepared to check the putty configuration of the igniter to case joint. These tests will also be conducted and the results will be summarized in the final report.

3.4 Full-Scale Static Test

All potential design changes will be adequately tested on the subscale level and shall be incorporated into the SRM-FWC static firing DM-6 for qualification. A critical postfire inspection will be performed on the new configuration as well as the baselined portions of the DM-6 joints.

An analysis will be performed to assess the results of the FWC-SRM field joints as they compare to the HPM-SRM field joint. The field joints are shown in figures 1 and 2 for HPM-SRM and FWC-SRM, respectively.

3.5 Review and Witness Team

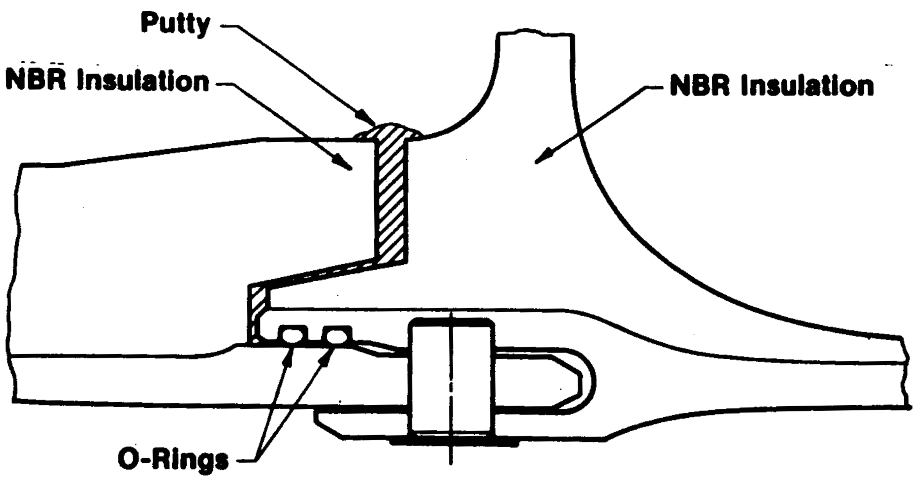
A review and witness team shall be established consisting of experienced engineers from Morton Thiokol and NASA to inspect and assess all test results. Anomalous conditions of joints from flown motors shall be critically inspected by members of this team. The team will determine the course of action to be taken as intermediate and final results become available.

4.0 SCHEDULE

The attached schedule reflects the time available to complete the testing and qualification of the preceding items.

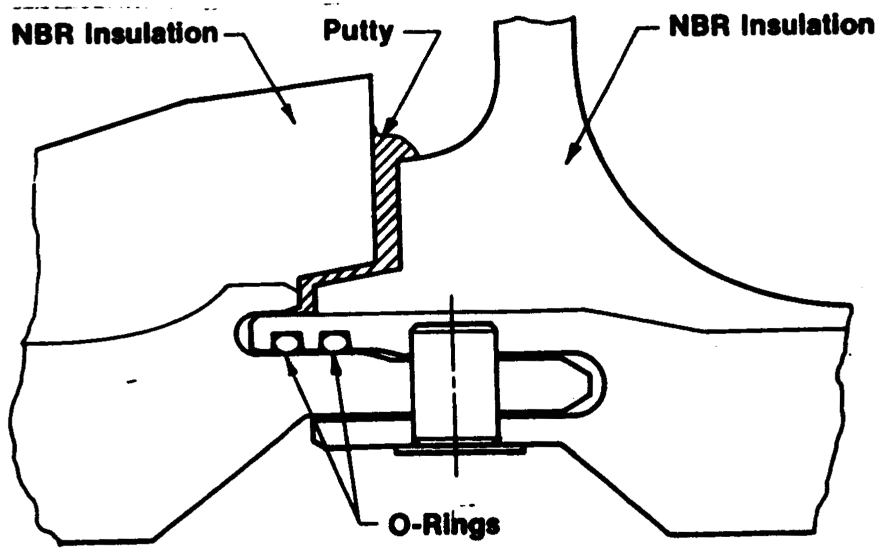
5.0 REPORT

A comprehensive report shall be prepared by Engineering documenting the tests and results. The original shall be released after the development testing, but prior to DM-6. A revision shall be released after DM-6 and a second and final revision shall be published after the aging program.



SRM-HPM Field Joint

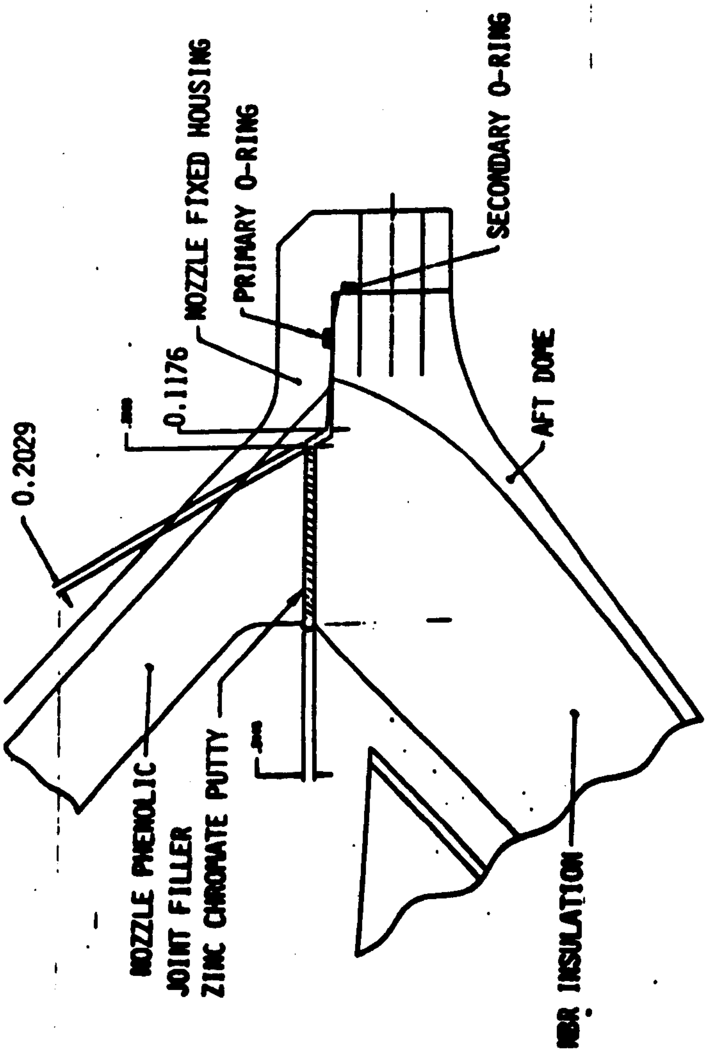
Figure 1



FWC-SRM Field Joint

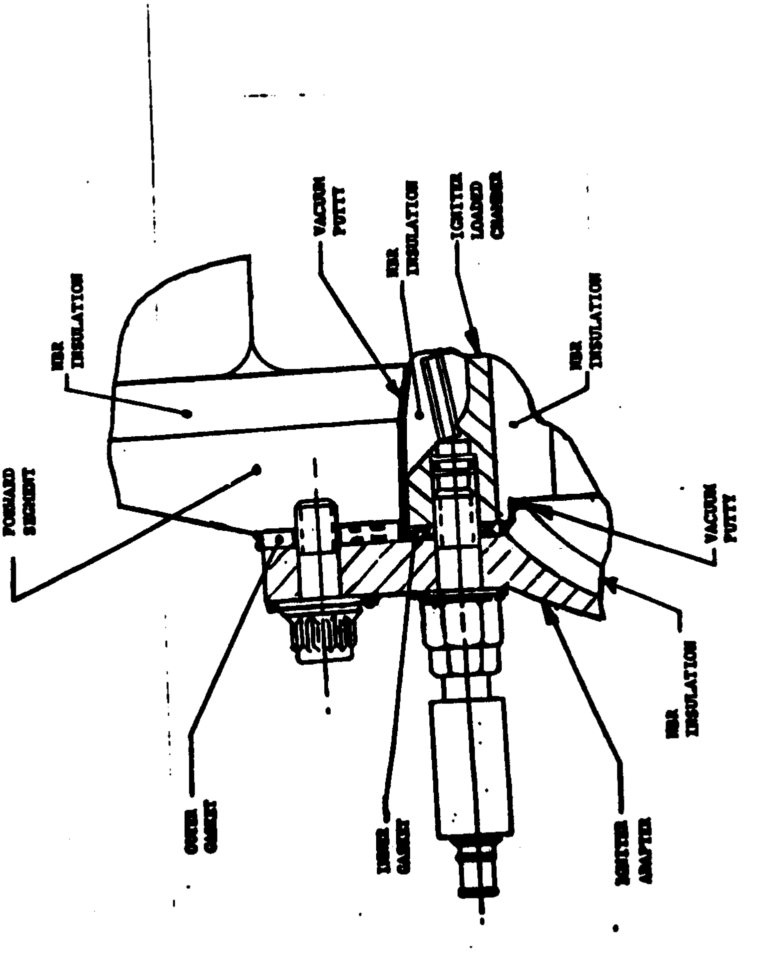
Figure 2

Thiokol Corporation, a Subsidiary of
MORTON THIOKOL INC.
Wasatch Division



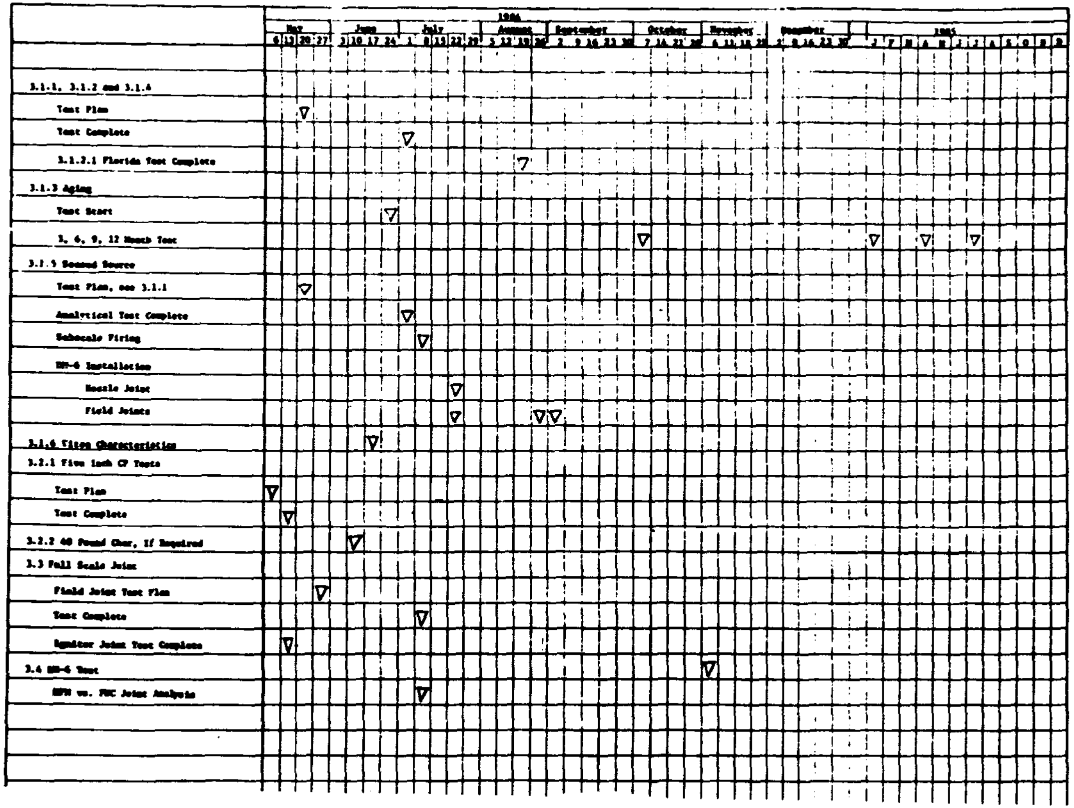
SRM NOZZLE-TO-CASE JOINT

PC 05-51-



SRM IGNITER-TO-CASE JOINT

PROGRAM SCHEDULE



Thiokol / WASATCH DIVISION
A DIVISION OF THIOKOL CORPORATION

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V-C

National Aeronautics and
Space Administration

George C. Marshall Space Flight Center
Marshall Space Flight Center Alabama
35812

NASA

EP25 (84-49) May 23, 1984

TO: EE11/Mr. Horton

FROM: EP25/Mr. Miller

SUBJECT: Evaluation of TWR-14359, "Program Plan, Protection of Space Shuttle SRM Primary Motor Seals"

The subject Program Plan has been evaluated as requested and the following comments are submitted for your consideration:

a. Page 3, Paragraph 3.1.2, **Physical Properties** - This Program Plan mentions compressibility testing of the zinc chromate putty, however no laboratory tests are proposed which will determine the extrusion characteristics (displacement in a free volume under compression load) of various candidates. This should be accomplished to provide a better understanding as to why various types of putty exhibit unlike extrusion patterns with identical layups.

b. Page 4, Paragraph 3.1.5, **Second Source for Putty** - A second source for zinc chromate putty is desired and needed, but due to the poor performance of the Randolph putty, a more immediate need exists for development of a replacement for the present Randolph putty. Recommendation to this effect was made in Memorandum EP25 (84-35).

c. Page 4, Paragraph 3.2.2, **40-Pound Char Motor** - The 40-pound char motor should be made a definite part of the Test Program. It is vital that the test article be capable of simulating the total joint configuration as close as possible, which includes zinc chromate putty. Provisions for the installation of the putty together with extended burn time and increased volumes capabilities are achievable with the larger motor and should be included in the total program.

d. Page 5, Paragraph 3.3, **Full-Scale Joint Tests**, Reference: first paragraph following "g". Please explain how potential fixes such as inducing paths through the putty at regular circumferential intervals and use of a soft rubber barrier between the putty and primary O-ring will be verified by hot firing prior to installation on DM-6.

e. Page 5, Paragraph 3.3, **Full-Scale Joint Tests**, Reference: Second paragraph following "g". The Test Plan should specify a hard requirement to verify all potential nozzle/aft dome joint changes on full-scale hardware. The case joint and nozzle joint configuration differences warrant separate full-scale nozzle/aft segment assembly tests.

PC 026425

f. **General** - Design changes to the insulation interfaces which will prevent degradation of the thermal barrier due to joint rounding under pressure should be investigated as a part of this effort. The present design of the case joint and nozzle interfaces where the zinc chromate putty is installed are oriented such that the joint gaps can vary from minimum to maximum dimensions around the circumference during assembly due to out of roundness and eccentricity. This condition which is present to some degree during every joint assembly operation, guarantees that some, or almost all of the zinc chromate putty in certain areas will be wiped off when the mating surfaces move parallel to each other during mating. This results in open insulation gaps with insufficient zinc chromate putty during motor operation because the joints tend to become round and concentric when the case is presssurized internally.

Questions concerning this memorandum should be referred to Mr. William L. Ray, 3-3809.

Ben Powers
for
John Q. Miller
Chief, Solid Motor Branch

cc:
EA01/Mr. Hardy
EE01/Dr. Littles
SA41/Mr. Mulloy
SA42/Mr. Wear
SA42/Mr. McIntosh
EE11/Mr. Coates
EE11/Mr. Jones
EP01/Mr. McCool
EP21/Mr. McCarty
EP25/Mr. Powers
EP25/Mr. Ray

PC 026426

V-D

National Aeronautics and
Space Administration

George C. Marshall Space Flight Center
Marshall Space Flight Center, Alabama
35812

NASA

Reply to Attn of: EP25 (83-119)

December 6, 1983

TO: EE51/Mr. Horton

FROM: EP25/Mr. Miller

SUBJECT: Request for Tests by the Contractor to Obtain Space Shuttle SRM Clevis Joint, Fixed Housing/Aft Segment Joint and Igniter Adapter/Forward Segment Joint Leak Check Data

It is requested that you take formal action to assure that the following tests are performed in a timely manner by the contractor, Morton/Thiokol, on SRM Hardware:

a. Case Clevis Joint Dual O-Ring Seal Leak Detection - Perform tests with full scale clevis joint hardware (short joints) to obtain the following data as a minimum:

(1) Post assembly pressure in the zinc chromate sealant cavity.

(2) Minimum and maximum volume of the zinc chromate sealant cavity, post assembly.

(3) Minimum pressure required to effect zinc chromate sealant blow through.

(4) Bleedback capability of the primary seal (from sealant cavity to cavity between the primary and secondary seals) at a variety of pressure values ranging from 10 psig up to a value which has been determined to effect sealant blowthrough. Various types of primary seal leakage conditions at predetermined leakage rates should be simulated.

(5) Determine maximum acceptable leakage rates at 200 psig which meets the 50 psig leak test criteria.

(6) Determine minimum pressure and time required to position O-rings for 50 psig leak check.

(7) Determine the volume of the cavity between the primary and secondary O-rings by analysis and flow test prior to and following the 200 psi O-ring positioning cycle.

(8) KSC and MTI GSE volumes should be simulated and the required temperature range should be duplicated as close as possible. Type II zinc chromate sealant should be used for all tests.

PC 026431

25

25th Anniversary
1958-1983

b. Nozzle Fixed Housing/Aft Segment Boss Joint - Perform tests with full scale hardware to accomplish the objectives in item a. above. The test designed to determine O-ring bleedback rate need not be repeated.

c. Igniter Adapter/Case Forward Segment Boss Joint - Perform tests with full scale hardware to accomplish the objectives in item a. above. Tests to determine pressure value required to position the seal is not required.

It is highly desirable to complete these tests prior to stacking of STS-12.

Questions concerning this memorandum should be referred to Mr. Leon Ray, 3-3809.

John Q. Miller
John Q. Miller
Chief, Solid Motor Branch

cc:
SA42/Mr. Wear
SA42/Mr. Denton
EE11/Mr. Coates
EP01/Mr. McCool
EP21/Mr. McCarty
EP25/Mr. Powers
EP25/Mr. Ray

V-E

National Aeronautics and
Space Administration

George C. Marshall Space Flight Center
Marshall Space Flight Center Alabama
35812

85-18

EP01 (85-48) March 7, 1985

TO: SA41/Mr. Mulloy
THRU: EE11/Mr. Horton

FROM: EP01/Mr. McCool

SUBJECT: Request for Initiation of Testing to Provide Data for Resolving the Burned O-Ring Seal Problem on the Space Shuttle SRM

Letter EP25 (83-119), from Mr. Miller (EP25) to Mr. Horton (EE11), subject: "Request for Tests by the Contractor to Obtain Space Shuttle SRM Clevis Joint, Field Housing/Aft Segment Joint and Ignition Adapter/Forward Segment Joint Leak Check Data" is referenced.

On December 6, 1983, this office requested via the referenced letter that the contractor obtain available full scale diameter, short stack hardware and conduct tests to provide data on zinc chromate putty behavior as related to affect on joint leak checks. Fourteen months have elapsed and no visible action has been taken to obtain and equip the short stack hardware although agreement was made to perform the test at the time of request. The only positive response by the contractor was the submittal of TWR-14359 on May 4, 1983, which contained a program plan followed by 5-inch CP motor tests, which were not designed to provide a solution to the burned O-ring problem. The acquisition of joint putty layup and leak check data on a high priority basis has become very important in view of the need to resolve the burned O-ring problems; accordingly, it is requested that you take the necessary action to direct that the following tasks be expeditiously performed by the contractor:

a. Subscale and full scale tests to determine effects of asbestos filled, cotton and talc filled, and non-filled zinc chromate putty on O-ring sealing integrity.

b. Full scale tests:

(1) Putty layup tests using current layup design.

(2) Putty layup tests using the attached figure 1 layup concept.

(3) Putty layup tests using the attached figure 2 layup concept.

PC 026413

(4) Repeat tests (1) and (2) except with vent slots located at 120-degree interval around the circumference as shown by attached figures 3 and 4. The slots are designed to prevent air entrapment and resulting volcanoes. Evaluation of layup effectiveness should be performed with flow meters to determine cavity volumes.

The above tasks are intended to complement TWR-14359 rather than replace the tests defined therein. We will be happy to assist the contractor in working out the details for the above proposals.

A.A. McCool
Director
Structures and Propulsion Laboratory

Enclosures:
As stated

cc:
SA42/Messrs. McIntosh/Denton
EE11/Mr. Coates
EE11/Mr. Jones
EP21/Mr. McCarty
EP25/Messrs. Miller/Powers/Ray
EE01/Dr. Littles

PC 026415

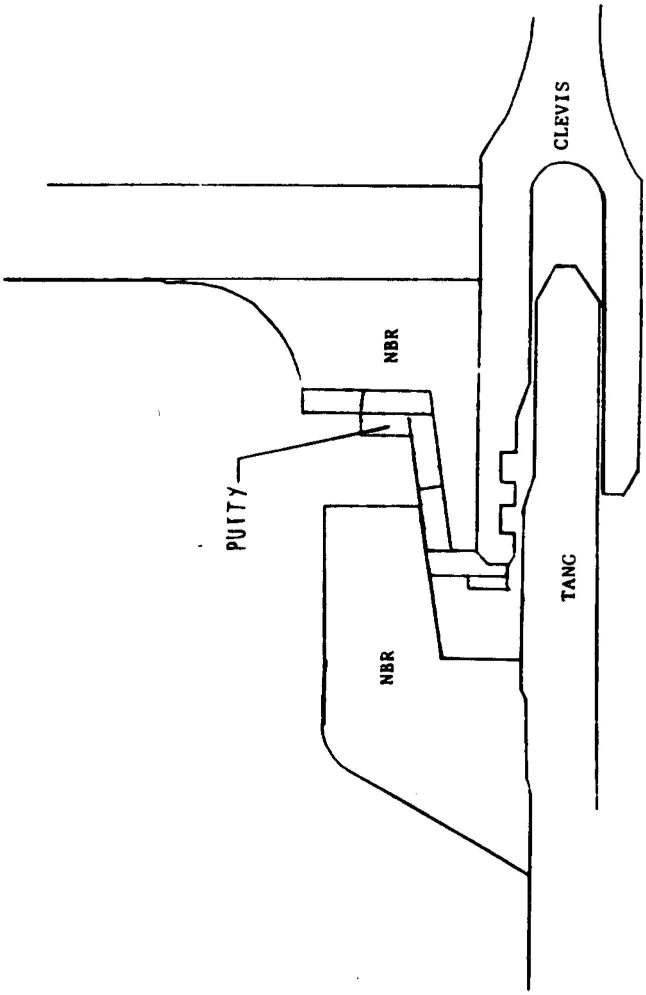


Figure 1

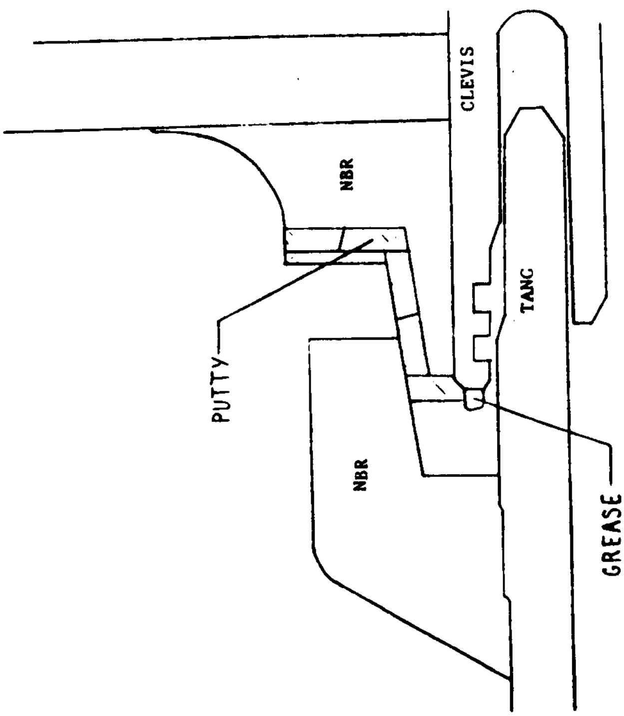


Figure 2

PC 026417

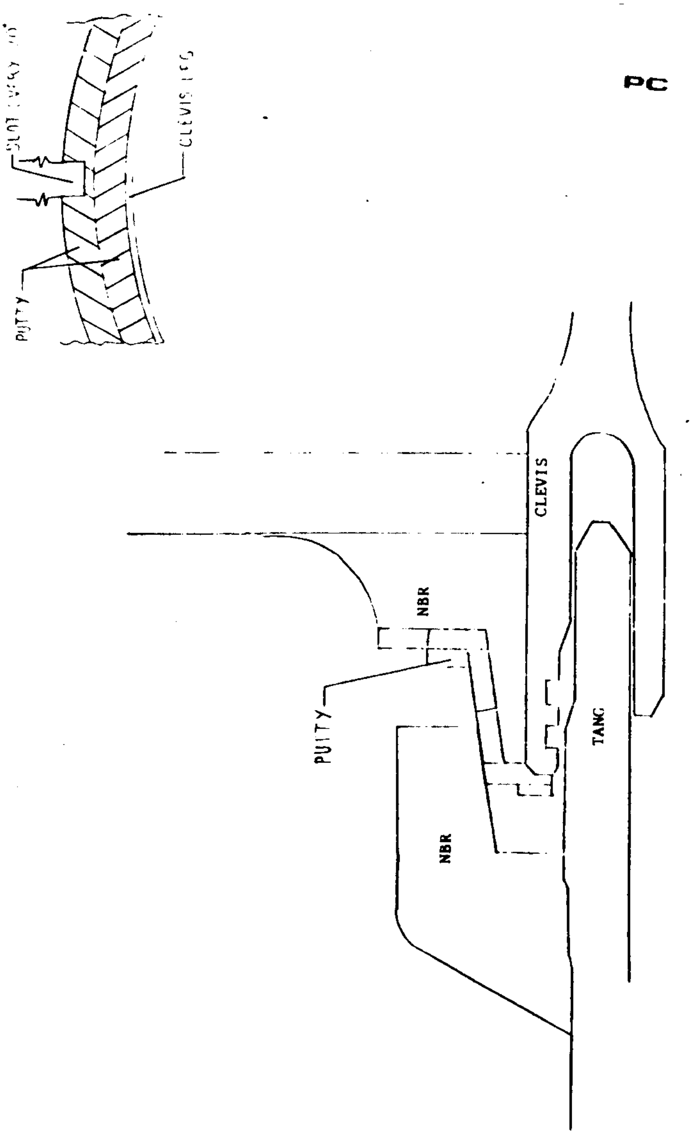


Figure 3

PC 026418

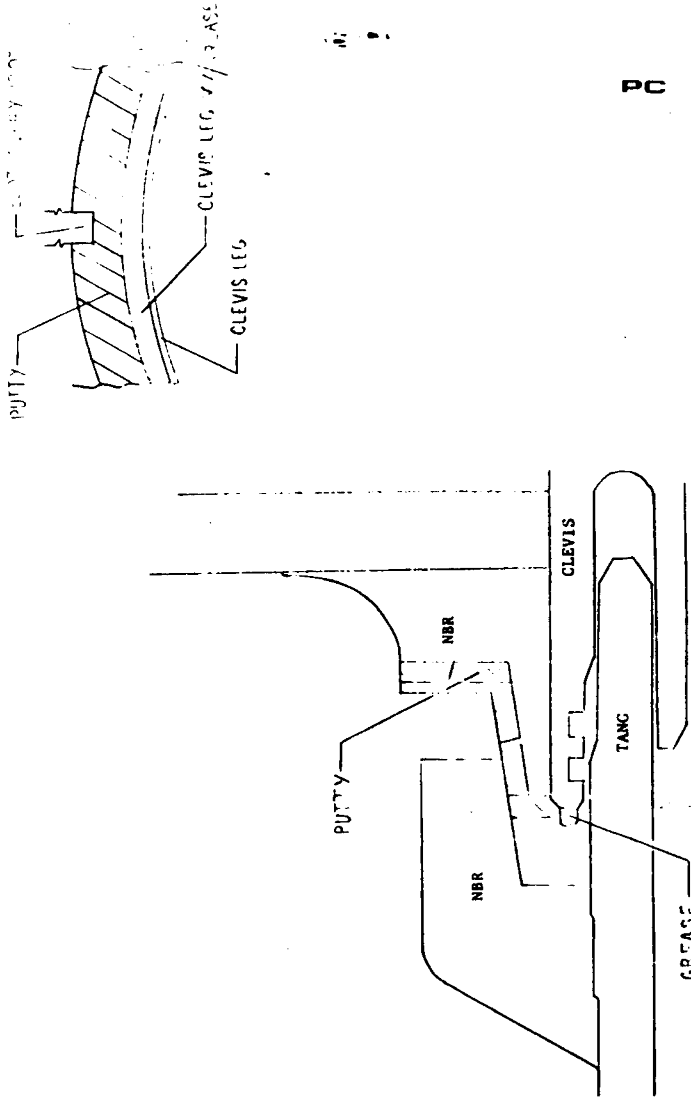


Figure 4

V-F

National Aeronautics and
Space Administration

George C. Marshall Space Flight Center
Marshall Space Flight Center Alabama
35812

NASA

Reply to Attn of EP25 (84-15)

February 22, 1984

TO: EE11/Mr. Horton

FROM: EP25/Mr. Miller

SUBJECT: Inspection of Fired SRM Pressure Joint During Disassembly

Please take the necessary action to reinstate detail post flight and post static firing inspection of specific pressure joints on the SRM which incorporate the thermal barrier and O-ring seal design concept. The inspection must be conducted at the time of disassembly to preclude destruction of data. The task should be performed by experienced, qualified engineering personnel and should be continued until the burned O-ring problem is understood and resolved.

The incidence of heat damaged O-rings on STS-2, QM-4 and on the recent flight of STS-11 warrants close surveillance of these areas to ensure that suspected anomalies are detected and properly recorded for assessment purposes. Recent discovery that the new type II zinc chromate sealant (thermal barrier material) would not adhere to the nozzle surface to which it was being applied, has opened up several unanswered questions, the most important being adhesion life of the sealant after installation on the SRM. Type II zinc chromate sealant was installed on all SRM's beginning with STS-8.

Areas of concern which warrant inspection are:

a. SRM case field joints.

b. SRM case nozzle boss to nozzle fixed housing joint,

c. SRM igniter to SRM case igniter boss.

d. Nozzle field splice joint.

for Bo Powers

John Q. Miller
Chief, Solid Motor Branch

PC 026428

cc:
EE01/Mr. Hardy
SA41/Mr. Mulloy
SA42/Mr. Wear
SA42/Mr. McIntosh
EP01/Mr. McCool
EP21/Mr. McCarty
EP25/Mr. Powers
EP25/Mr. Ray

PC 026429

2

V-G

22 July 1985

PROGRESS REPORT

APPLIED MECHANICS DEPARTMENT
Cost Center 287x

SPACE SHUTTLE

SRM O-Ring Erosion Problem

This problem has escalated so badly in the eyes of everyone, especially our customer, NASA, that NASA has gone to our competitors on a propriety basis and solicited their experiences on their joint configurations.

This whole week has been spent concepting ideas on how to eliminate the O-ring erosion problem. The new ground rule is to present every idea regardless of impact to cost weight schedule or whatever. Eleven hours of just group meetings has been spent discussing the problem and potential solutions. This does not include the many hours of informal meetings held with smaller groups.

One thing is increasingly obvious as time passes on this problem. If the company and/or Engineering does not assign specific people to this task (with no other work allowed - this being an absolute requirement) to secure a timely solution, then we stand in danger of having one of our competitors solve our problem via an unsolicited proposal. This thought is almost as horrifying as having a flight failure before a solution is implemented to prevent O-ring erosion.

Roger M Boisjoly
7/22/85

7 -723

V-H

MORTON THIOKOL INC.
Wasatch Division

Interoffice Memo

1 October 1985
E150/RVE-86-47

TO: A. J. McDonald, Director
Solid Rocket Motor Project

FROM: Manager, SRM Ignition System, Final Assembly, Special Projects and Ground Test

CC: B. McDougall, B. Russell, J. McCluskey, D. Cooper, J. Kilminster, B. Brinton, T. O'Grady, B. MaçBeth, J. Sutton, J. Elwell, I. Adams, F. Call, J. Lamere, P. Ross, D. Fullmer, E. Bailey, D. Smith, L. Bailey, B. Kuchek, Q. Eskelsen, P. Petty, J. McCall

SUBJECT: Weekly Activity Report
1 October 1985

EXECUTIVE SUMMARY

HELP! The seal task force is constantly being delayed by every possible means. People are quoting policy and systems without work-around. MSFC is correct in stating that we do not know how to run a development program.

GROUND TEST

1. The two (2) GTM center segments were received at T-24 last week. Optical measurements are being taken. Significant work has to be done to clean up the joints. It should be noted that when necessary SICBM takes priority.

2. The DM-6 test report less composite section was released last week.

ELECTRICAL

As a result of the latest engineering analysis of the V-1 case it appears that high stress risers to the case are created by the phenolic DFI housings and fairings. As it presently stands, these will probably have to be modified or removed and if removed will have to be replaced. This could have an impact on the launch schedule.

PC 102554

TC 2018 (REV 7-84)

A. J. McDonald, Director
1 October 1985
E150/RVE-86-47
Page 2

FINAL ASSEMBLY

One SRM 25 and two SRM 26 segments along with two SRM 24 exit cones were completed during this period. Only three segments are presently in work. Availability of igniter components, nozzles and systems tunnel tooling are the present constraining factors in the final assembly area.

IGNITION SYSTEM

1. Engineering is currently rewriting igniter gask-o-seal coating requirements to allow minor flaws and scratches. Bare metal areas will be coated with a thin film of HD-2 grease. Approval is expected within the week.

2. Safe and Arm Device component deliveries is beginning to cause concern. There are five S&A's at KSC on the shelf. Procurement, Program Office representatives visited Consolidated Controls to discuss accelerating scheduled deliveries. CCC has promised 10 A&M's and 30 B-B's no later than 31 October 1985.

O-RINGS AND PUTTY

1. The short stack finally went together after repeated attempts, but one of the o-rings was cut. Efforts to separate the joint were stopped because some do not think they will work. Engineering is designing tools to separate the pieces. The prints should be released tomorrow.

2. The inert segments are at T-24 and are undergoing inspection.

3. The hot flow test rig is in design, which is proving to be difficult. Engineering is planning release of these prints Wednesday or Thursday.

4. Various potential filler materials are on order such as carbon, graphite, quartz, and silica fiber braids; and different putties. They will all be tried in hot flow tests and full scale assembly tests.

5. The allegiance to the o-ring investigation task force is very limited to a group of engineers numbering 8-10. Our assigned people in manufacturing and quality have the desire, but are encumbered with other significant work. Others in manufacturing, quality, procurement who are not involved directly, but whose help we need, are generating plenty of resistance. We are creating more instructional paper than engineering data. We wish we could get action by verbal request but such is not the case. This is a red flag.

Bob

R. V. Ebeling

V-I

National Aeronautics and
Space Administration

George C. Marshall Space Flight Center
Marshall Space Flight Center, Alabama
35812

NASA

Reply to Attn of: EP25 (79-23)

February 6, 1979

TO: Distribution

FROM: EP25/Mr. Ray

SUBJECT: Visit to Precision Rubber Products Corporation and Parker Seal Company

The purpose of this memorandum is to document the results of a visit to Precision Rubber Products Corporation, Lebanon, TN, by Mr. Eudy, EE51 and Mr. Ray, EP25, on February 1, 1979 and also to inform you of the visit made to Parker Seal Company, Lexington, KY on February 2, 1979 by Mr. Ray. The purpose of the visits was to present the O-ring seal manufacturers with data concerning the large O-ring extrusion gaps being experienced on the Space Shuttle Solid Rocket Motor clevis joints and to seek opinions regarding potential risks involved.

The visit on February 1, 1979, to Precision Rubber Products Corporation by Mr. Eudy and Mr. Ray was very well received. Company officials, Mr. Howard Gillette, Vice President for Technical Direction, Mr. John Hoover, Vice President for Engineering, and Mr. Gene Hale, Design Engineer attended the meeting and were presented with the SRM clevis joint seal test data by Mr. Eudy and Mr. Ray. After considerable discussion, company representatives declined to make immediate recommendations because of the need for more time to study the data. They did; however, voice concern for the design, stating that the SRM O-ring extrusion gap was larger than that covered by their experience. They also stated that more tests should be performed with the present design. Mr. Hoover promised to contact MSFC for further discussions within a few days. Mr. Gillette provided Mr. Eudy and Mr. Ray with the names of two consultants who may be able to help. We are indebted to the Precision Rubber Products Corporation for the time and effort being expended by their people in support of this problem, especially since they have no connection with the project.

The visit to the Parker Seal Company on February 2, 1979, by Mr. Ray, EP25, was also well received; Parker Seal Company supplies the O-rings used in the SRM clevis joint design. Parker representatives, Mr. Bill Collins, Vice President for Sales, Mr. W. B. Green, Manager for Technical Services, Mr. J. W. Kosty, Chief Development Engineer for R&D, Mr. D. P. Thalman, Territory Manager and Mr. Dutch Haddock, Technical Services, met with Mr. Ray, EP25, and were provided with the identical

PC 026477

SRM clevis joint data as was presented to the Precision Rubber Products Company on February 1, 1979. Reaction to the data by Parker officials was essentially the same as that by Precision; the SRM O-ring extrusion gap is larger than they have previously experienced. They also expressed surprise that the seal had performed so well in the present application. Parker experts would make no official statements concerning reliability and potential risk factors associated with the present design; however, their first thought was that the O-ring was being asked to perform beyond its intended design and that a different type of seal should be considered. The need for additional testing of the present design was also discussed and it was agreed that tests which more closely simulate actual conditions should be done. Parker officials will study the data in more detail with other Company experts and contact MSFC for further discussions in approximately one week. Parker Seal has shown a serious interest in assisting MSFC with this problem and their efforts are very much appreciated.

William L. Ray

William L. Ray
Solid Motor Branch, EP25

Distribution:
SA41/Messrs. Hardy/Rice
EE51/Mr. Eudy
EP01/Mr. McCool

PC 026478

V-J

COMPANY PRIVATE

MORTON THIOKOL INC.

Wasatch Division

Interoffice Memo

31 July 1985
2870:FY86:073

TO: R. K. Lund
Vice President, Engineering

CC: B. C. Brinton, A. J. McDonald, L. H. Sayer, J. R. Kapp

FROM: R. M. Boisjoly
Applied Mechanics - Ext. 3525

SUBJECT: SRM O-Ring Erosion/Potential Failure Criticality

This letter is written to insure that management is fully aware of the seriousness of the current O-Ring erosion problem in the SRM joints from an engineering standpoint.

The mistakenly accepted position on the joint problem was to fly without fear of failure and to run a series of design evaluations which would ultimately lead to a solution or at least a significant reduction of the erosion problem. This position is now drastically changed as a result of the SRM 16A nozzle joint erosion which eroded a secondary O-Ring with the primary O-Ring never sealing.

If the same scenario should occur in a field joint (and it could), then it is a jump ball as to the success or failure of the joint because the secondary O-Ring cannot respond to the clevis opening rate and may not be capable of pressurization. The result would be a catastrophe of the highest order - loss of human life.

An unofficial team (a memo defining the team and its purpose was never published) with leader was formed on 19 July 1985 and was tasked with solving the problem for both the short and long term. This unofficial team is essentially nonexistent at this time. In my opinion, the team must be officially given the responsibility and the authority to execute the work that needs to be done on a non-interference basis (full time assignment until completed).

PC 102534

COMPANY PRIVATE

R. K. Lund - 2 - 31 July 1985

It is my honest and very real fear that if we do not take immediate action to dedicate a team to solve the problem, with the field joint having the number one priority, then we stand in jeopardy of losing a flight along with all the launch pad facilities.

Roger M Boisjoly

R. M. Boisjoly

Concurred by:

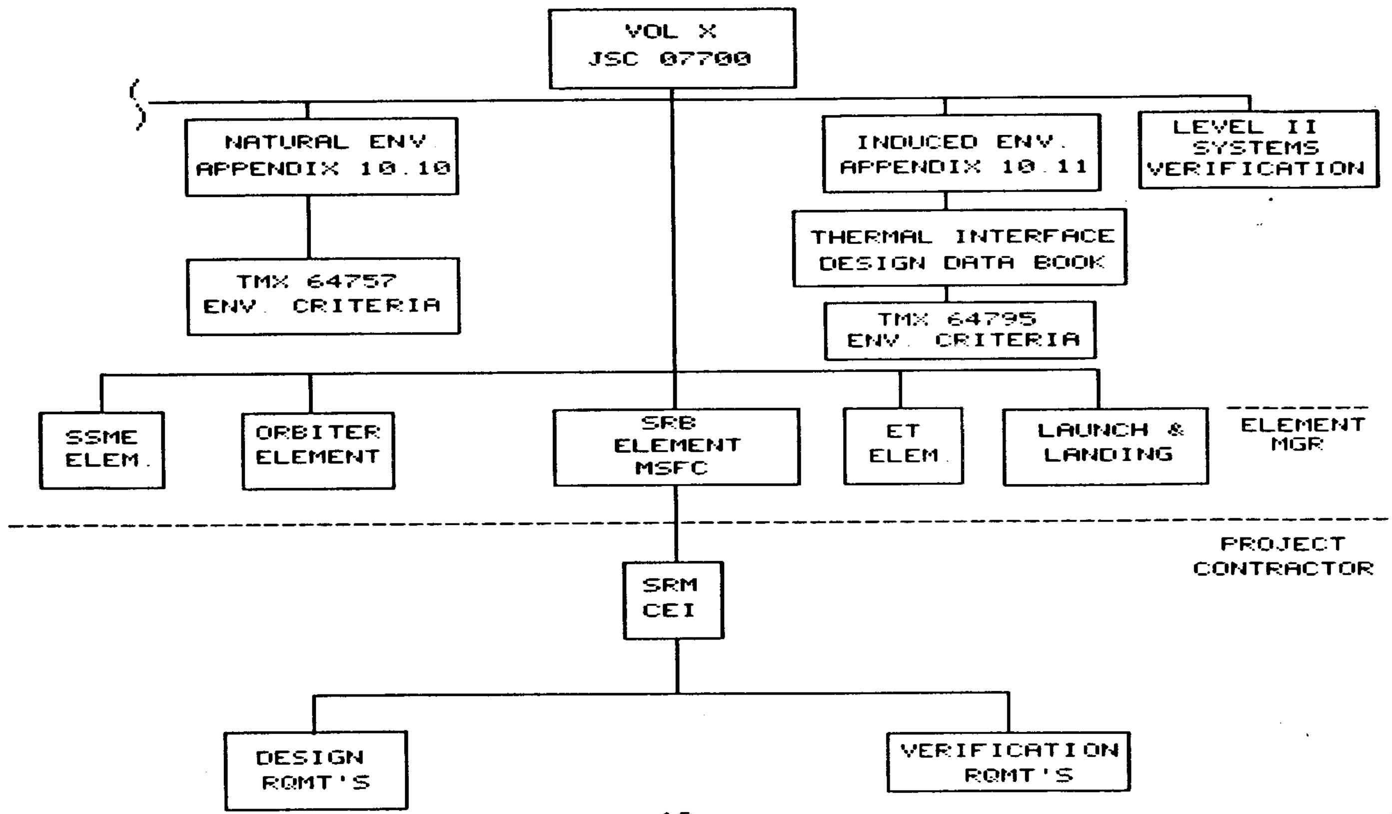
Jack R Kapp

J. R. Kapp, Manager
Applied Mechanics

NATURAL/INDUCED ENVIRONMENT REQUIREMENTS

R. H. KOHRS

MAY 19, 1986


VOL X
JSC 07700
NATURAL ENV
APPENDIX 10.10
TMX 64757
ENV. CRITERIA
INDUCED ENV.
APPENDIX 10.11
THERMAL INTERFACE
DESIGN DATA BOOK
TMX 64795
ENV. CRITERIA
LEVEL II
SYSTEMS
VERIFICATION
SSME
ELEM.
ORBITER
ELEMENT
SRB
ELEMENT
MSFC
ET
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LAUNCH &
LANDING
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PROJECT
CONTRACTOR
SRM
CEI
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RQMT'S
VERIFICATION
RQMT'S

VOLX.SKE

JSC 07700
VOLUME X
REVISION D

National Aeronautics and
Space Administration

Lyndon B. Johnson Space Center
Houston, Texas 77058

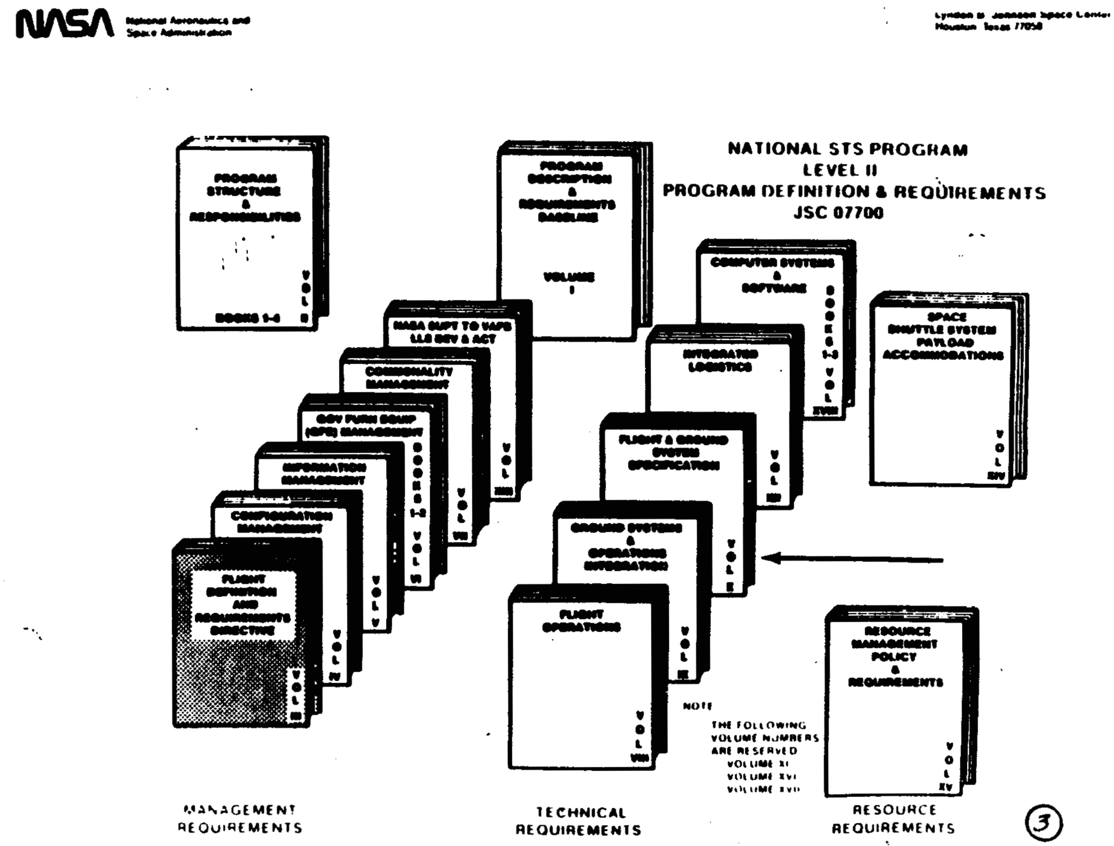
NATIONAL
SPACE TRANSPORTATION SYSTEMS
PROGRAM

SPACE SHUTTLE FLIGHT AND GROUND SYSTEM SPECIFICATION

LEVEL II PROGRAM DEFINITION
AND REQUIREMENTS

SEPTEMBER 30, 1983

2


NASA National Aeronautics and Space Administration
Lyndon B. Johnson Space Center Houston Texas 77058
NATIONAL STS PROGRAM
LEVEL II
PROGRAM DEFINITION & REQUIREMENTS
JSC 07700
PROGRAM STRUCTURE & RESPONSIBILITIES
BOOKS 1-4
PROGRAM DESCRIPTION & REQUIREMENTS BASELINE
VOLUME I
NASA SUPT TO VAFB LLS DEV & ACT
COMMONALITY MANAGEMENT
GOV FURN EQUIP (GFE) MANAGEMENT
INFORMATION MANAGEMENT
CONFIGURATION MANAGEMENT
FLIGHT DEFINITION AND REQUIREMENTS DIRECTIVE
COMPUTER SYSTEMS & SOFTWARE
INTEGRATED LOGISTICS
FLIGHT & GROUND SYSTEM SPECIFICATION
GROUND SYSTEMS & OPERATIONS INTEGRATION
FLIGHT OPERATIONS
SPACE SHUTTLE SYSTEM PAYLOAD ACCOMMODATIONS
RESOURCE MANAGEMENT POLICY & REQUIREMENTS
NOTE
THE FOLLOWING VOLUME NUMBERS ARE RESERVED
VOLUME XI
VOLUME XVI
VOLUME XVII
MANAGEMENT REQUIREMENTS
TECHNICAL REQUIREMENTS
RESOURCE REQUIREMENTS
3

shall not exceed 0.5 psig maximum at a boil-off rate of 2.9 lb/sec. The H_2 vent system shall not interface with the Orbiter but shall vent directly to atmosphere in flight. In addition to providing ET relief protection, the vent valves shall be capable of being actuated open, prior to launch, by ground command. The electrical command and pneumatic supply will be provided by GSE. Capability shall be provided to monitor the main propulsion LH_2 system pressure when vehicle or ground power is not applied to the flight instruments.

3.2.2.1.16 LO_2 Compatibility. Any material used internally in the liquid oxygen system of the Space Shuttle System main propulsion subsystems shall be compatible as determined by NHB 8060.1.

3.2.2.1.17 Design Environments.

3.2.2.1.17.1 Natural Environment. The Shuttle Flight Vehicle design shall satisfy the natural environment design requirements specified in Appendix 10.10.

3.2.2.1.17.2 Induced Environment. Each element of the Shuttle Flight Vehicle shall be capable of withstanding the induced environments imposed during transportation, ground operations, handling and flight operations as defined in Appendix 10.11. Each interface between elements shall be designed to withstand the induced environments defined in the applicable ICD.

3.2.2.1.17.2.1 Ascent Heating Design Criteria. In general, all elements of the Space Shuttle System shall be designed to withstand limiting induced ascent aerodynamic and plume heating environments, encompassing all baseline reference missions. The Orbiter vehicle for which limit ascent aerodynamic heating environments coupled with reuse criteria would result in unnecessary weight and cost penalties, shall be designed to meet reuse requirements considering the frequency of occurance of the ascent heating environments resulting from statistical treatment of the baseline reference missions and shall be shown to have single-mission survivability for limit ascent aerodynamic heating case encountered on any mission during the lifetime of the vehicle. The applicable environments are defined in Appendix 10.11.

3.2.2.1.17.3 TPS Absorption. All TPS material and installation design shall minimize absorption and entrapment of liquids or gases which would degrade thermal or physical performance or present a fire hazard (wicking), and shall not require draining, drying or any dedicated purge system from refurbishment through launch.

83 3.2.2.1.18 Flow Induced Vibration. All flexible hoses and bellows shall be designed to exclude or minimize flow induced

83 Refer to the Deviation/Waiver Page in front of the document.

4

8.0 METEOROID. The Space Shuttle shall be designed for at least a 0.95 probability of no penetration during the maximum total time for 500 missions in orbit, using the meteoroid model defined in Section 2.5.1 of TMX-64627.

8.1 METEOROID IMPACT. Space Shuttle meteoroid impact requirements shall be specified below:

a. Pressure Loss. The Space Shuttle manned volume shall be protected from meteoroid impact damage which would result in pressure loss when subjected to the meteoroid flux model as defined in TMX-64627.

b. Functional Capability. The Space Shuttle shall provide protection against loss of functional capability of selected critical items when subjected to the meteoroid flux model as defined in TMX-64627. The probability of no penetration shall be assessed on each item dependent upon function criticality.

9.0 ASTRODYNAMIC CONSTANTS. The values given in Sections 1.6 and 2.7 of TMX-64627 shall be used.

10.0 THERMAL.

10.1 GROUND THERMAL ENVIRONMENT. The ground thermal environment, including air temperature, solar radiation, and sky temperature limits, are specified in Table 10.1-1 and Figure 10-1. Also, see Sections 2.5 and 2.6 of TMX-64757.

10.1.1 Ambient Temperature for SRB Propellant Temperature Predictions. The appropriate KSC and VAFB monthly mean and extreme ambient temperatures listed below shall be used to establish SRB propellant temperatures and thrust performance. The low and high extremes are the 2.5 and 97.5 percentile average monthly temperatures.

5

NASA TECHNICAL MEMORANDUM

NASA TM X-64757

TERRESTRIAL ENVIRONMENT (CLIMATIC) CRITERIA GUIDELINES FOR USE IN AEROSPACE VEHICLE DEVELOPMENT, 1973 REVISION

Glenn E. Daniels, Editor
Aero-Astrodynamics Laboratory

July 5, 1973

NASA

George C. Marshall Space Flight Center
Marshall Space Flight Center, Alabama

6

MSFC - Form 3190 (Rev June 1971)

Table 10.1-1

Ground Thermal Environment

Thermal Environment Factor	Ferry Sites	Vertical Flight
Air Temperature (Degrees F) Design High	103	99
Low	20	31
Solar Radiation (Btu/ft²-hr) Design High	See Figure 10-1	
Low (Diffuse)		
Sky Temperature (Degrees F) Design High	50	
Low	-22	5

Local Standard Time - Hour

Time of Day	Design High Solar Radiation		Design Low Solar Radiation	
Hour	BTU/ft²/hr	gm-cal/cm²/min	BTU/ft²/hr	gm-cal/cm²/min
0500	0	0.00	0	0.00
1100	363	1.64	70	0.32
1300			80	0.36
1400	363	1.64		
2000	0	0.00	0	0.00

7

2.5.3.4 **Solar Radiation during Extreme Conditions**

When ground winds occur exceeding the 95, 99, or 99.9 percentile design winds given in this document in Section V, the associated weather normally is such that clouds, rain, or dust are generally present; therefore, the intensity of the incoming solar radiation will be less than the maximum values given in Tables 2.3 and 2.4. Maximum values of solar radiation intensity to use with corresponding wind speeds are given in Table 2.5.

TABLE 2.5 SOLAR RADIATION MAXIMUM VALUES ASSOCIATED WITH EXTREME WIND VALUES

Maximum Solar Radiation (Normal Incident)						
Steady-State Ground Wind Speed at 18 m Height	Huntsville, New Orleans River Transportation, Gulf Transportation, Eastern Test Range, Western Test Range, Sacramento, West Coast Transportation and Wallops Test Range			White Sands Missile Range		
($m\ sec^{-1}$)	($kJ m^{-2} sec^{-1}$)	($g\text{-}cal\ cm^{-2} min^{-1}$)	($BTU\ ft^{-2} hr^{-1}$)	($kJ m^{-2} sec^{-1}$)	($g\text{-}cal\ cm^{-2} min^{-1}$)	($BTU\ ft^{-2} hr^{-1}$)
10	0.84	1.20	266	1.05	1.50	332
15	0.56	0.80	177	0.70	1.00	221
≥20	0.35	0.50	111	0.56	0.80	177

2.6 <u>Temperature</u>

Several types of temperatures at the earth's boundary layer may be considered in design. These are as follows:

a. Air temperature normally measured at 1.22 meters (4 ft) above a grass surface.

b. Changes of air temperature (Usually the rapid changes which occur in less than 24 hours are considered.)

c. Surface or skin temperature measured of a surface exposed to radiation.

d. Temperatures within a closed compartment.

All of the above will be discussed in the following subsections.

9

2.6.1 Air Temperature Near the Surface

Surface air temperature extremes (maximum, minimum, and the 95 percentile values) and the extreme minimum sky radiation (equal to the outgoing radiation) are given in Table 2.6 for various geographical areas. Maximum and minimum temperature values should be expected to last only a few hours during a daily period. Generally, the maximum temperature is reached after 12 noon and before 5 p.m., while the minimum temperature is reached just before sunrise. Table 2.7A shows the maximum and minimum air temperatures which have occurred on each hour at Kennedy Space Center, but not necessarily on the same day, although these curves represent a cold and hot extreme day. The method of sampling the day (frequency of occurrence of observations) will result in the same extreme values if the same period of time for the data is used, but the 95 percentile values will be different for hourly, daily, and monthly data reference periods. Selection of the reference period depends on engineering application. Table 2.7B gives month mean temperatures, standard deviations and 2.5 and 97.5 percentiles of values of temperature for Kennedy Space Center, Florida and Vandenberg AFB, California.

2.6.2 Extreme Air Temperature Change

a. For all areas the design values of extreme air temperature changes (thermal shock) are:

(1) An increase of air temperature of 10°C (18°F) with a simultaneous increase of solar radiation (measured on a normal surface) from 0.50 g-cal cm^{-2} min^{-1} (110 BTU ft^{-2} hr^{-1}) to 1.85 g-cal cm^{-2} min^{-1} (410 BTU ft^{-2} hr^{-1}) may occur in a 1-hour period. Likewise, the reverse change of the same magnitude may occur for decreasing air temperature and solar radiation.

(2) A 24-hour change may occur with an increase of 27.7° C (50°F) in air temperature in a 5-hour period, followed by 4 hours of constant air temperature, then a decrease of 27.7°C (50°F) in a 5-hour period, followed by 10 hours of constant air temperature.

b. For Eastern Test Range (Kennedy Space Center), the 99.9 percentile air temperature changes are as follows:

(1) An increase of air temperature of 5.6°C (11°F) with a simultaneous increase of solar radiation (measured on a normal surface) from 0.50 g-cal cm^{-2} min^{-1} (110 BTU ft^{-2} hr^{-1}) to 1.60 g-cal cm^{-2} min^{-1} (354 BTU ft^{-2} hr^{-1}), or a decrease of air temperature of 9.4°C (17°F) with a simultaneous decrease of solar radiation from 1.60 g-cal cm^{-2} min^{-1} (354 BTU ft^{-2} hr^{-1}) to 0.50 g-cal cm^{-2} min^{-1} (110 BTU ft^{-2} hr^{-1}) may occur in a 1-hour period. 9

2.26

TABLE 2.7 MAXIMUM AND MINIMUM SURFACE AIR TEMPERATURES AT EACH HOUR FOR EASTERN TEST RANGE[4]

Time	Annual Maximum		Annual Minimum	
	°C	°F	°C	°F
1 a.m.	28.9	84	1.1	34
2	28.9	84	0.6	33
3	29.4	85	-1.1	30
4	28.3	83	-0.6	29
5	28.3	83	-1.1	28
6	29.4	85	-1.1	27
7	30.6	87	-1.7	26
8	30.6	87	-2.2	25
9	31.7	89	-0.6	28
10	33.9	93	1.1	30
11	35.0	95	2.2	36
12 noon	35.6	96	5.0	41
1 p.m.	37.2	99	5.6	42
2	35.6	97	5.0	41
3	35.6	97	5.6	42
4	35.6	97	5.6	42
5	35.6	97	5.6	42
6	35.0	95	3.9	39
7	33.3	92	2.2	36
8	31.7	89	2.2	36
9	30.0	86	1.7	35
10	30.0	86	1.7	35
11	30.0	86	1.1	34
12 mid	30.0	86	1.1	34

4. Based on 10 years of record for Patrick Air Force Base and Kennedy Space Center.

(10)

2.0 APPLICABLE DOCUMENTS. The below listed documents form a part of this appendix to the extent specified herein. These documents shall be individually approved as baseline requirements. The "Current Issue" of each document may be determined from JSC 08102, Space Shuttle Program Level II Baseline Description and Status Report.

Contractor Handbooks

SD73-SH-0069-1 (Current Issue)	Structural Design Loads Data Book, Baseline Vehicle and Missions Ref. Para. 3.3, 3.3.1, Table 10.11.1
SD73-SH-0069-2 (Current Issue)	Orbiter Structural Design Loads Data Book Ref. Para. 3.3, 3.3.2, Table 10.11.1
SD73-SH-0069-3 (Current Issue)	Structural Design Loads Data Book, External Tank Ref. Para. 3.3, 3.3.3, Table 10.11.1
SD73-SH-0069-4 (Current Issue)	Structural Design Loads Data Book, Solid Rocket Boosters Structural Loads Ref. Para. 3.3, 3.3.4, Table 10.11.1
SD73-SH-0181-1 (Current Issue)	Aerodynamic Heating Data Book, Orbiter - Ascent (Books I, II and III) Ref. Para. 3.1, 3.1.1, Table 10.11.1
SD73-SH-0181-2 (Current Issue)	Aerodynamic Heating Data Book External Tank - Ascent (Books I and II) Ref. Para. 3.1, 3.1.1, Table 10.11.1
SD73-SH-0181-3 (Current Issue)	Aerodynamic Heating Data Book, Shuttle Vehicle Booster - Ascent (Books I and II) Ref. Para. 3.1, 3.1.1, Table 10.11.1

11

SD73-SH-0181-4 (Current Issue)	Aerodynamic Heating Data Book, Space Shuttle Main Engine Ref. Para. 3.1, 3.1.1, Table 10.11.1
SD73-SH-0181-5 (Current Issue)	Aerodynamic Heating Data Book, Shuttle Launch Facility Ref. Para. 3.1, 3.1.1, Table 10.11.1
SD73-SH-0181-7 (Current Issue)	Aerodynamic Heating Data Book, Lightweight External Tank-Ascent Ref. Para. 3.1
SD74-SH-0082 (Current Issue)	Acoustics and Shock Data Book, Space Shuttle System Ref. Para. 3.4, 3.5, Table 10.11.1
SD74-SH-0144 (Current Issue)	Thermal Interfaces Design Data Book Ref. Para. 3.2, Table 10.11.1

SD 74-SH-0144D

SPACE SHUTTLE PROGRAM
THERMAL INTERFACES DESIGN DATA BOOK

DECEMBER 1977

SDM BASELINE

Contract NAS9-14000
IRD No. SE-699T2
WBS 2.2.1

Prepared by

W.G. Casey, Supervisor
RCC and Penetrations Analysis

C.A. Laren
RCC and Penetrations Analysis

Approved by

P.C. Marhoff, Manager
Thermal Analysis

N.F. Witte, Director
Shuttle Aero Sciences

Rockwell International
Space Division

13

Rockwell International
Space Division

1.0 INTRODUCTION

1.1 SCOPE

Contained herein are the thermal design data required to complete the thermal interface definitions for the following interface control documents (ICD's):

ICD 2-12001	Orbiter Vehicle/External Tank
ICD 2-14001	Orbiter Vehicle/Solid Rocket Booster
ICD 2-24001	External Tank/Solid Rocket Booster
ICD 13M-15000	Space Shuttle Orbiter Vehicle/Main Engine

1.2 APPLICABLE DOCUMENTS

The natural environments that have an effect on the thermal design are defined in the following documents:

JSC 07700, Space Shuttle Flight and Ground Systems Specification, Volume X, Appendix 10.10, Natural Environment Design Requirements.

NASA TM X-64795, Distribution of Eight Meteorological Variables at Cape Kennedy, Florida, and Vandenberg Air Force Base, California; Marshall Space Flight Center (dated November 19, 1973).

The induced environments applicable to the Shuttle system elements performing as part of the integrated flight vehicle are defined in the following data books:

SD 73-SH-0181-1A, Space Shuttle Aerodynamic Heating Data Book – Orbiter Ascent, Volume I (dated February 1975)

SD 73-SH-0181-2, Space Shuttle Aerodynamic Heating Data Book – External Tank Ascent, Volume II (dated June 1976).

SD 73-SH-0183-3, Space Shuttle Aerodynamic Heating Data Book – Shuttle Vehicle Booster Ascent, Volume III (dated September 1976).

SD 73-SH-0181-4, Space Shuttle Aerodynamic Heating Data Book – Space Shuttle Main Engine Ascent, On-Orbit and Entry, Volume IV (dated September 1977)

These books form part of the Space Shuttle Flight and Ground Systems Specification, JSC 07700, Volume X, Appendix 10.11, Induced Environment Design Requirements. The induced environments data are available on magnetic tape records as specified in Reference 1 for the orbiter, Reference 2 for the external tank, and Reference 3 for the Shuttle vehicle booster.

(14)

2.4 SURFACE TEMPERATURE RESPONSES

Surface temperature responses are provided to establish the radiant sources and sinks for the Shuttle vehicle from prelaunch through ascent. These data are not to be construed as material temperature limits. They are intended as thermal interface data required for element thermal design analysis.

2.4.1 Prelaunch

The prelaunch, post-fill surface temperature histories are presented herein for the hot and cold day environments specified in the Space Shuttle Flight and Ground Systems Specification, JSC 07700, Volume X, Appendix 10.10, Natural Environment Design Requirements. The 95th percentile day for July at Cape Kennedy was used for the extreme hot day environment. For the extreme cold day environment, the 5th percentile day for January at Vandenberg Air Force Base was used. External tank fill was assumed at 1100 hours for all cases.

2.4.1.1 Orbiter Vehicle

The prelaunch environmental temperatures for the fuselage lower external surfaces (Zones 01 through 05, Figure 2.1-1) are presented in Figures 2.4.1.1-1 through 2.4.1.1-4. For the wing leading edge (Zones 06 through 09, Figure 2.1-1), the external surface temperature histories for prelaunch are given in Figures 2.4.1.1-5 and 2.4.1.1-6. The temperature variations for the wing lower external surfaces (Zones 010 through 012, Figure 2.1-1) during prelaunch are provided in Figures 2.4.1.1-7 through 2.4.1.1-9.

2.4.1.2 External Tank (ET)

Figures 2.4.1.2-1 through 2.4.1.2-3 provide the temperature response of the external tank surfaces defined in Figure 2.1-2 when subjected to a hot day prelaunch environment. The cold day prelaunch environment temperature variations for the external tank surfaces (Figure 2.1-2) are presented in Figures 2.4.1.2-4 through 2.4.1.2-6. The temperatures to be used for the ET crossbeam surfaces (Figure 2.1-3) are 98°F for hot day and 29°F for cold day prelaunch environments.

2.4.1.3 Solid Rocket Booster (SRB)

Temperature profiles for the SRB external surfaces defined in Figure 2.1-4 are provided in Figure 2.4.1.3-1 for the hot day prelaunch environment and Figure 2.4.1.3-2 for the cold day prelaunch environment.

2.4.1.4 Space Shuttle Main Engine (SSME)

Hot day prelaunch environmental temperature histories for the Space Shuttle main engine interface envelope, defined in Figure 2.4.1.4-3, are presented in Figure 2.4.1.4-1. The SSME compartment surface temperatures are given in Figure 2.4.1.4-2 for a cold day prelaunch environment.

NASA TECHNICAL MEMORANDUM

NASA TM X-64795

DISTRIBUTIONS OF EIGHT METEOROLOGICAL VARIABLES AT CAPE KENNEDY, FLORIDA AND VANDENBERG AIR FORCE BASE, CALIFORNIA

By M. E. Graves, R. L. King, and S. Clark Brown
Aero-Astrodynamics Laboratory

November 19, 1973

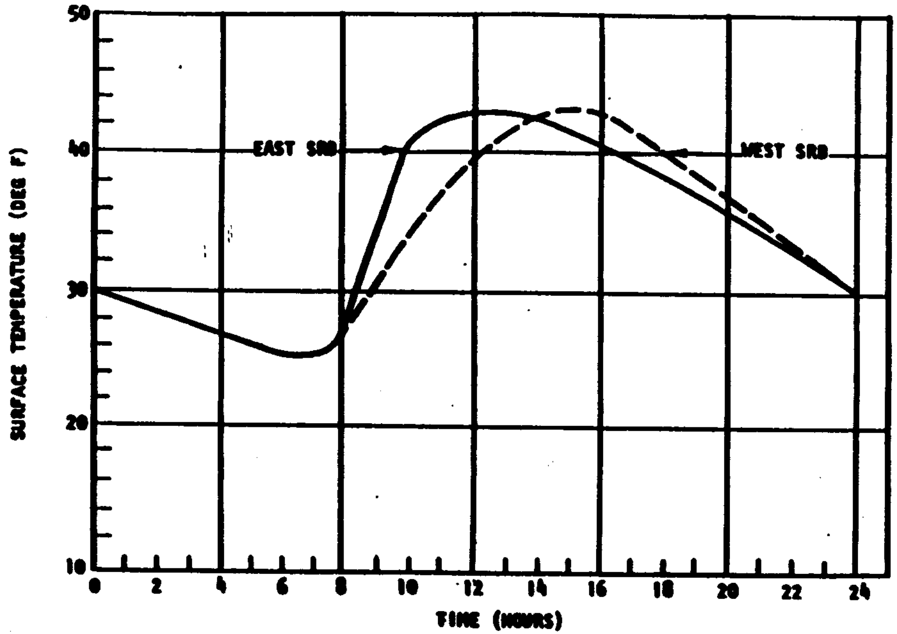
NASA

George C. Marshall Space Flight Center
Marshall Space Flight Center, Alabama

16

Rockwell International
Space Division

17



AMBIENT TEMP °F 37.4 36.0 34.5 48.9 50.9 50.3 42.4 39.5

Figure 2.4.1.3-2. Solid Rocket Booster Surface Temperature, Zones B1 Through B10 - Prelaunch Environment, Cold Day

*ET FILL COMPLETE AT 1100 HRS

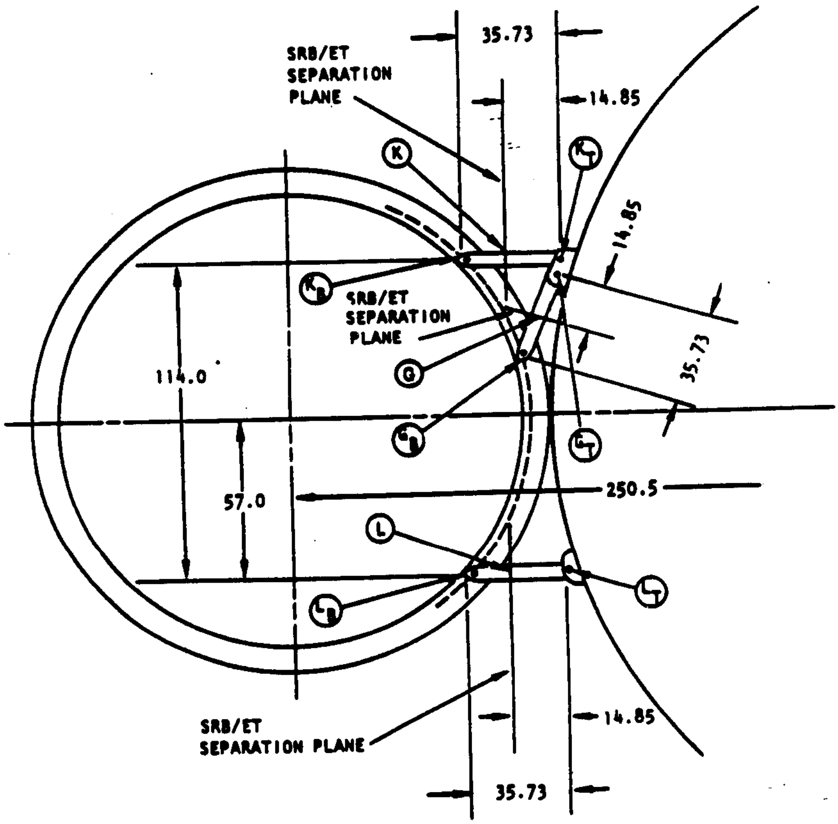


Figure 3.1.2-4. ET/SRB Aft Attachment

3.2.3 Solid Rocket Booster

Temperature extremes for the SRB side of the following conduction interfaces:

ET/SRB forward attachment (Figure 3.1.2-3)

ET/SRB aft attachment (Figure 3.1.2-4)

are presented in Table 3.2.3-1 for hot day (maximum) and cold day (minimum).

Table 3.2.3-1. Solid Rocket Booster/External Tank Conduction Interface Temperatures—Prelaunch Environment

Location	Temperature (deg F)	
	Hot Day (maximum)	Cold Day (minimum)
Forward attachment, Location F_B	89	39
Forward attachment, Location F	90	26
Aft attachment, Locations G_B, K_B, and L_B	96	21

3.2.4 Space Shuttle Main Engine

Temperature histories for the SSME side of the following interfaces:

LPFTP flange (Figure 3.1.1-3)

LPOTP flange (Figure 3.1.1-4)

Gimbal bearing (Figure 3.1.1-5)

Heat shield attachment (Figure 3.1.1-6)

TVC actuator attachment (Figure 3.1.1-7)

are presented in Figures 3.2.4-1 through 3.2.4-5.

3.2.5 Mobile Launch Platform

Maximum temperatures for the MLP side of the following condition interfaces:

SRB/MLP hold-down support hardware (Figure 3.1.3-1)

are presented in Table 3.2.5-1

Specification No. CPW1-3300

PRIME EQUIPMENT CONTRACT END ITEM
DETAIL SPECIFICATION
PART I OF TWO PARTS

PERFORMANCE, DESIGN AND VERIFICATION REQUIREMENTS
SPACE SHUTTLE HIGH PERFORMANCE, SOLID
ROCKET MOTOR LIGHTWEIGHT
CPW1-3300
FOR
SPACE SHUTTLE SOLID
ROCKET MOTOR PROJECT
OPERATIONAL FLIGHT
(STS-8, 12 & Subs)

17 February 1984

PENDING NASA APPROVAL

DR NO. 2-2

PREPARED FOR

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
GEORGE C. MARSHALL SPACE FLIGHT CENTER
MARSHALL SPACE FLIGHT CENTER, ALABAMA 35812

BY

Morton Thiokol, Inc.
Wasatch Division
P.O. Box 524, Brigham City, Utah 84302 801/863-3511

maintenance and refurbishment operations.

3.2.6.3 Personnel Safety. Provisions for personnel safety shall be in accordance with the following:

a. Safety Devices. Known hazards which cannot be eliminated through design selection shall be reduced to an acceptable level through the use of appropriate safety devices as part of the system, subsystem, or equipment.

b. Warning Devices. Where it is not possible to preclude the existence or occurrence of a known hazard, devices shall be employed for the timely detection of the condition and the generation of an adequate warning signal. Warning signals and their application shall be designed to minimize the probability of wrong signals or of improper personnel reaction to the signal.

3.2.6.4 Explosive and/or Ordnance Safety. The propellants for the HPML and the igniter shall meet the requirements of hazard classification 2 as defined in the Army Material Command Regulation Safety Manual AMCR 385-100, or DoD Contractor's Safety Manual for Ammunition, Explosives, and Related Dangerous Materials, DoD 4145.26. The HPML segments and ignition system less initiators shall have a DoT explosive classification of Class B.

3.2.7 Environment.

3.2.7.1 Natural Environment. The HPML shall withstand the natural environments defined in JSC 07700, Volume X, Appendix 10.10 and the air and sea temperature environments and salinity of SE-019-043-2H. 31°

3.2.7.2 Induced Environment. The HPML shall withstand the induced environmental conditions as defined in the following documents:

Thermal
Base Heating - SD73-SH-0181-3
Launch & Ascent - SD73-SH-0181-3
Re-entry - SE-019-053-2H
Interface - SD74-SH-0144, ICD 3-44003

25° + 21°

Loads
Vibration, Acoustic & Shock SE-019-049-2H and SE-019-067-2H (as changed by approved Deviation RDW-0012R4) In case of conflict, SE-019-049-2H shall take precedence over SE-019-067-2H.
Prelaunch through Separation - SE-019-057-2H, Book 1

21

TABLE V

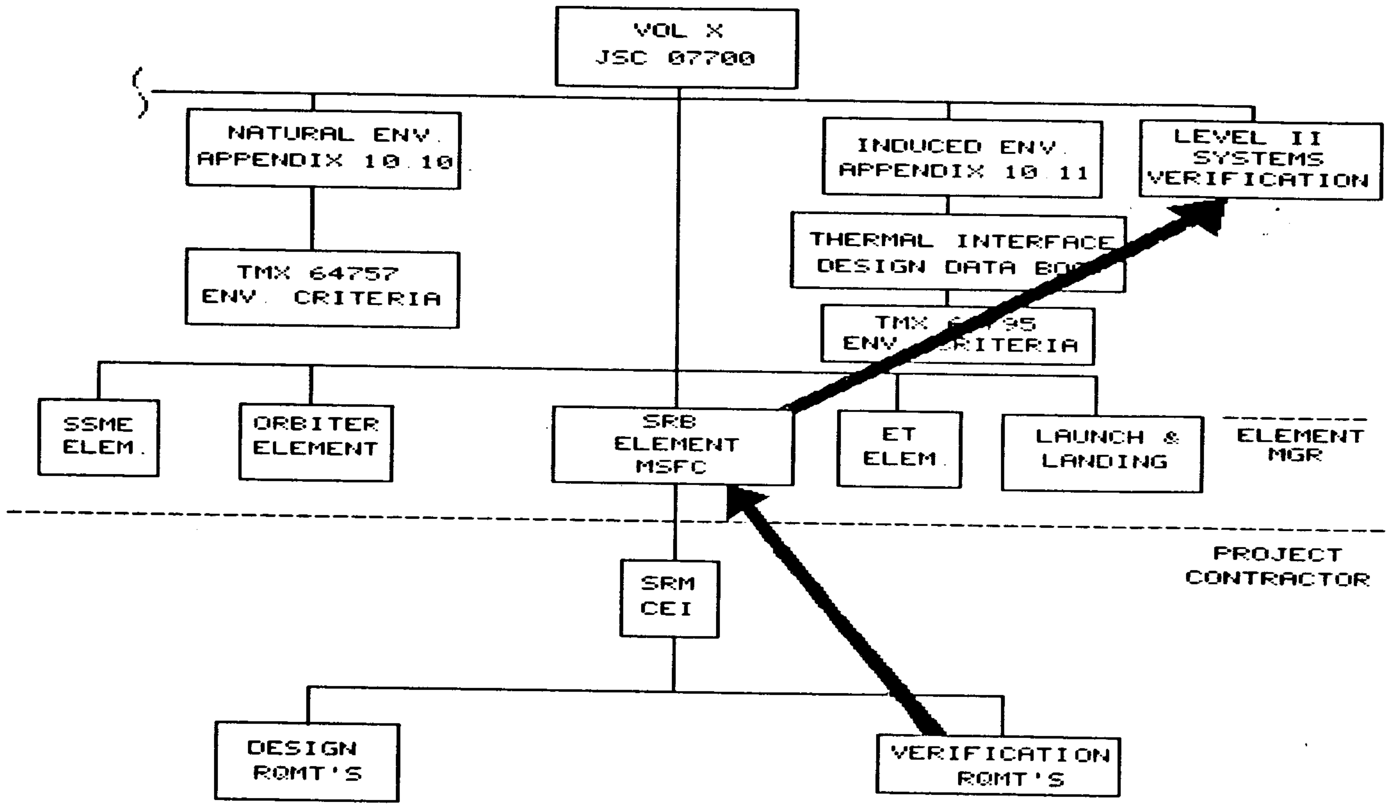
	VERIFICATION CROSS REFERENCE INDEX	CPW1-3300 Dated 17 February 1984 Page I-92

REQUIREMENTS FOR VERIFICATION

VERIFICATION METHOD:	VERIFICATION PHASE:
1. Similarity	A. Development
2. Analysis	B. Qualification
3. Inspection	C. Acceptance
4. Demonstration	D. Preflight
5. Test	E. Flight
	F. Postflight
N/A - Not Applicable	

Section 3.0 Performance/ Design Requirement Reference	Verification Methods							Section 4.0 Verification Requirement Reference
	N/A	A	B	C	D	E	F	
3.2.6.3 Personnel Safety		2	2					4.2.1.2, 4.2.2.2
3.2.6.4 Explosive and/or Ordnance Safety		5	5					4.2.1.5, 4.2.2.5
3.2.7 Environment	X							
3.2.7.1 Natural Environment		2	2					4.2.1.2, 4.2.2.2,
3.2.7.2 Induced Environment		2	2					4.2.1.2, 4.2.2.2
3.2.8 Transportability/ Transportation		1	1		4		4	4.2.1.1, 4.2.2.1, 4.2.5.4, 4.2.7
3.2.9 Storage	X							
3.2.9.1 Post Acceptance Requirements				2				4.2.3.2
3.2.9.2 Storage/ Age Control		3	3		3			4.2.1.3, 4.2.2.3, 4.2.5.3

22


VOL X
JSC 07700
NATURAL ENV.
APPENDIX 10.10
TMX 64757
ENV. CRITERIA
INDUCED ENV.
APPENDIX 10.11
LEVEL II
SYSTEMS
VERIFICATION
THERMAL INTERFACE
DESIGN DATA BO
TMX
ENV. CRITERIA
SSME
ELEM.
ORBITER
ELEMENT
SRB
ELEMENT
MSFC
ET
ELEM.
LAUNCH &
LANDING
ELEMENT
MGR
PROJECT
CONTRACTOR
SRM
CEI
DESIGN
RQMT'S
VERIFICATION
RQMT'S

VOLX1.SKE

SPECIFICATION VERIFICATION

O PROJECTS REQUIRED TO SHOW THAT EACH REQUIREMENT IN THE CONTRACT SPECIFICATION HAS BEEN VERIFIED

O ELEMENT MANAGERS REQUIRED TO VERIFY COMPLIANCE WITH LEVEL II REQUIREMENTS THRU VERIFICATION COMPLETION NOTICES

23

- 1S - 1

Rockwell International
Space Systems Group

SPACE SHUTTLE SYSTEM
INTEGRATED ELEMENTS

NASA

VERIFICATION COMPLETION NOTICE

(FRF/FMOF)

PAGE 1 OF 3

IVLN NO.: 12

VCN NO.: 12A11

IVLN TITLE: BASELINE MISSION CAPABILITY

DATE:

CONSTRAINT: FMOF

SCOPE OF IVLN

The Baseline Mission Capability IVLN, as applicable to this VCN, verifies those activities and interrelationships that apply to FMOF constraints. STS-1 flight performance has been evaluated by trajectory simulations utilizing propulsion tag values and specified SRB performance. In addition to flight performance reserves and intact abort, yaw steering, insertion point accuracy, ET disposal and flight personnel loads have been evaluated and found satisfactory for STS-1 flight. (See Continuation Sheet)

JSC — 07700 VOL X VERIFICATION REQUIREMENTS COMPLETED

3.2.1.1.2	3.2.1.1.3.4	3.2.1.1.17
3.2.1.1.3	3.2.1.1.4	3.2.1.2.14
3.2.1.1.3.1	3.2.1.1.4.1	3.2.2.1.17.2 ←
3.2.1.1.3.2	3.2.1.1.5	3.2.2.1.17.2.1
3.2.1.1.3.3.1	3.2.1.1.6	3.3.1.2.2.1
3.2.1.1.3.3.2	3.2.1.1.11	3.3.1.3.3.10
		3.3.1.2.5.1

(SEE THE BACK SIDE OF THIS SHEET FOR INCOMPLETE VERIFICATION OR EXCEPTIONS)

APPROVALS

Except as noted on the back side of this sheet:

(1) Endorsement of this VCN by the Project Managers shall signify completion of element verification to the extent identified on the above cited IVLN and as further defined in the applicable Shuttle Master Verification Plan.

(2) Endorsement by the Systems Integration Technical Managers shall signify completion of the verification tasks and products defined in the attached VIS-2 data file segments.

(3) Systems Integration Manager approval signifies completion of the total integrated systems verification task for the identified network.

ROCKWELL			NASA			
	SIGNATURE AND ORGN	DATE		SIGNATURE AND ORGN	DATE	SIGNATURE AND ORGN / DATE
TECHNICAL MANAGERS	[illegible] - Aero	1/6/81	PROJECT	[illegible] for [illegible]	1-23-81	[illegible] 1/19
	R. H. [illegible] - Ascent	1/6/81		[illegible]		[illegible] 1/19
				[illegible]	1-9-81	[illegible] 1-19-81
	[illegible] for GDS-[illegible]	12-31-80	TECHNICAL MGRS	[illegible]	1/8/81	[illegible] 1-19-81
	[illegible]			[illegible]	1/1/81	[illegible] 1/26/81
	[illegible]	1/15/81		[illegible] ES3	1/1/81	
	[illegible]	1/15/81		[illegible]	1/15/81	
	[illegible]			[illegible]		
	[illegible]	1/15/81		[illegible]	1/16/81	
	[illegible] EXT. LOADS	1/16/81				
	[illegible] CPE	1/19/81				
SYSTEM	[illegible]	1/16/81	SYSTEM	[illegible]	3-17-81	
	PROGRAM MANAGEMENT:			PROGRAM MANAGEMENT: SIGNATURE/DATE		

(24)

Flight 5 + Subs

Rockwell International
Space Systems Group

SPACE SHUTTLE SYSTEM
INTEGRATED ELEMENTS

NASA

VERIFICATION COMPLETION NOTICE

PAGE 1 OF 2

IVLN NO.: 12

VCN NO.: 12A21

IVLN TITLE: BASELINE MISSION CAPABILITY

DATE: OCT 27 1982

CONSTRAINT: OPS

SCOPE OF IVLN

This Baseline Mission Capability IVLN identifies the activities and interrelationships occuring during the Orbital Flight Test phase of the program applicable to veriying the capability of the vehicle to perform the baseline missions specified in the applicable paragraphs of JSC-07700-10. The verification activities accomplished prior to the first flight are identified in VCN No. 12A11. (See continuation sheet)

JSC – 07700 VOL X VERIFICATION REQUIREMENTS COMPLETED

3.2.1.1.2	3.2.1.1.3.4	3.2.1.1.17
3.2.1.1.3	3.2.1.1.4	3.2.1.2.14
3.2.1.1.3.1	3.2.1.1.4.1	3.2.2.1.17.2 ←
3.2.1.1.3.2	3.2.1.1.5	3.2.2.1.17.2.1
3.2.1.1.3.3.1	3.2.1.1.6	3.3.1.2.2.1
3.2.1.1.3.3.2	3.2.1.1.11	3.3.1.3.3.10

(SEE THE BACK SIDE OF THIS SHEET FOR INCOMPLETE VERIFICATION OR EXCEPTIONS)

APPROVALS

Except as noted on the back side of this sheet:

(1) Endorsement of this VCN by the Project Manager shall signify completion of element verification to the extent identified on the above cited IVLN and as further defined in the applicable Shuttle Master Verification Plan.

(2) Endorsement by the Systems Integration Technical Managers shall signify completion of the verification tasks and products defined in the attached VIB-2 data file segments.

(3) Systems Integration Manager approval signifies completion of the total integrated systems verification task for the identified network.

ROCKWELL s/*

	SIGNATURE AND ORGN	DATE
TECHNICAL MANAGERS	R.H.Lassen Acoustics	11-4-82
	R.H.Lassen Flutter	11-4-82
	R.A. Weaver Integ.	11-5-82
	R.A.Burg MPS	11-5-82
	W.T.Schleich AGN&C	11-15-82
	W.T.Schleich Sep	11-15-82
	J.W.Haney Aerothermo	11-22-82
	M.R.Schall AscFlt Perf.	11-15-82
	E.D.Schlessinger Therm	11-22-82
	W.E. Bornemann Aero	11-24-82
	R.E. Gatto Ext lds	12-1-82
SYSTEM	*signatures on file B. A. Gerstner PROGRAM MANAGEMENT:	

NASA

	SIGNATURE AND ORGN	DATE	SIGNATURE AND ORGN	DATE
PROJECT			J. Lovingood	12-8-82
	J.B. Odom	12-8-82	R. H. Gray	11-5-82
	L.B. Mulloy	12-8-82		
TECHNICAL MGRS	M.K. Craig Sep	11-3-82	K.J. Cox GN&C	11-3-82
	B.B. Roberts Sep	11-3-82	H.J. Brasseaux MPS	12-14-
	J.G. Hondros Aero	11-16-82	M.C. Coody Acou	11-2-82
	J.G. Hondros Asc Flt Perf.	11-16-82	G. Strouhal Therm	11-15-
	A.C. Mackey Ext Lds	11-10-82		
	D.B. Lee Thermo	11-22-82		
	C.T. Modlin Flutter	11-29-82		
SYSTEM	PROGRAM MANAGEMENT:		SIGNATURE/DATE	

25

VI-B

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DIRECTOR, RESEARCH AND ENGINEERING

MAXIME FAGET, NASA JOHNSON SPACE CENTER (RETIRED)
SPACE INDUSTRIES

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DOMINIC SANCHINI , ROCKWELL INTERNATIONAL

SAMUEL TENNANT, AEROSPACE CORPORATION

DAVID L. WINTERHALTER, NASA HEADQUARTERS
ACTING DIRECTOR, PROPULSION DIVISION

JOHN YOUNG, NASA JOHNSON SPACE CENTER
CHIEF, ASTRONAUT OFFICE

June 19, 1986

NATIONAL RESEARCH COUNCIL
COMMISSION ON ENGINEERING AND TECHNICAL SYSTEMS
Committee on NASA Scientific and Technological Program Reviews

PANEL ON TECHNICAL EVALUATION OF NASA'S PROPOSED REDESIGN OF THE SPACE SHUTTLE SOLID ROCKET BOOSTER

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Dr. Robert H. Korkegi (Co-Director)

Dr. Myron F. Uman (Co-Director)

Ms. Viviane Scott (Adm. Assistant)

VI-C

INFORMATION REQUIREMENT DESCRIPTION

Title:	Type	No.	Date Rev.
Failure Mode Effects Analysis	2	RA-26703	

Submittal Schedule:

Submit updates at 6-month intervals with submittal linked to nearest schedule CIL FRR submittal.

Contract SOW Reference:

Exhibit A, 3.3.3 - Engineering Support

Use:

To identify critical failure modes to be used as a basis for support of: (1) Additional Design Action; (2) Safety Analysis; and (3) Mission Contingency Planning.

Interrelationship:

Scope-Contents-Format-Maintenance-Government Furnished Data:

1. Scope/Contents - Failure mode effects analysis will be prepared for each Orbiter Vehicle Subsystem, including the following:

   a. System/Subsystem/Assembly/Item - Identification of item for which the FMEA is being conducted.

   b. Prepared/Approved By - Identification of analyst who performed the FMEA and individuals responsible for overall FMEA effort.

   c. Revision - Date individual pages are revised.

   d. Item Identification:

      (1) Name

      (2) Identification Number - Drawing number by which the Contractor identifies and describes each component or module.

      (3) Drawing Reference Designation - Identification of the component or module on the schematic.

      (4) Quantity - Total number of items in the subsystem.

   e. FMEA Number - A number that uniquely identifies the subsystem, component, and failure mode.

   f. Function - Concise statement of the function performed.

INFORMATION REQUIREMENT DESCRIPTION

Title:	No.	Date Rev.
Failure Mode Effects Analysis	RA-2678B	

2. Scope/Contents (Continued)

g. Failure Mode and Cause - Identification of the specific failure mode after considering the four basic failure conditions:

(1) Premature operation.
(2) Failure to operate at a prescribed time.
(3) Failure to cease operation at a prescribed time.
(4) Failure during operation.

For each applicable failure mode, describe the major cause(s) including operational and environmental stress factors, if known.

h. Mission/Phase - Phase of mission in which failure occurs, e.g., Prelaunch: checkout, countdown; Flight: boost phase, earth orbit, etc.

i. Failure Effect on - Subsystem, interfacing subsystem, mission/crew, element and/or vehicle as required.

j. Failure Detection Method - A description of the methods by which the failure could be detected.

k. Correcting Action - An identification of correcting action, automatic or manual, which would be taken to circumvent the failure. Include statement of alternate means of operation and redundancy available after failure.

l. Failure Mode Criticality Category Designation - Categorize the failure mode criticality in relation to crew safety and mission effect. Include an identification of all items not meeting redundancy requirements during intact aborts.

Equipment other than criticality 1 shall be further evaluated in accordance with the redundancy hardware screens described below. A notation will be made identifying each screen the hardware does not pass.

(1) The redundant elements are not capable of checkout during the normal mission turnaround sequence, or

(2) Loss of a redundant element if not readily detectable by the flight crew, or

(3) All redundant elements can be lost by a single credible cause or event such as contamination or explosion.

m. Ground Rules and Assumption - Statement of all ground rules and assumptions followed during the performance of FMEA.

INFORMATION REQUIREMENT DESCRIPTION

Title: Failure Mode Effects Analysis	No. RA-267E3	Data Rev.

2. Scope/Contents (Continued)

   n. Remarks/Hazards - Statement of any remarks, recommendations, and potential hazards as required.

   o. Vehicle Effectivity - Identification of the vehicle effectivity for the failure mode identified.

3. Format - To be prepared in Contractor's format.

4. Maintenance - To be maintained by page revision/total reissuance, as applicable.

5. Government Furnished Data - Not applicable.

INFORMATION REQUIREMENT DESCRIPTION

Title:	Type	No.	Date Rev.
Failure Mode Effects Analysis	2	RA-267E3	

Submittal Schedule:

Submit updates at 6-month intervals with submittal linked to nearest schedule CIL FRR submittal.

Contract SOW Reference:

Exhibit A, 3.3.3 - Engineering Support

Use:

To identify critical failure modes to be used as a basis for support of: (1) Additional Design Action; (2) Safety Analysis; and (3) Mission Contingency Planning.

Interrelationship:

Scope-Contents-Format-Maintenance-Government Furnished Data:

1. Scope/Contents - Failure mode effects analysis will be prepared for each Orbiter Vehicle Subsystem, including the following:

   a. System/Subsystem/Assembly/Item - Identification of item for which the FMEA is being conducted.

   b. Prepared/Approved By - Identification of analyst who performed the FMEA and individuals responsible for overall FMEA effort.

   c. Revision - Date individual pages are revised.

   d. Item Identification:

      (1) Name

      (2) Identification Number - Drawing number by which the Contractor identifies and describes each component or module.

      (3) Drawing Reference Designation - Identification of the component or module on the schematic.

      (4) Quantity - Total number of items in the subsystem.

   e. FMEA Number - A number that uniquely identifies the subsystem, component, and failure mode.

   f. Function - Concise statement of the function performed.

INFORMATION REQUIREMENT DESCRIPTION

Title:	NO.	Date Rev.
Failure Mode Effects Analysis	RA-2678B	

2. Scope/Contents (Continued)

g. Failure Mode and Cause - Identification of the specific failure mode after considering the four basic failure conditions:

(1) Premature operation.
(2) Failure to operate at a prescribed time.
(3) Failure to cease operation at a prescribed time.
(4) Failure during operation.

For each applicable failure mode, describe the major cause(s) including operational and environmental stress factors, if known.

h. Mission/Phase - Phase of mission in which failure occurs, e.g., Prelaunch: checkout, countdown; Flight: boost phase, earth orbit, etc.

i. Failure Effect on - Subsystem, interfacing subsystem, mission/crew, element and/or vehicle as required.

j. Failure Detection Method - A description of the methods by which the failure could be detected.

k. Correcting Action - An identification of correcting action, automatic or manual, which would be taken to circumvent the failure. Include statement of alternate means of operation and redundancy available after failure.

l. Failure Mode Criticality Category Designation - Categorize the failure mode criticality in relation to crew safety and mission effect. Include an identification of all items not meeting redundancy requirements during intact aborts.

Equipment other than criticality 1 shall be further evaluated in accordance with the redundancy hardware screens described below. A notation will be made identifying each screen the hardware does not pass.

(1) The redundant elements are not capable of checkout during the normal mission turnaround sequence, or

(2) Loss of a redundant element if not readily detectable by the flight crew, or

(3) All redundant elements can be lost by a single credible cause or event such as contamination or explosion.

m. Ground Rules and Assumption - Statement of all ground rules and assumptions followed during the performance of FMEA.

INFORMATION REQUIREMENT DESCRIPTION

Title:	No.	Date Rev.
Failure Mode Effects Analysis	RA-267E3	

2. Scope/Contents (Continued)

n. Remarks/Hazards - Statement of any remarks, recommendations, and potential hazards as required.

o. Vehicle Effectivity - Identification of the vehicle effectivity for the failure mode identified.

3. Format - To be prepared in Contractor's format.

4. Maintenance - To be maintained by page revision/total reissuance, as applicable.

5. Government Furnished Data - Not applicable.

VI-D

Date: January 31, 1984
Page 1 of 34
Approved: V.P. Ostrander
V.P. Ostrander,
Mgr., Reliability
Space Shuttle Program

RELIABILITY

DESK INSTRUCTION

No. 100-2G

FLIGHT HARDWARE

FAILURE MODE EFFECTS ANALYSIS (FMEA)
&
CRITICAL ITEMS LIST (CIL)

References: (1) EOM 70 1-5.1.1.1
70 2-6.3.1

(2) Reliability Desk Instruction No. 100-1 - Reliability Evaluation

(3) Reliability Desk Instruction No. 100-12 - Shuttle Element Interface

INTRODUCTION

Reliability of the design is the ultimate responsibility of Design. However, it is incumbent on other Engineering functions, including Reliability, to support the design engineer in discharging his responsibilities. The Failure Mode Effect Analysis (FMEA) is primary reliability technique for providing design and program support and constitutes a documented record of the design status and coordinated decisions.

1.0 PURPOSE

This desk instruction defines the procedures for generating, documenting and maintaining Failure Mode Effects Analyses (FMEA) and Critical Items Lists (CIL) for the Space Shuttle Orbiter subsystems in order to verify design adequacy with respect to inherent reliability.

2.0 DEFINITIONS

1. Failure - is the inability of a system, subsystem, component, or part to perform its required function within specified limits under specified conditions for a specified duration.

2. Failure Mode - a description of the manner in which an item can fail.

3. Hazard - is the presence of a potential risk situation caused by an unsafe act or condition.

4. Redundancy (depth of) - describes the available (number of) ways of performing a function.

5. Backup Mode of Operation - describes the available ways of performing a function utilizing "like" (identical) hardware.

6. Alternate Mode of Operation - describes any additional ways of performing a function utilizing "unlike" hardware.

7. Criticality - is the categorization of a hardware item by the worst case potential direct effect of failure of that item. In assigning hardware criticality, the availability of redundancy (backup or alternate) modes of operation is considered. Assignment of functional criticality, however, assumes the loss of all redundant (backup or alternate) hardware elements. The definition of criticality is shown in Table 2.0.

Table 2.0 - Criticality Definition

CRITICALITY	POTENTIAL EFFECT OF FAILURE
1	Loss of life or vehicle.
2	Loss of mission.
3	All others.
1R	Redundant hardware element, all of which if failed, could cause loss of life or vehicle.
2R	Redundant hardware element, all of which if failed, could cause loss of mission.

NOTE: See Appendix B, paragraph 3.1.1, Ground Rules, sub-paragraphs 1 and 2.

8. Single Failure Point (SFP) - is a single item of hardware, the failure of which would lead directly to loss of life, vehicle, or mission. Where safety considerations dictate that abort be initiated when a redundant item fails, that item is also considered a single failure point.

9. Functional Mode - identifies each function to be performed by the item being analyzed.

10. Multiple Order Failure - describes the failure due to a single cause or event of all units which perform a necessary (critical) function.

11. Critical Item - a single failure point and/or a redundant element in a life or mission-essential application where:
    a. Redundant elements are not capable of checkout during the normal ground turnaround sequence.
    b. Loss of a redundant element is not readily detectable in flight.
    c. All redundant elements can be lost by a single credible cause or event such as contamination or explosion.

12. Kit - For the purposes of this desk instruction, a kit is defined as a temporary addition or modification to the Orbiter or its subsystems to satisfy unique requirements for a specific mission.

13. Post Landing Safing Operations - For the purposes of this desk instruction, post landing safing operations are defined as those activities performed after landing to prepare the Orbiter for hangar operations. This includes the deservice and draining of all hazardous fluids, safing of unused ordnance, application of ground power and cooling, removal of potentially hazardous components, pods and payloads, purging and venting of gases and the installation of protective covers.

14. Prelaunch Operations - Prelaunch operations for propulsion subsystems is defined as beginning with propellant loading for each specific subsystem. For all other subsystems prelaunch operations commence with start of main engine conditioning.

3.0 FMEA/CIL PREPARATION

FMEA's will be prepared jointly by the responsible designer and the assigned Reliability Subsystem Analyst (RSA) in accordance with the attached format, Appendix D (Ground Rules and Criteria) and as shown in FIGURE 1. Safety, other engineering disciplines, and technical support functions (see EOM Directive

70 2-6.3.1) will provide support as required. Where the FMEA deviates from the instructions and ground rules contained herein, appropriate notation will be included within the "Ground Rules and Criteria" of the FMEA preface.

3.1 SCHEDULE

Reliability, in coordination with Design, will define the schedule and depth of detail for each FMEA to be prepared for the Orbiter in support of contractual requirements, and issue an FMEA schedule.

3.2 CONTENT

Each subsystem FMEA/CIL shall be prefaced by the following information, sequenced as indicated:

1.0 INTRODUCTION
2.0 QUALITATIVE RELIABILTY SUMMARY
  2.1 SUBSYSTEM DESCRIPTION AND EFFECTIVE DATE
  2.2 SIGNIFICANT UNDEFINED DESIGN AREAS
  2.3 CRITICAL ITEMS SUMMARY
3.0 GROUND RULES AND CRITERIA
4.0 DISPLAYS AND CONTROLS INDEX
5.0 LIST OF REFERENCE DOCUMENTS
6.0 SCHEMATICS

The backup information, including rationale and anaylses involved in assessing failure modes and their effects, generally is not included in the final FMEA and CIL package. Where such information exists in the form of notes, calculations, IL's, references and other similar material, it will be retained by the responsible RSA. Should the RSA be reassigned, he will turn over the material to his supervisor.

3.3 ANALYSIS REQUIREMENTS

1. FMEA's will be performed for each functional mode of a subsystem or functional kit. Electrical FMEA's will be conducted to the "black box" level and within the "black box" to pursue functions which have single failure point potential effect on the orbiter safety or mission success. The level of detail required in mechanical FMEA's below the component level in pursuit of critical failure modes will vary. Standard design, such as check valves, relief valves, isolation valves, etc., require only common types of failure causes to be listed.

EXAMPLE: Failure Mode - internal/external leakage.
Cause - poppet/seat damage, contamination, structural failure.

When a component is a non-standard type of design or is unique in application or contains unusual/unique failure modes of a critical nature, a more detailed analysis is required. Piece parts and their failure modes and effects that could result in component critical failure modes must be identified and included in the "CAUSE" section of the component FMEA for each component failure mode of concern.

EXAMPLE: Spring - fracture, structural failure - poppet fails to seat.

2. FMEA's for mechanical systems and avionics will interface at the connector. (See section 4.3.4, Mechanical/Electrical Interface.)

3. All identified failure modes will be assigned two criticalities (functional and hardware) based on the definitions in section 2.0, Definitions, and procedures contained in sections 4.3.1 and 4.3.2, Hardware and Functional Criticality Determination.

4. The criticality assigned to pressure carriers (pressure lines and vessels) shall reflect the worst case failure effect. These include potential shrapnel damage to the vehicle/subsystems resulting from rupture of non-filament wound tanks, potential overpressurization caused by releasing substantial quantities of fluids from ruptured lines or tanks, or depletion of consumables. Where released fluids are flammable or oxidizers and the possibility of an ignition source exists, appropriate notation will be entered under "HAZARDS" for Safety action. (See Appendix B, paragraph 3.1.1, Ground Rules, subparagraphs 13, 14, and 15.)

5. Failures which could occur during all mission phases from prelaunch through deactiviation (including safing & purging) of subsystems subsequent to landing and during ferry flights shall be considered, regardless of occurrence probability. Documentation of prelaunch analysis is required only for items classified as criticality 1/1.

6. All ordnance/pyrotechnic items will be listed in the CIL according to the most severe effect (criticality 1 or 2) of a premature operation.

7. Each hardware or function critical item summary will include a count of the total number of critical failure modes per item, by criticality, classified either structural or functional (see paragraph 4.1.11).

8. Critical item summaries for kits will be included, but identified separately.

9. FMEA's will not be required on structures, wire harnesses, cables and electrical connectors. For all critical circuits where a short between adjacent connector contacts could reslut in loss of crew (MSC D&P Standard No. 32), the design schematics shall be reviewed to verify that this condition does not exist. The incorporation of a switch on the ground side that precludes an adjacent contact short to result in crew loss is considered acceptable for meeting the MSC D&P Standard No. 32 requirement.

   For all other critical circuits, separation of redundant functions will be verified by selective review of design schematics to insure that the requirements for separation have been incorporated and complied with.

10. Logic diagrams (ref. Desk Instruction 100-1, Reliability Evaluation) will be developed only where required to provide proper correlation between schematics and FMEA's.

11. Those components that are criticality 3 (functional and hardware) in the electrical circuits by "black box" criticality may be listed on one FMEA for for that circuit. Those components that are hardware criticality 1 or 2 will have individual FMEA's. Those components that are criticality 1R or 2R, and appear in the CIL, will have individual FMEA's.

4.0 IMPLEMENTATION

A program has been developed to provide computer printout of FMEA and CIL data. Format examples of these printouts are shown in FIGURES 2 and 3. The following section contains instructions for documenting the FMEA. Data entry sheets (FIGURES 4 and 5) will be completed by the RSA as information becomes available. The information will be entered into the computer and the RSA will receive a copy of the resultant data printout (FIGURES 6 and 7) which will comprise a working document of the information stored in the computer and a baseline for additional inputs or revisions.

4.1 DATA ELEMENTS

The following procedure describes the information to be filled out on Data Sheets 1 and 2 (FIGURES 4 and 5). Each data descriptor is preceded by the entry code for that item (e.g., LV1, Subsystem ID). These codes also are shown on the examples of the FMEA and CIL formats, FIGURES 2 and 3, for information.

DATA SHEET NO. 1

4.1.1 (DI, LV1, LV2) DATA IDENTIFIER: This line uniquely identifies the component being analyzed and the "update" information to be taken.

a. Circle "A", "R" or "D" to indicate appropriate action --
   A - Add a new record (component or assembly).
   R - Review an existing record by adding, deleting, or revising an element(s) of that record.
   D - Delete an entire record and all information in that record.

b. SUBSYSTEM ID (LV1): Enter the last two digits of the applicable designator and dash number. (See TABLE 4.0).

c. COMPONENT ID (LV2): Enter a number which uniquely identifies the particular component being available. If an existing schematic identifier is available, it may be used. For computer printout purposes, the first digit(s) of the number shall be selected to indicate the assembly. The use of special characters such as periods or dashes will be avoided.

4.1.2 (C1) ASSEMBLY NAME: Enter the name of the assembly.

4.1.3 (C1, J1) ITEM NOMENCLATURE: Enter the nomenclature for the component. In the first block (C2), give the basic identifying noun. Enter any additional modifiers or description on the J1 line. A typical example is "Valve, Solenoid", where "valve" is the basic identifier.

Table 4.0 - IDENTIFIERS & SUBSYSTEM NAMES

01-5 PURGE, VENT & DRAIN

02-1 LANDING DECELERATION

02-2 DOCKING MECHANISM

02-3 SEPARATION MECHANISM

ACTUATION MECHANISMS

02-4 DOORS

ET Umbil door
Star Tracker
Air Data Sensor

02-4A HATCHES

02-4B PAYLOAD BAY DOORS

02-4C RUDDER/SPEEDBRAKE, BODY FLAP

02-5 PAYLOAD RETENTION/DEPLOYMENT MECHANISMS

02-6 HYDRAULICS

03-1 MAIN PROPULSION

03-2 REACTION CONTROL
03-2A AFT

03-2B FORWARD

03-3 ORBITAL MANEUVER

04-1 ELECTRICAL POWER - CYRO

04-1A ELECTRICAL POWER - FUEL CELL

04-2 AUXILIARY POWER (APU)

05-1 GUIDANCE, NAVIGATION & CONTROL

COMMUNICATIONS & TRACKING

05-2A AUDIO
05-2B UHF
05-2C TACAN
05-2D ALTIMETER
05-2F MICROWAVE SCAN BEAM LANDING (MSBLS)
05-2G S-BAND
05-2J PAYLOAD INTERRAGATOR
05-2K CLOSED CIRCUIT TV (TV)
05-2R KU-BAND COMM & RADAR

05-3 DISPLAYS & CONTROLS

05-4 INSTRUMENTATION

05-5 DATA PROCESSING & SOFTWARE & COMPUTERS

*05-6 ELECTRICAL POWER DISTRIBUTION & CONTROL

05-8 BACKUP FLIGHT CONTROL

06-1 ATMOSPHERIC REVITALIZATION (ARS, ARPCS, Airlock)

06-2 LIFE SUPPORT

06-3 ACTIVE THERMAL CONTROL & WATER SPRAY BOILER

07-1 CREW PROVISIONS, ACCOMMODATIONS & EMERGENCY EGRESS

07-2 CREW ESCAPE - 102 PRE-AAMOD ONLY

07-3 TUNNEL ADAPTER

*See TABLE 5.0 for EPD&C/INTERFACING SUBSYSTEM IDENTIFIERS

Table 5.0 - EPD&C/INTERFACING SUBSYSTEM IDENTIFIERS

ELECTRICAL	INTERFACE	MECHANICAL SUBSYSTEMS
05-6AA	01-5	Purge, Vent & Drain
05-6AB	01-5	Vent Doors
05-6B	02-1	Landing Deceleration
05-6BA	02-1	Landing Gear Control
05-6BB	02-1	Brake & Anti-Skid
05-6BC	02-1	Nosewheel Steering
05-6C	02-2	Docking Mechanism
05-6D	02-3	Separation
05-6DA	02-3	Carrier A/C Separation
		ACTUATION MECHANISMS SUBSYSTEMS
05-6EA	02-4A	Hatches
05-6EB	02-4B	Payload Bay Door
05-6EC	02-4C	Rudder/Speedbrake, Body Flap
05-6ED	02-4	ET Umbilical Doors
05-6EE	02-4	ADP Deploy & Htr
05-6EF	02-4	Star Tracker Doors
05-6EG	02-4	Freon Radiator Deploy
05-6EH	02-4	Rendezvous Radar & Comm. Antenna Deploy
05-6F	02-5	Payload Retention, Manipulator Positioning
05-6G	02-6	Hydraulics
05-6IA	02-5	Remote Manipulator Arm
05-6IB	02-5	Manipulator Deploy Control
05-6IC	02-5	Manipulator Latch Control
05-6ID	02-5	Manipulator Arm Shoulder Jettison & Retention Arm Jettison
05-6IE	02-5	DAC Camera-PLB OPS
		PROPULSION SUBSYSTEMS
05-6J	03-1	Main Propulsion
05-6KA	03-2A	Reaction Control-Aft
05-6KF	03-2F	Reaction Control-Fwd
05-6L	03-3	Orbital Maneuvering
05-6LA	03-3	OMS Auxiliary Kit
		POWER GENERATION SUBSYSTEMS
05-6MA	04-1A	Electrical Power Generation - Fuel Cell
05-6MB	04-1	Electrical Power Generation - Cyro
05-6N	04-2	Auxiliary Power Unit

Table 5.0 - (Cont'd.)

ELECTRICAL	INTERFACE	AVIONICS SUBSYSTEMS
05-6O	05-1	Guidance, Navigation & Control
		Communications & Tracking:
05-6PA	05-2A	Audio
05-6PB	05-2B	UHF
05-6PC	05-2C	TACAN
05-6PD	05-2D	Altimeter
05-6PF	05-2F	Microwave Scan Beam Landing (MSBLS)
05-6PG	05-2G	S-Band
05-6PJ	05-2J	Payload Interragator
05-6PK	05-2K	Closed Circuit TV (TV)
05-6PH		Ground Command Interface Logic (GCIL)
05-6PR	05-2R	Ku-Band Comm. & Radar
05-6Q	05-3	Displays & Controls
05-6R	05-4	Instrumentation
05-6S	05-5	Data Processing & Software
05-6T	05-8	Backup Flight Control
05-6	05-6	Electrical Power Distribution & Control
		ECLSS SUBSYSTEM
05-6U	06-1	Atmospheric Revitalizatin - ARS, ARPCS
05-6UA	06-1	Airlock Environmental Control
05-6V	06-2	Smoke Detection, Fire Suppression
05-6VA	06-1	ARPCS
05-6VB	06-2	Galley
05-6VC	06-2	Waste Management
05-6VD	06-2	Water Management
05-6W	06-3	Active Thermal Control
05-6Y	07-1	Crew Station & Equipment
05-6Z	07-2	Crew Escape

4.1.4 (J10) FUNCTION: Describe the function performed by the component. Also, enter the component designator(s) as identified on the design schematic.

4.1.5 (E4, C5, C6) QUANTITY: Enter the total number of items having identical part numbers performing the same function in the subsystem. The E4 field will reflect the total quantity in Arabic numerals. The C5 and C6 fields will reflect written quantities.

4.1.6 (C7, C8) PART NUMBER:

a. C7) ROCKWELL PART NUMBER: Enter the appropriate Rockwell part number in accordance with the following DRM/SRM example, starting at the most left-hand-position -
   (1) V070-XXXXXX (Airborne, In-House)
   (2) MEXXX-XXXX (SCD)
   (3) MCXXX-XXXX (Procurement Spec)
   Note: Dash numbers to basic part numbers are required when the basic part number has dash numbers having differences in the failure mode and effects.
b. (C8) SUPPLIER PART NUMBER & SUPPLIER NAME: Enter supplier part number and supplier name (abbreviate if required) when available for ME and MC part numbers.

4.1.7 (C11-14) REFERENCE DOCUMENTS: Enter the referenced schematic diagram first, followed by the related block diagrams, logic diagrams, etc.

4.1.8 (C9, C10) FMEA PREPARED BY: Enter the initials and last name of the Reliability Subsystem Analyst and Design Engineer who prepared the subsystem FMEA.

DATA SHEET NO. 2

4.1.9 (DI, LV1, LV1) DATA IDENTIFIERS: Indicate appropriate action and specify the subsystem and component ID as described for Data Sheet No. 1.

4.1.10 (LV3) FAILURE MODE SEQUENCE: Assign different sequence numbers (e.g., 1, 2 - 5) for various failure modes of the specified component.
Do not use leading zero's.
Example: (LV3) _ _ _ 1, NOT (LV3) 0 0 0 1.

4.1.11 (C31, J130, C32) FAILURE MODE: Enter first the basic failure mode (keyword) (C31), then any additional modifiers (J130) necessary to fully describe the specific failure mode - the exact manner in which the item fails. Failure mode keyword identifiers are listed below. Selection should include but not be limited to those listed.

FAILURE MODE KEYWORD IDENTIFIERS

• STRUCTURAL FAILURE (RUPTURE)	• INADVERTENT OPERATION	• PREMATURE OPERATION
• PHYSICAL BINDING/JAMMING	• INTERMITTENT OPERATION	• DELAYED OPERATION
• FAILS TO REMAIN OPEN/CLOSED	• ERRATIC OPERATION	• ERRONEOUS OUTPUT
• FAILS MID-TRAVEL	• ERRONEOUS INDICATION	• LOSS OF OR PARTIAL OUTPUT
• FAILS TO OPEN/CLOSE	• RESTRICTED FLOW	• SHORTED
• INTERNAL/EXTERNAL LEAKAGE	• FAILS TO START/STOP	• OPEN (ELECTRICAL)
• FAILS OUT OF TOLERANCE	• FAILS TO SWITCH	• LEAKAGE (ELECTRICAL)

Appendix B, paragraph 3.1.1, sub-paragraph 13, reflects the ground rule to be used for external leakage. For OV-102 pre-AA mod only, those failure modes which result in a criticality classification of 1 and 2, or 1R and 2R, and appear in the CIL (item 4.1.22) shall be classified further as structural or functional failures by circling "S" or "F" in the C32 field. The following guidelines apply:

STRUCTURAL (S) - A failure mode involving structural failure of a pressure vessel, component housing, fluid lines, attach fittings, or load-carrying members such as cranks or rods.

FUNCTIONAL (F) - A failure mode, generally within a component, which negates the described component function. This type of failure would include binding, leakage, failure to open or close, or loss of output. The failure cause could be improper installation of parts or structural failure of power transmitting parts such as gear teeth, shafts or springs; however, in such instances the mode is still classified as functional. Electrical and electronic component failures would normally fall in this category.

4.1.12 OV-102 PRE-AA MOD ONLY:

(C62-66) APPLICABLE MISSIONS: Enter an "X" in the block of the mission to which the FMEA applies.

Horizontal Flight Test	C62
Vertical Flight Test	C63
Ferry Flight	C64
Operational Flights	C65
Specific Orbital Mission	C66

Note: "Operational Flights" and "Specific Orbital Mission" are not to be used for the duration of OV-101 and OV-102 flight test programs.

OPERATIONAL VEHICLE(S):

(C83 - C86) VEHICLE EFFECTIVITY: Enter an "X" in the appropriate block(s) to which the FMEA appplies.

Orbiter Vehicle	102	C83
Orbiter Vehicle	099	C84
Orbiter Vehicle	103	C85
Orbiter Vehicle	104 & SUBS	C86

4.1.13 (C33-37) MISSION PHASE(S): Enter an "X" in appropriate box(es) to indicate when the specified effects would be manifested. If the failure occurs at discrete points in time within a given mission phase, and different effects may be observed, it may be necessary to define the subphase or event under "EFFECTS".

4.1.14 (C38, C58) ABORT CRITICAL COMPONENTS:

a. For those items whose criticality is increased to 1/1 during an abort resulting from unrelated failures, enter the word "Abort" (C38 - six spaces only), followed by the appropriate acronym(s); i.e.,
   - (C58) RTLS - Return to Landing Site
   - (C58) AOA - Abort Once Around
   - (C58) ATO - Abort to Orbit

b. For non-redundant modes where normal mission effect is criticality 3 but are hardware criticality 1 unique to intact abort, classify these modes as hardware criticality 1 and functional criticality 1. Add in J10 (FUNCTION) the notation, "Unique to Intact Abort". Add appropriate intact abort notation in a. above.

Additional information must also be entered under J240, EFFECT(S) - see paragraph 4.1.16. NOTE: For SSME induced aborts, maximum two engine burn time is approximately twelve minutes. If "TIME TO EFFECT" is equal to or greater than twelve minutes, there is no change in criticality.

4.1.15 (J380) CAUSE(S): Enter causes including but not limited to those listed below and amplify as necessary. See paragraph 4.4.2 for instructions on supplier furnished piece parts.

CAUSES

• CONTAMINATION	• TEMPERATURE (HIGH/LOW)	• INADVERTENT OPERATION/ACTIVATION
• MECHANICAL SHOCK	• THERMAL SHOCK	• VIBRATION
• VACUUM	• PRESSURE (HIGH/LOW)	• PROCEDURAL ERROR
• ACOUSTICS	• IONIZING RADIATION	• CHEMICAL REACTION
• OVERLOAD	• ACCELERATION	• LOSS OF/IMPROPER INPUT
• MISHANDLING OR ABUSE	• ELECTROMAGNETIC FIELDS	• PIECE-PART STRUCTURAL FAILURE

4.1.16 (J240) EFFECT(S): Enter the letters (A), (B), (C) or (D) as defined in the headings of Appendix A, together with the words under each heading describing the effects on the subsystem, interfaces, mission, and crew/vehicle, respectively, and explain. If the identified effect is not listed, describe briefly. Where the effect is the same for two or more of the above, consolidate entries. Specify if there is no effect on a specific category or categories and provide a brief explanation. In those instances when time to abort requires automatic operation or immediate dependence on a parallel subsystem and such is provided, the effect on mission is "None" with explanation for each mission phase as appropriate. See section 4.1.21d. for screening of functional criticality 3 failure modes. For those items identified as abort critical (see paragraph 4.1.14) enter, subsequent to the (A), (B), (C) and (D) entries, the criticality and effects per the following example:

"Crit 1 for RTLS - Loss of additional engine-vehicle loss"

or

"Crit 1 for RTLS - Incomplete propellant dump, stability problem, probable vehicle loss."

Where functional criticality is 1R or 2R per paragraph 4.3.2 and hardware criticality is 3, the appropriate entry for "FUNCTIONAL" effects should be included. The "FUNCTONAL" effects entry relative to the loss of all functional redundancy will be entered per the following example:

(E) FUNCTIONAL CRITICALITY EFFECT:

Possible loss of crew/vehicle (specify) or probable loss of crew/vehicle (describe) or loss of crew/vehicle.

4.1.17 (C39) TIME TO EFFECT:

Immediate	-	less than 1 second
Seconds	-	1 to 50 seconds
Minutes	-	50 seconds to 50 minutes
Hours	-	50 minutes to 20 hours
Days	-	20 hours to mission completion

Enter the descriptor which indicates shortest credible time or time range available to correct the situation before the effect is manifested.

4.1.18 (C40-45) FAILURE DETECTABLE: Enter "YES" or "NO" in the block following "IN FLIGHT" and "GROUND TURNAROUND". If either answer is "YES", indicate how it can be detectable -- symptoms, instrumentation, etc. Include measurement number from MML (Master Measurements List) where applicable and available. (See section 4.3.5, Instrumentation FMEA's.) Development flight instrumentation (DFI) measurements will not be used as a means of detectability.

4.1.19 (J490) CORRECTING ACTION: Describe any action, automatic or manual, which may be taken to circumvent the specified failure. Also identify any alternate means (utilizing "unlike" hardware) of accomplishing the function performed by the item or its assembly. If none, so indicate. For instruments (sensors, transducers, etc.) that provide measurements assessed as critical to vehicle/crew safety or mission continuation, the FMEA shall identify the redundant or alternate measurements by Measurement List identification number.

4.1.20 (C77) NUMBER OF SUCCESS PATHS REMAINING AFTER FIRST FAILURE: With respect to the item being evaluated, indicate the number of ways remaining to perform the function after the first failure. You may leave the block blank for non-critical functions.

4.1.21 (C53, C55-57) REDUNDANCY SCREEN: For all criticality 1R, 2 and 2R failure modes (see FIGURE 8 and paragraphs 4.3.1 and 4.3.2) circle "P" (PASS), "F" (FAIL), or "NA" (NOT APPLICABLE) for each of the following tests:

a. Redundant elements are capable of checkout during normal ground turnaround with no vehicle design modification. Where a subsystem is characterized by redundant strings and the status of each string can be verified during ground turnaround, no individual component(s) in any one string should be shown as failing this screen.

   NOTE: This screen is not applicable under the following conditions:
   (1) Pyrotechnic devices, excluding electrical control circuitry.
   (2) Non-redundant item.

b. Loss of a redundant element is readily detectable during flight. Where a subsystem is characterized by redundant strings and the status of each string can be verified in flight, no individual component(s) in any one string should be shown as failing this screen.

   NOTE: This screen is not applicable under the following conditions:
   (1) Standby redundancy (redundant paths were only one path is operational at any given time).
   (2) All functional paths of any subsystem which is inoperative (during such inoperative periods). This groundrule does not apply if the redundant elements are operative during any normal mission phase; i.e., the screen is considered applicable if the element is operative during any normal mission phase.
   (3) Pyrotechnic devices.
   (4) Mechanical linkage.
   (5) Non-redundant item.
   (6) Subtier level redundant functional path(s) (power/control circuits, etc., failures where the primary functional path (LRU, etc.) is

Criticality 1R3 or 2R3 and the primary redundancy would not be degraded (i.e., loss of two of the subtier functional paths would not result in an abort decision).

c. Failure of an element to pass this screen should be in direct relation to the noted failure mode under normally expected environmental conditions. Consideration of environmental extremes as caused which could induce "multiple order failure" is limited to abnormal conditions generally resulting from some other failure. Where multiple failures must first occur to result in environmental extremes, such events may be considered non-credible. As a ground rule, it may be assumed that hardware items will be qualified and properly installed to withstand the "design-to" environmental envelope. The following are typical questions to be answered in this phase of the analysis:

(1) Contamination:
   (a) Are the items being evaluated susceptible to contamination?
   (b) Is contamination a credible event or does the design (including filters) result in this failure mode being categorized as non-credible?

(2) Explosion:
   (a) Is there a credible source?
   (b) Must other multiple failures occur first to result in the explosion?
   (c) Is the explosion catastrophic to crew or vehicle?
   (d) Is the container frangible?
   (e) Are the items being considered susceptible to this type of damage in view of their physical characteristics and location; i.e., shielding?

3. Temperature:
   (a) Are components susceptible to damage or failure from high temperature?
   (b) Other than as a result of multiple failures, is such exposure credible? This implies temperature peaks or sustained levels sufficient to cause catastrophic effects on the component in a short time. For example, temperature increases to certain levels merely increase electronic parts failure rates - the actual failure and time of occurrence are still probabilistic.

(4) Vibration, Shock, Acceleration, Acoustics, etc.:
(a) Assuming that components are qualified and properly installed to withstand design environments, can a credible cause be identified which would cause these levels to be exceeded?
(b) Are vibration/shock/acceleration-sensitive redundant units physically oriented or separated to reduce the chance of multiple failure from the same cause(s) and is there sufficient analysis and test data to verify the failure as non-credible?

(5) Fire:
Do not consider fire as one of the single events or causes in failing screen "C". NASA has edicted that fire not be considered one of the events (NB/83-L 216).

If none of the redundancy screens are applicable, enter "NA" in the C53 field and briefly explain reason for REMARKS/HAZARDS.

d. Screening of Functional Criticality 3 Failure Modes
(1) Where the failure modes have been identified as non-critical for loss of all redundancy (Criticality 3), enter "NA" in the C53 field. Enter under "REMARKS" the notation, "Criticality 3 failure mode - loss of all redundancy would have no effect on the mission or crew/vehicle safety". In such cases, minimum entries on Data Sheet No. 2 consist of DI, LV1, LV2, LV3, C31, C53 and J240. For functional Criticality 3 items, J240 must contain a brief explanation regarding the assigned criticality.

(2) Where a component has an identified failure mode in the Criticality 1 of 2 category, and additional functional Criticality 3 failure modes are identified, these Criticality 3 modes will be treated as described in para. (1) above.

4.1.22 CRITICALITY:
a. (C54 - HARDWARE) - Enter 1, 2 or 3 based on the definitions in section 2.0 and the ground rules contained in section 4.3.1 and Appendix B, paragraph 3.1.1, sub-paragraph 1.
b. (C67 - FUNCTIONAL) - Enter 1, 2, 1R, 2R or 3 based on the definitions in section 2.0 and the ground rules contained in section 4.3.2 and Appendix B, paragraph 3.1.1, sub-paragraph 1.

4.1.23 (J500) REMARKS/HAZARDS: Identify potential hazards resulting from the specified failure. Enter the words "Hazard Potential" followed by appropriate explanation and any other comments or recommendations that might prove useful in evaluating the system. Indicate requirements for additional instrumentation, and any other special consideration.

4.1.24 (J600) DISPOSITION AND RATIONALE: For criticality 1 and 2 items, and/or 1R, 2R items that fail a redundancy screen and/or hardware criticality 2 items where the screen is NA, in all of the following categories to describe the retention criteria. Each category must reflect a description of rationale for retention of the item:

a. Design - Identification of design features which minimize the occurrence of the failure mode and causes.
b. Test - identification of specific tests accomplished to detect failure mode and causes during acceptance tests, certification tests, and checkout tests.
c. Inspection - Statement that specific inspection points are included to determine that specific failure mode causes are not inadvertently manufactured into the hardware.
d. Failure History - Provide an indication that the hardware or similar hardware has been used successfully and that a history of generic failures does not exist. If the hardware is new to this program, so state.

4.1.25 (C9, C10) APPROVAL: Responsible Reliability and Design approval signatures as follows:

a. Subsystem FMEA package: Design/Reliability Manager

b.	Figure 2 (FMEA) - PREPARED BY:	Design ______	Responsible
		Reliability ______	Analyst's Name
	APPROVED BY:	Design ______	Signature
		Reliability (Analyst)	Signature
c.	Figure 3 (CIL) - APPROVED BY:	Design (Supervisor)	Signature
		Reliability (Supervisor)	Signature

NOTE: The initial issue of a CIL sheet will be signed by the Reliability Supervisor. Signatures will not be required on subsequent issues unless the CIL sheet is revised.

0246I/10

4.2 REVISIONS & SUBMITTALS

1. Revisions to the FMEA will be made as follows:

   a. New Data:

      (1) To identify new components or failure modes, use the data entry sheets and follow the instruction given in section 4.1.

      (2) To add information to a component or failure mode record, either a blank data sheet or the appropriate page of the data printout working copy may be used.

         (a) Data Entry Sheets - Using a blank data sheet, circle "R" (Revise) on the "Data Identifier" line (DI, FIGURE 4 or 5) and enter the correct subsystem/component/(failure mode) ID number to identify the record to which the information is to be added. Fill in complete blocks of information to be added (e.g., Disposition block), and submit for keypunching.

         (b) Data Printout - Circle "R" (Revise) on the "Data Identifier" line (DI, FIGURES 6 and 7) of the record to which new information is to be added. Using a colored pen or pencil, enter the information in the appropriate blocks and submit for keypunch.

   b. Data Entry Change:

      Circle the "R" (Revise) on the "Data Idenifier" line (DI, FIGURE 4 or 5) and either "red-line" the appropriate sheet of the data printout or re-enter the data as it should appear, using the appropriate data entry sheet as described in section 4.2, paragraph

      (a) Data Entry Sheets. To clear the "J" field of any remaining unwanted information, asterisk (*) the blank lines within the block on the master record and supporting record work sheets.

   c. Data Deletion:

      (1) To delete data, circle the "R" on the "Data Identifier" line (DI, FIGURE 6 or 7) of the appropriate data printout sheet, cross out the entry to be deleted with a colored pen or pencil

and submit for keypunching. If a blank data sheet is used, enter an asterisk (*) in the block which corresponds to the entry to be deleted.

(2) To delete the entire record (i.e., all data pertaining to a particular failure mode or component) and all related entries, circle the "D" (Delete) on the appropriate sheets of the data printout. Again, filling out the data identifier line of a blank data sheet will accomplish the same purpose. All information pertaining to a particular component or failure mode will be deleted.

d. Data Identifier Change:
To change a data identifier (LV1, 2 or 3), it is necessary to delete the entire record under the old number and re-entrer (add) under a new number. The B999/revision data on computer reports is automated and prints the date of the latest update or revision.

e. Identification of Revisions/Changes:
Identify each line changed with a vertical black bar on the left-hand margin of the page.

2. FMEA/CIL Submittal

a. Critical Items List (CIL)
Updates will include the following:
(1) Any new CIL items
(2) Updates to existing items having technical changes affecting the following sections:
(a) function
(b) failure mode
(c) failure effects
(d) criticality
(e) abort critical components
(f) failure detectability (redundancy screen)
Other changes will be incorporated when pages are submitted for the above reasons.

b. FMEA's
Updates of the FMEA's will be at six month intervals linked to nearest scheduled CIL FRR publication. Only those changes related to the CIL submittals (a above) and other technical changes will be submitted.

4.3 IMPLEMENTATION GROUND RULES (See also Appendix B - Ground Rules and Criteria)

4.3.1 HARDWARE CRITICALITY DETERMINATION

Hardware criticality will be determined by the categorization of the singular effect of the identified failure mode on the subsystem/vehicle (See FIGURE 10). FIGURE 8 illustrates the analytical logic for criticality determination of all functional hardware.

1. Reliability Engineering identifies hardware where if redundancy fails the effect would be critical.

2. Reliability and Design Engineering jointly identify those equipments with (single point) criticality 1 or 2 failure modes. Those equipments that are not criticality 1 because they incorporate redundancy are then screened further, as described in paragraph 4.1.21, and appropriate entries made in the FMEA data sheet.

   NOTE: The criticality of instrumentation and test ports will be assessed according to their function. Test ports, when capped, shall be treated as a structural part of the component and not be considered further. Where instrumentation (e.g., pressure transducer) penetrates the wall of a component or line and structural failure of the joint would result in gross leakage, the failure mode shall be considered as a failure of the component or line. The criticality of the instrumentation, therefore, would not be affected in such instances.

3. The criticality of those systems which are to be used only in the event of an emergency shall be established strictly on the basis of direct failure effect on crew, vehicle, or mission, regardless of the number of prior failures which must occur before the use of the system is required.

All other backup or standby equipment (e.g., relief valves, cross-feed valves, etc.) shall be assigned criticality in the normal manner.

4.3.2 FUNCTIONAL CRITICALITY DETERMINATION

Functional criticality will be determined by the categorization of the effect on the subsystem/vehicle of loss of all redundancy (like or unlike) for the identified failure mode (See FIGURE 10). FIGURE 8 illustrates the analytical logic for criticality determination of all functional hardware.

1. Reliability Engineering identifies hardware if all like or unlike redundancy fails the effect would be critical.

2. Reliability and Design Engineering jointly identify those equipments with criticality 1R or 2R failure modes.

4.3.3 CIL CONTENT CRITERIA

1. The following classification of failure modes will be entered in the CIL:
   a. All functional/hardware criticality category 1/1's, 2/2's, and 1R2's.
   b. All criticality category 1R3's and 2R3's that fail one or more redundancy screens.
   c. All failure modes that become criticality category 1/1 during intact abort.

2. CIL Section 12.0 - Critical Items List orbiter modifications to support special missions:

   This section of the Critical Items List contains those critical items associated with Orbiter subsystems that have been added to or modified by Orbiter Mission Kits to support special missions. These CIL items will only apply to specific vehicle missions as noted in this specific CIL subsection.

   This CIL section contains the single failure points and criticality 1R and 2R CIL items identified by the Failure Mode Effects Analysis (FMEA) conducted on the Orbiter subsystems that have been added to or modified to support special mission application. These vehicle changes are identified by individual Mission Kits which are installed specifically for these special missions and would be removed when the mission objectives have been achieved. Each vehicle change is identified by a Master Change Record (MCR) and is referenced in the applicable FMEA.

Each critical failure mode identified in the vehicle modification section is categorized on a separate Critical Items List form which includes the failure causes, effects, and rationale for retention. CIL dispositions and rationale are contained on individual CIL sheets and those that are generally applicable to all components are contained in Section 3.0.

A critical items list summary is included for each major vehicle modification. Additions will be made to this CIL section to maintain this document current with the vehicle flight configuration. CIL page revisions are indicated by revision date.

NOTE: Prior to each CIL submittal, notify the CIL coordinator of any input to CIL Section 12.0.

4.3.4 MECHANICAL/ELECTRICAL INTERFACE

For mechanical components having an electrical interface, the mechanical FMEA will consider only the effects of "black box" functional failure (e.g., loss of output, premature signal, etc.). Where it becomes necessary to conduct an FMEA within the "black box" because of the assigned criticality (1 or 2), the FMEA will be conducted by Avionics Reliability who will be provided with the following information in the mechanical FMEA regarding the failure effects on the mechanical system:

a. Enter under "CAUSE" (J380), each applicable failure mode of the electromechanical device reflecting the avionics malfunction causing the failure mode; i.e., loss of electrical power, premature electrical signal, etc.

b. Identify under "REFERENCE DOCUMENTS" (C11-14), the specific mechanical/avionics interface.

The electrical interface FMEA will be included in the appropriate section of the avionics FMEA. For criticality 1, 1R, 2 or 2R failure modes, the mechanical FMEA, which considers the effect of "black box" loss of function, will indicate in the "REMARKS" section the avioiics FMEA number of the "black box". The avionics FMEA of the "black box" will contain a similar reference to the appropriate mechanical FMEA. At the earliest point in time when the mechanical analyst can ascertain that the "black box" is criticality 1, 1R, 2 or 2R, it shall be his responsibility to convey to the Avionics Reliability group copies of his worksheets to facilitate initiation of detailed avionics analysis effort.

4.3.5 INSTRUMENTATION FMEA'S

Instrumentation (e.g., sensors, signal conditioners, etc.) may be provided by either Avionics Instrumentation or by a specific Design group. In either case, instrumentation FMEA's will be included in the FMEA for the using subsystem. Criticality 3 instrumentation may be listed on one FMEA form by family or type. FMEA's for criticality 1 or 2 and criticality 1R and 2R instrumentation that fail a redundancy screen or the screen is "NA", will be completed in their entirety and included in the using subsystem CIL. Avionics Reliability will provide support as required to identify failure modes, retention rationale, etc.

A copy of each instrumentation FMEA completed by a Mechanical Reliability group will be provided to Avionics Reliability.

4.4 SUPPLIER FMEA UTILIZATION

4.4.1 GENERAL

In many instances, depending on the cost, complexity, and state of development of the design, suppliers will be required to develop and submit FMEA's reflecting their area of design responsibility. The submissions will precede the joint supplier/Rockwell PDR or CDR. (See the applicable PDRD for content requiremennts and submittal schedules.) FIGURE 9 shows the overall supplier FMEA flow as related to the in-house effort.

4.4.2 FMEA UTILIZATION

Upon receipt of a supplier FMEA, the responsible RSA will compare the identified failure modes with those called out on his corresponding subsystem FMEA and update his FMEA as required (see section 4.2) to include any failure modes not already identified relating to subsystem effect. Supplier FMEA's will be reviewed for single failure points below the black box level by Rockwell and analyzed for corrective action directly with the supplier as part of their design review. Where Rockwell does not concur with portions of the supplier analysis, telephone contact with the supplier Reliability Engineer normally should suffice to resolve any differences. If not, the matter shall be resolved through normal Rockwell data handling procedures. Supplier black boxes will be identified in the subsystem FMEA based on the supplier schematic or drawing part identification number. For criticality 1 or 2 and 1R or 2R (CIL only) electronic black boxes, the piece parts (or if all or many circuits, so state) identified by the supplier FMEA which are single point failures that have a direct critical effect on the vehicle will be described with reference to the supplier FMEA in the "CAUSE" section of the applicable failure mode identified at the subsystem level. Parts will be listed only in those cases where less than five parts are involved.

4.5 ELEMENT CONTRACTOR FMEA CORRELATION

Requirements and procedures for conducting interfacing analyses and for element integration tasks are contained in Reliability Desk Instruction 100-12 (Shuttle Element Interface).

4.6 GFE

For items identified as GFE hardware, NASA will identify those which require FMEA's and will perform FMEA's on the hardware identified to the level defined by their ground rules. Upon completion of the FMEA, NASA will provide Rockwell with a copy. In addition to the completed copy, a preliminary copy may be transmitted. Upon receipt of the GFE FMEA, Rockwell will evaluate the interface effects on the Orbiter defined by the GFE FMEA.

Appropriate comments shall be included to insure that this area is correct and complete. Rockwell will conduct FMEA's for all interfaces between CFE and GFE. The Rockwell FMEA will consider all failure modes consistent with this desk instruction. The analysis is to consider as a "CAUSE" any failure mode identified by the GFE FMEA which could produce a failure in the CFE interface. Where GFE failures are identified as a "CAUSE", the appropriate GFE FMEA and document number shall be identified as a part of the "CAUSE" section. In addition, those vehicle failures which could cause FMEA failure modes will be identified to NASA in the comments to the GFE FMEA. Where structural failures are identified, appropriate hazards analyses shall be included in available.

The accountability of CIL items for GFE will be NASA. Those CIL items resulting from interface failure modes will be a part of the Rockwell CIL.

Exceptions to this instruction will be identified and concurred in jointly by Rockwell and NASA and documented as a part of letters of agreement.

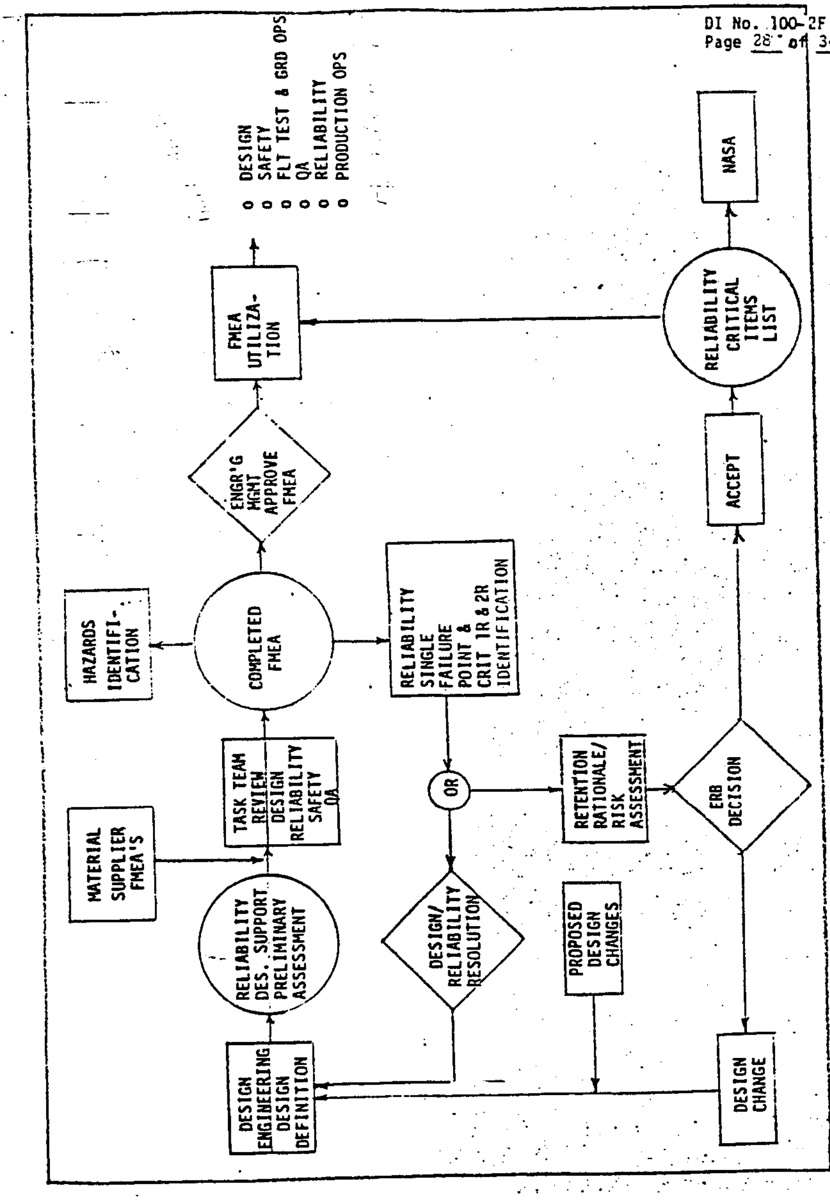


FIGURE 1 - FMEA PROCESS

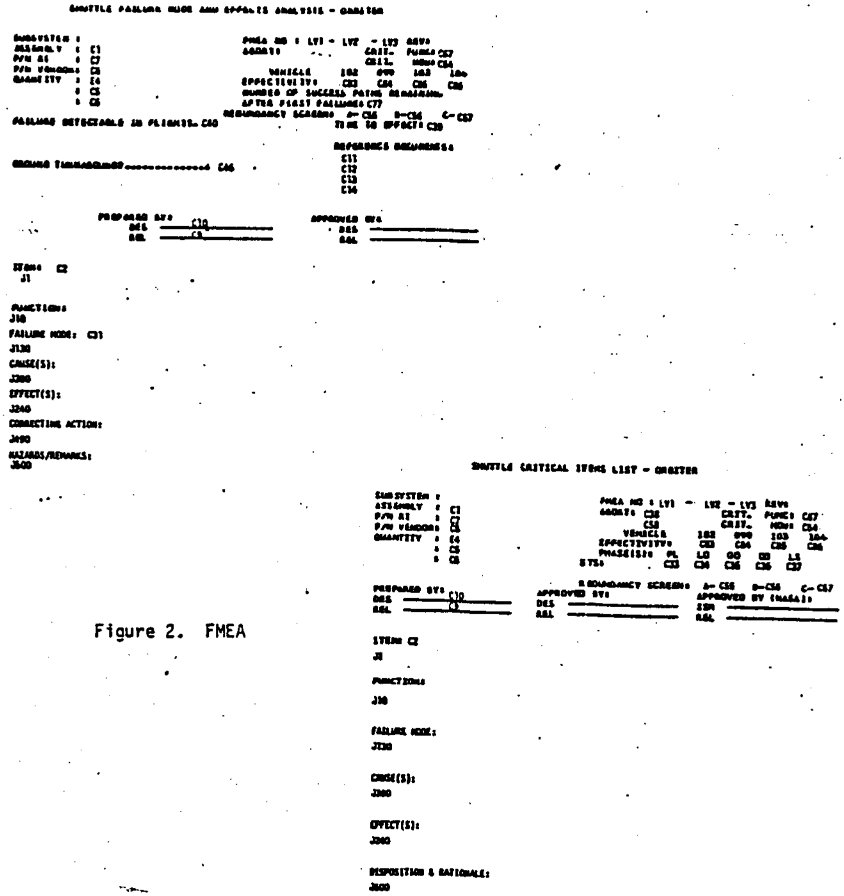


Figure 2. FMEA

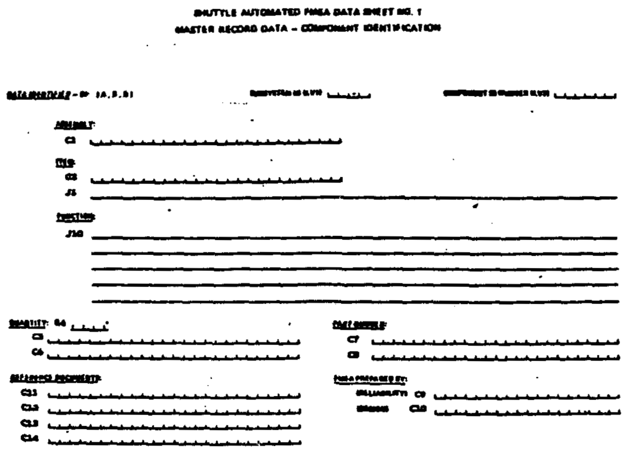


Figure 4. Data Sheet No. 1

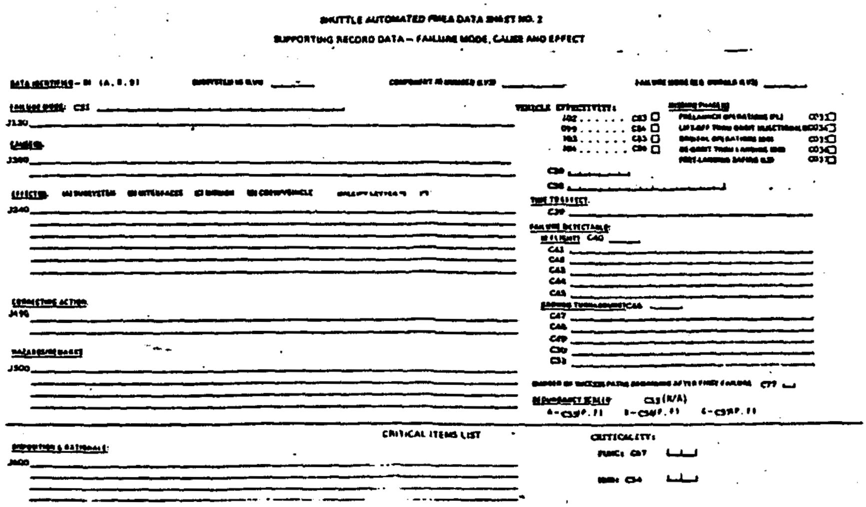


Figure 5. Data Sheet No. 2

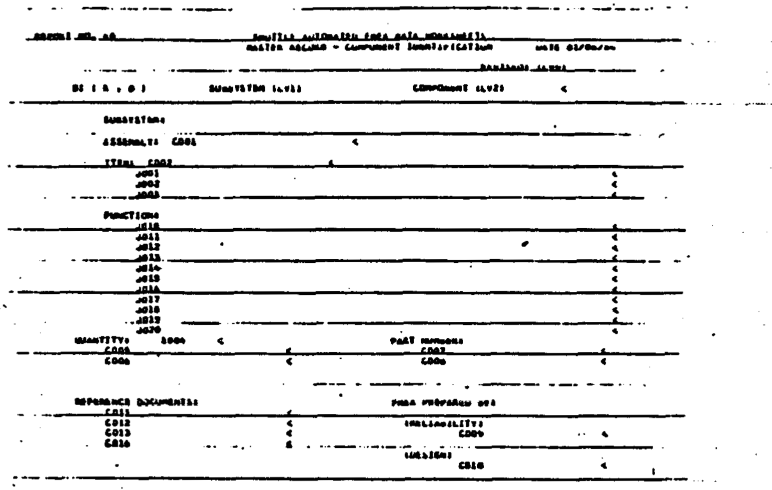


Figure 6. Data Printout, Component-Related Data

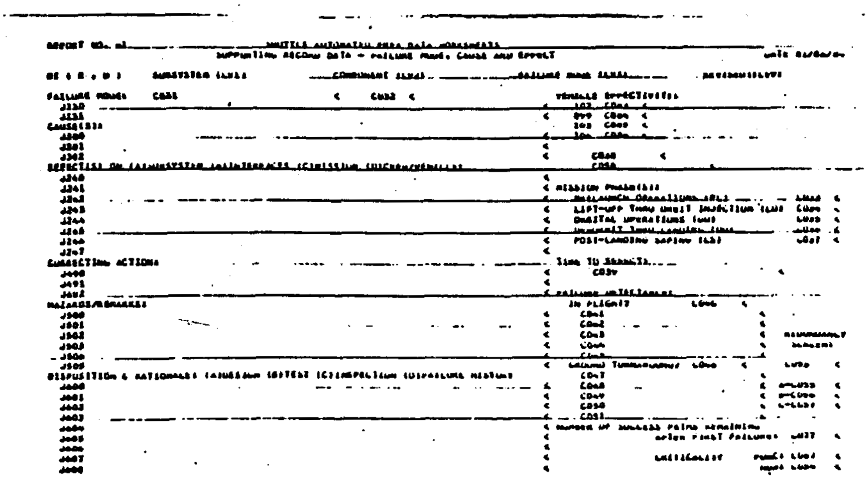


Figure 7. Data Printout, Failure Mode-Related Data

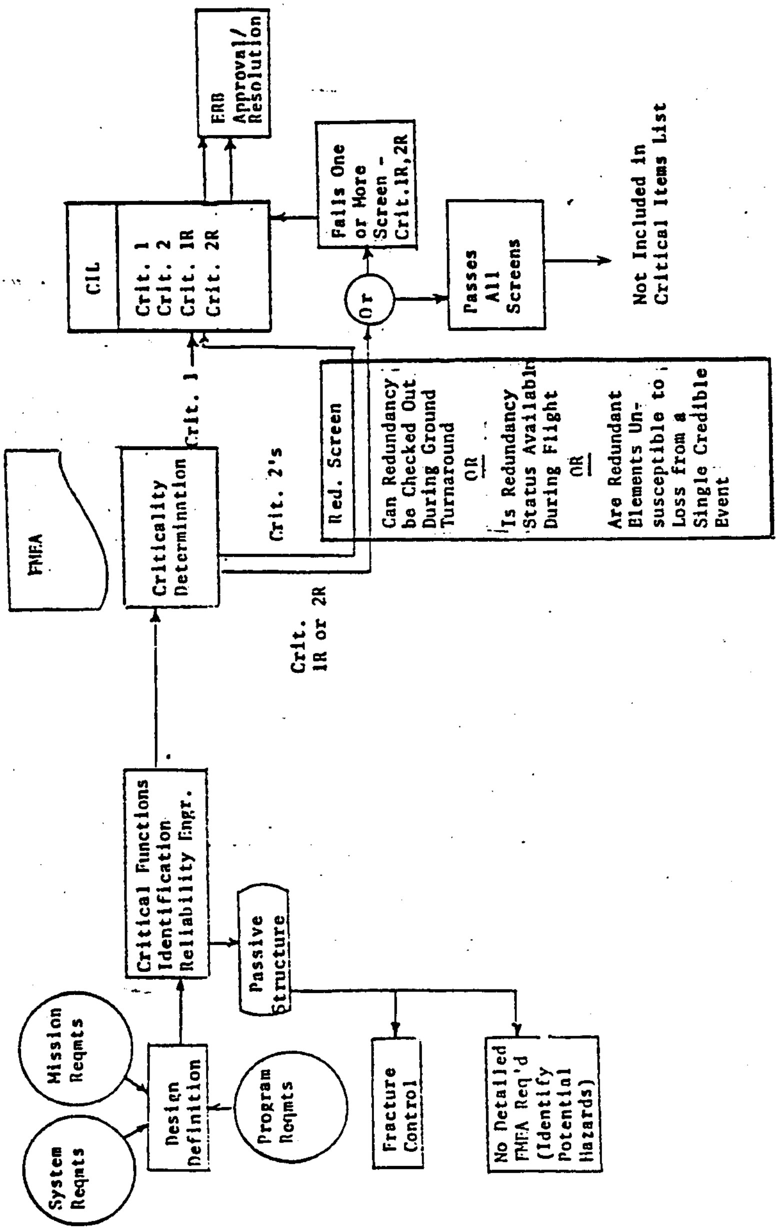


FIGURE 8 - FMEA SCREENING

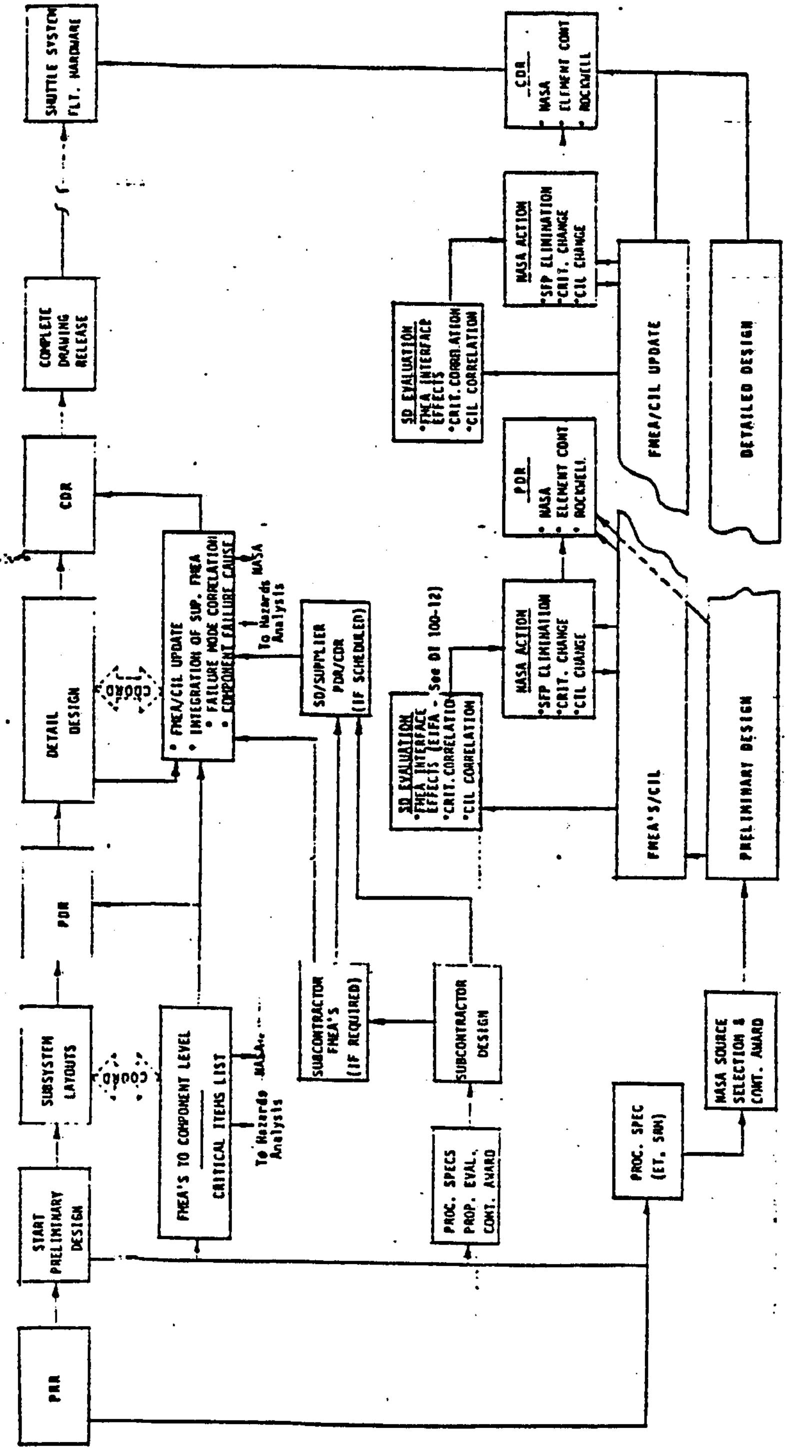


Figure 9 - FMEA FLOW

CRITICALITY CATEGORY
CROSS REFERENCE TABLE

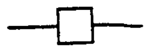
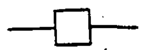
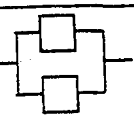
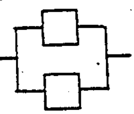
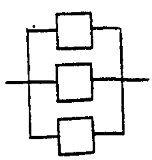
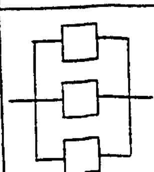
FUNCTION / LEVEL OF REDUNDANCY	BLOCK DIAGRAM	CRITICALITY CATEGORY: FUNCTIONAL DEFINITIONS		CRITICALITY CATEGORY: HARDWARE DEFINITIONS
LIFE OR VEHICLE ESSENTIAL / NO REDUNDANCY		1 (CIL)		1 (CIL)
MISSION ESSENTIAL / NO REDUNDANCY		2 (CIL)		2 (CIL)
LIFE OR VEHICLE ESSENTIAL / DUAL REDUNDANCY		1R (CIL)		2 (CIL)
		PASSED SCREEN	FAILED SCREEN	
MISSION ESSENTIAL / DUAL RED[illegible]CY		2R	2R (CIL)	3
LIFE OR VEHICLE ESSENTIAL / TRIPLE REDUNDANCY		1R	1R (CIL)	3
MISSION ESSENTIAL / TRIPLE REDUNDANCY		2R	2R (CIL)	3
ALL NON-ESSENTIAL / ALL LEVELS OF REDUNDANCY		3		3

FIGURE 10

FAILURE EFFECTS

(A) ON FUNCTION OR SUBSYSTEM:
- No Effect
- Loss of Redundancy
- Functional Degradation
- Subsystem Degradation
- Loss of Function
- Loss of Subsystem

(B) ON INTERFACE FUNCTIONS OR SUBSYSTEMS:
- No Effect
- Loss of Interface Redundancy
- Degradation of Interface Function
- Degradation of Interface Subsystem
- Loss of Interface Function
- Loss of Subsystem

(C) ON MISSION:
- No Effect
- See Note Below for Criticality 2 Modes
- Mission Modification
- Loss of Entry Capability - Rescue

(D) ON CREW/VEHICLE:
- No Effect
- Possible Loss of Crew/Vehicle (Specify)
- Probable Loss of Crew/Vehicle (Conditions)
- Loss of Crew/Vehicle

NOTE: The following instruction is intended to clarify what should be entered in the FMEA/CIL under "EFFECTS ON MISSION" (item C under entry J240) for identified criticality 2 failure modes.

Criticality 2 failure (modes) are defined as: (1) single failures which would cause "loss of mission", and (2) failures wherein the next associated failure would cause loss of crew/vehicle (Appendix B, Section 3.1.1, Ground Rules, subparagraph 1).

The following chart (Mission Effects - Criticality 2 Failure Modes) is included as a guideline for entries under "EFFECTS ON MISSION". The term "abort decision" should only be used where there really is a decision.

MISSION EFFECTS - CRITICALITY 2 FAILURE MODES

ENTRIES UNDER "EFFECTS ON MISSION" IN FMEA/CIL FOR CRITICALITY 2 FAILURE MODES	RATIONALE/TYPICAL SITUATIONS AND EXAMPLES (GUIDELINES NOT TO BE ENTERED IN FMEA/CIL)
1. LOSS OF CAPABILITY TO - - - (ADD SPECIFICS OF CAPABILITY LOST DUE TO IDENTIFIED FAILURE).	FAILURE MODE AFFECTS ACCOMPLISHMENT OF MISSION OBJECTIVE(S) ONLY. EXAMPLE: JAMMED GEAR BOX (SFP) WHICH PRECLUDES RELEASE OR DEPLOYMENT OF A PAYLOAD.
2. ABORT DECISION IF FAILURE OCCURS PRIOR TO ENTRY COMMITMENT.	LOSS OF REDUNDANCY WHEREIN NEXT ASSOCIATED FAILURE WILL CAUSE LOSS OF CREW/VEHICLE. EXAMPLES: APU/HYD, ECLSS FUNCTIONS, POWER GENERATION.
3. NONE - REDUNDANT SIMULTANEOUS MEANS AVAILABLE TO - - - (SPECIFY FUNCTION TO BE PERFORMED.)	SIMULTANEOUS USAGE OF REDUNDANT MEANS TO PERFORM A TIME CRITICAL FUNCTION. EXAMPLE: FAILURE OF ONE OF TWO REDUNDANT PYROTECHNIC DEVICES IN THE SEPARATION SYSTEM.
4. CONTINUE MISSION - DUTY CYCLE SAME AS ABORT.	EXAMPLE: LOSS OF ONE MOTOR WHILE OPENING A DOOR WHICH SUBSEQUENTLY MUST BE CLOSED FOR ENTRY.
NOTE: WHEN DECISION IS TIME DEPENDENT:	
5. ABORT AOA OR ATO IF FAILURE OCCURS DURING ASCENT PHASE.	LOSS OF ONE OMS ENGINE PRECLUDES MISSION COMPLETION.
6. ABORT UNLESS MISSION CAN BE COMPLETED WITH ONE SSME OUT.	MISSION 1 CAN BE COMPLETED IF ONE SSME SHUTS DOWN BETWEEN 347.3 AND 401.5 SECONDS OF ASCENT. MISSION 2 HAS NO ENGINE OUT MISSION COMPLETION CAPABILITY.

3.0 GROUND RULES AND CRITERIA

The following ground rules and criteria are of a general category for guidance, as applicable, in conducting and interpreting an FMEA. The applicable ground rules and criteria will be a part of the information which prefaces each FMEA. (See section 3.2, FMEA Content.)

3.1 GENERAL GROUND RULES AND CRITERIA

3.1.1 GROUND RULES

1. Criticality definitions are those delineated in NHB 5300.4 (1D-1), as illustrated in FIGURE 10 (Criticality Category Cross Reference Table). For the purpose of this analysis, hardware criticality 2 is further defined; i.e., dual redundancy where:
   a. The first failure would result in loss of mission.
   b. The next related failure would result in loss of life/vehicle.

2. Criticality 1R and 2R assumes failure of all like and unlike redundancy: A backup item, if when it is called upon to work, performs a function different from the item it is backing up, it should be classified based upon the effect if it does not work when operated. If the backup item performs the same function as the item it is backing up, the backup should be classified as an unlike redundant item.

3. Loss of mission is defined as follows:
   a. Operation payload interface hardware failure as it would result in loss of payload primary performance.
   b. Orbiter subsystem failure as it would result in unplanned mission termination for non-safety of flight reasons.

4. Categorization of a hardware item by the worst case potential effect of failure of that item will define criticality.

5. Failure modes that could propagate to interfacing subsystems or experiments will be identified.

6. When defining FMEA's/CIL for a particular subsystem, interfacing subsystems will be considered to be operating within their specified tolerances.

7. GFE FMEA data will be utilized in evaluating the GFE/Rockwell International interfaces for the vehicle FMEA and CIL.

8. Failure detectability assumes the availablity of telemetry or a crewman responding to monitored displays. Failure detectability also assumes other means of failure detection, where feasible, such as a crew response to physical stimuli; i.e., smell, sound, etc.

9. Specific FMEA criteria and assumptions will be defined for each subsystem.

10. Identical components used for different functions will be treated separately in the FMEA.

11. Simultaneous failure of redundant components is identified where the failure cause encompasses both components.

12. Subsystem analysis will include an evaluation of the effects of instrumentation failure upon/within the subsystem.

13. External Leakage:
    a. The external leakage mode of functional hardware items from any source (except mating of two surfaces by welding, brazing, or permaswage) will be considered. If this mode raises the criticality of the items in question, it will be documented and the potential leak source identified under "CAUSE(S)". Otherwise, the external leakage will be treated generically by media. However, in those instances where external leakage results in a hardware criticality 1 effect, the failure mode will be documented regardless of the basic criticality of the item being considered. Where applicable, seal failure should be listed as a cause and worst case (complete seal failure) shall be assumed, considering

also any restrictive protection provided by barrier design, where such data are available from Design. Hazards associated with the loss of fluid in excess of requirements will be documented and covered by Hazard Analysis, but will not affect criticality (see section 3.3, paragraph 4.)

b. The internal leak mode of functional hardware items will be considered. In those instances where internal leakage could result in a hardware criticality 1 or 2 failure mode effect (fail open or fail closed due to pressure lockup), "internal leakage" shall be entered in the "CAUSE" section of the most appropriate identified failure mode entry in lieu of "cause" is acceptable.

c. Where external or internal leak paths are protected by static or dynamic redundant (verifiable) seals, the leak path effect will be reduced by one criticality level.

d. Pressure carriers (lines, pressure vessels) will be classified by worst case mode including external leakage. Lines will be entered generically for each independent media. Special lines (i.e., mechanical bellows, flex lines, etc.) will be entered individually. Tanks will be entered individually.

14. The failure of any tank containing fluid media which, due to its location in an enclosed vehicle compartment, could cause compartment overpressurization leading to structural failure (vehicle loss) will be classified hardware criticality 1 for tank rupture mode.

15. All lines will be designated the criticality applicable to the functional loss effect resulting from loss of medium with notation in the "REMARKS" section of the FMEA as to the potential hazard due to compartment overpressurization resulting from line rupture. The main engine cryogenic feedlines will be treated as fluid tanks.

3.1.2 CRITERIA

1. The FMEA and CIL will consider failures beginning with preflight/prelaunch operations through post landing safings at Edwards Air Force Base/Kennedy Space Center (EAFB/KSC).
   a. Prelaunch operations at KSC/VAFB are defined as beginning with propellant loading for each specific propulsion subsystem. For all other subsystems, prelaunch operations commence with start of main engine conditioning.

   b. Post landing safing operations include those activities performed after landing to prepare the orbiter for hangar operations and are defined as follows:
      (1) Deservice and draining of hazardous fluids.
      (2) Safing of unused ordnance.
      (3) Application of ground power and cooling.
      (4) Removal of potentially hazardous components.
      (5) Removal of pods and payloads.
      (6) Purging and venting of gases.
      (7) Installation of protective covers.

2. Redundancy is defined as the use of more than one means of accomplishing a given task or function where all must fail before there is an overall failure of the function.
   a. Operational Redundancy - redundant elements, all of which are fully energized during the subsystem operating cycle. Operational redundancy includes load sharing redundancy wherein redundant elements are connected in such a manner that, upon failure of one unit, the remaining redundant elements will continue to perform the subsystem function. It is not necessary to switch out the failed element nor to switch in the redundant element.

   b. Standby Redundancy - redundant elements that are non-operative (i.e., have no power applied) until they are switched into the subsystem upon failuare of the primary element. In these cases, as well as pyrotechnic devices, mechanical linkage and inoperative functional paths of any subsystem, redundancy screen B is considered not applicable and so marked. This approach is based on the fact that these areas are subject to ground checkout; they are redundant and therefore provide a degree of protection in flight; it is confirmed that when called upon to operate, the inability to operate would be detected in flight for appropriate corrective action.

3. Where redundancy exists in the subsystem, the redundancy is considered during the analysis of a failure of the component.

4. "Alternate means of operation" refers to accomplishment of a function and not necessarily to redundancy or restoration of a failed function.

5. When fire hazards resulting from short circuits or other hardware failure modes are identified, consideration will be given to the effect of fire propagation to adjacent redundant equipment as a potential loss of the function.

   Potential safety concerns created by component failure modes will be identified and handled through Hazards Analyses as required by EOM 70 1-4.2.5 and by NHB 5300.4 (1D-1).

6. Reference documents in the FMEA include released and controlled engineering drawings or specifications, when available.

7. The following are used as aids in determining the failure modes and causes of subsystem hardware failures:

   a. Generic failure modes and causes.
   b. Released and controlled component, assembly, and detail engineering drawings and specifications.
   c. Training aids, as available; e.g., cross section drawings, photographs, exploded drawings (not referenced in FMEA).
   d. Actual hardware, if available.
   e. Use experience, including failure history and similar components.

f. Controlled and released operational procedures.

g. Component FMEA's prepared by component suppliers.

8. Failure of structural items (primary and secondary) will not be considered as a part of this ananysis. (Structural items are assumed to be designed to preclude failure by use of adequate design safety factors.)

9. FMEA's of criticality 1 or 2 "black boxes" providing electrical signal interface to mechanical components are included in the applicable avionics FMEA package with appropriate cross-referencing in the "REFERENCE DOCUMENTS" section of both the appropriate avionics and mechanical FMEA report, where available.

10. The failure mode "Fails to Operate" will *not* be addressed for fuses. Use of a fuse with a higher current capacity then specified (wrong size installed or rating misidentified) is not considered a fuse failure mode.

VII-A

Designation: D 395 - 85

AMERICAN SOCIETY FOR TESTING AND MATERIALS
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If not listed in the current combined index, will appear in the next edition.

Standard Test Methods for RUBBER PROPERTY—COMPRESSION SET[1]

This standard is issued under the fixed designation D 395; the number immediately following the designation indicates the year of original adoption or, in the case of revision, the year of last revision. A number in parentheses indicates the year of last reapproval. A superscript epsilon (ϵ) indicates an editorial change since the last revision or reapproval.

These methods have been approved for use by agencies of the Department of Defense and for listing in the DoD Index of Specifications and Standards.

1. Scope

1.1 These test methods cover the testing of rubber intended for use in applications in which the rubber will be subjected to compressive stresses in air or liquid media. They are applicable particularly to the rubber used in machinery mountings, vibration dampers, and seals. Two methods are covered as follows:

Method	Section
A—Compression Set Under Constant Force in Air	7–10
B—Compression Set Under Constant Deflection in Air	11–14

1.2 The choice of method is optional, but consideration should be given to the nature of the service for which correlation of test results may be sought. Unless otherwise stated in a detailed specification, Method B shall be used.

1.3 Method B is not suitable for vulcanizates harder than 90 IRHD.

1.4 The values stated in SI units are to be regarded as the standard.

1.5 *This standard may involve hazardous materials, operations, and equipment. This standard does not purport to address all of the safety problems associated with its use. It is the responsibility of whoever uses this standard to consult and establish appropriate safety and health practices and determine the applicability of regulatory limitations prior to use.*

2. Applicable Documents

2.1 *ASTM Standards:*

D 1349 Practice for Rubber—Standard Temperatures and Atmospheres for Testing and Conditioning[2]

D 3040 Practice for Preparing Precision Statements for Standards Related to Rubber and Rubber Testing[2]

D 3182 Practice for Rubber—Materials, Equipment, and Procedures for Mixing Standard Compounds and Preparing Standard Vulcanized Sheets[3]

D 3183 Practice for Rubber—Preparation of Pieces for Test Purposes from Products[3]

D 3767 Practice for Rubber—Measurement of Dimensions[3]

E 145 Specification for Gravity-Convection and Forced-Ventilation Ovens[4]

NOTE 1—The specific dated edition of Practice D 3040 that prevails in this document is referenced in the Precision section.

3. Summary of Methods

3.1 A test specimen is compressed to either a deflection or by a specified force and maintained under this condition for a specified time and at a specified temperature.

3.2 The residual deformation of a test specimen is measured 30 min after removal from a suitable compression device in which the specimen had been subjected for a definite time to compressive deformation under specified conditions.

3.3 After the measurement of the residual deformation, the compression set as specified in the

[1] These test methods are under the jurisdiction of ASTM Committee D-11 on Rubber and are the direct responsibility of Subcommittee D11.10 on Physical Testing.

Current edition approved Sept. 27, 1985. Published November 1985. Originally published as D 395 – 34. Last previous edition D 395 – 84.

[2] *Annual Book of ASTM Standards*, Vols 09.01 and 09.02.

[3] *Annual Book of ASTM Standards*, Vol 09.01.

[4] *Annual Book of ASTM Standards*, Vol 14.02.

appropriate method, is calculated according to Eqs (1) and (2).

4. Significance and Use

4.1 Compression set tests are intended to measure the ability of rubber compounds to retain elastic properties after prolonged action of compressive stresses. The actual stressing service may involve the maintenance of a definite deflection, the constant application of a known force, or the rapidly repeated deformation and recovery resulting from intermittent compressive forces. Though the latter dynamic stressing, like the others, produces compression set, its effects as a whole are simulated more closely by compression flexing or hysteresis tests. Therefore, compression set tests are considered to be mainly applicable to service conditions involving static stresses. Tests are frequently conducted at elevated temperatures.

5. Test Specimens

5.1 Specimens from each sample may be tested in duplicate (Option 1) or triplicate (Option 2). The compression set of the sample in Option 1 shall be the average of the two specimens, expressed as a percentage. The compression set of the sample in Option 2 shall be the median (middle most value) of the three specimens expressed as a percentage.

5.2 The standard test specimen shall be a cylindrical disk cut from a laboratory prepared slab.

5.2.1 The dimensions of the standard specimens shall be:

Type	1[A]	2[B]
Thickness, mm (in.)	12.5 ± 0.5 (0.49 ± 0.02)	6.0 ± 0.2 (0.24 ± 0.01)
Diameter, mm (in.)	29.0 ± 0.5 (1.14 ± 0.02)	13.0 ± 0.2 (0.51 ± 0.01)

[A] Type 1 specimen is used in Methods A and B.
[B] Type 2 specimen is used in Method B.

5.2.2 When cutting the standard specimen, the circular die having the required inside dimensions specified in 5.2.1 shall be rotated in a drill press or similar device and lubricated by means of a soap solution. A minimum distance of 13 mm (0.51 in.) shall be maintained between the cutting edge of the die and the edge of the slab. The cutting pressure shall be as light as possible to minimize cupping of the cut edges. The dies shall be maintained carefully so that the cutting edges are sharp and free of nicks.

5.3 An optional method of preparing the standard specimen may be the direct molding of a circular disk having the dimensions required for the method used and specified in 5.2.1.

NOTE 2—It should be recognized that an equal time and temperature, if used for both the slab and molded specimen, will not produce an equivalent state of cure in the two types of specimen. A higher degree of cure will be obtained in the molded specimen. Adjustments, preferably in the time of cure, must be taken into consideration if comparisons between the specimens prepared by different methods are to be considered valid.

NOTE 3—It is suggested, for the purpose of uniformity and closer tolerances in the molded specimen, that the dimensions of the mold be specified and shrinkage compensated for therein. A two-plate mold with a cavity 13.0 ± 0.1 mm (0.510 ± 0.004 in.) in thickness and 29.20 ± 0.05 mm (1.148 ± 0.002 in.) in diameter, with overflow grooves will provide Type 1 specimens for Method A and Method B. A similar mold but having a cavity of 6.3 ± 0.3 mm (0.25 ± 0.012 in.) in thickness and 13.2 ± 0.1 mm (0.52 ± 0.004 in.) in diameter will provide Type 2 specimens for Method B.

5.4 When the standard test specimen is to be replaced by a specimen taken from a vulcanized rubber part of greater thickness than the one indicated in 5.2.1, the sample thickness shall be reduced first by cutting transversely with a sharp knife and then followed by buffing to the required thickness in accordance with Practice D 3183.

5.5 An alternative method of preparing specimens is by plying up cylindrical disks cut from a standard sheet prepared in accordance with Practice D 3182 using the specimen sizes specified in 5.2.1 and cutting as described in 5.2.2.

5.5.1 The disks shall be plied, without cementing, to the thickness required. Such plies shall be smooth, flat, of uniform thickness, and shall not exceed seven in number for Type 1 specimens and four in number for Type 2 specimens.

5.5.2 Care shall be taken during handling and placing of the plied test specimen in the test fixture by keeping the circular faces parallel and at right angles to the axis of the cylinder.

5.5.3 The results obtained on plied specimens may be different from those obtained using solid specimens and the results may be variable, particularly if air is trapped between disks.

5.5.4 The results obtained on the specimens prepared by one of the methods may be compared only to those prepared by the same method.

5.6 For routine or product specification testing, it is sometimes more convenient to prepare specimens of a different size or shape, or both. When such specimens are used, the results should be compared only with those obtained from specimens of similar size and shape and not with those obtained with standard specimen. For such cases, the product specification should define the specimen as to the size and shape. If suitable specimens cannot be prepared from the product, the test method and allowable limits must be agreed upon between the producer and the purchaser.

6. Conditioning

6.1 Store all vulcanized test specimens or product samples to be tested at least 24 h but not more than 60 days. When the date of vulcanization is not known, make tests within 60 days after delivery by the producer of the article represented by the specimen.

6.2 Allow buffed specimens to rest at least 30 min before specimens are cut for testing.

6.3 Condition all specimens before testing for a minimum of 3 h at 23 ± 2°C (73.4 ± 3.6°F). Specimens whose compression set properties are affected by atmospheric moisture, shall be conditioned for a minimum of 24 h in an atmosphere controlled to 50 ± 5 % relative humidity.

METHOD A—COMPRESSION SET UNDER CONSTANT FORCE IN AIR

7. Apparatus

7.1 *Dial Micrometer*—A dial micrometer, for measuring specimen thickness, in accordance with Practice 3767, Method A1.

7.2 *Compression Device,* consisting of a force application spring and two parallel compression plates assembled by means of a frame or threaded bolt in such a manner that the device shall be portable and self-contained after the force has been applied and that the parallelism of the plates shall be maintained. The force may be applied in accordance with either 7.2.1 or 7.2.2.

7.2.1 *Calibrated Spring Force Application*—The required force shall be applied by a screw mechanism for compressing a calibrated spring the proper amount. The spring shall be of properly heat-treated spring steel with ends ground and perpendicular to the longitudinal axis of the spring. A suitable compression device is shown in Fig. 1. The spring shall conform to the following requirements:

7.2.1.1 The spring shall be calibrated at room temperature 23 ± 5°C (73.4 ± 9°F) by applying successive increments of force not exceeding 250 N (50 lbf) and measuring the corresponding deflection to the nearest 0.2 mm (0.01 in.). The curve obtained by plotting the forces against the corresponding deflections shall have a slope of 70 ± 3.5 kN/m (400 ± 20 lbf/in.) at 1.8 kN (400 lbf). The slope is obtained by dividing the two forces above and below 1.8 kN by the difference between the corresponding deflections.

7.2.1.2 The original dimensions of the spring shall not change due to fatigue by more than 0.3 mm (0.01 in.) after it has been mounted in the compression device, compressed under a force of 1.8 kN (400 lbf), and heated in the oven for one week at 70°C ± 2°C (158 ± 3.6°F). In ordinary use, a weekly check of the dimensions shall show no greater change than this over a period of 1 year.

7.2.1.3 The minimum force required to close the spring (solid) shall be 2.4 kN (530 lbf).

7.2.2 *External Force Application*—The required force shall be applied to the compression plates and spring by external means after the test specimen is mounted in the apparatus. Either a calibrated compression machine or known masses may be used for force application. Provision shall be made by the use of bolts and nuts or other devices to prevent the specimen and spring from losing their initial deflections when the external force is removed. The spring shall have essentially the same characteristics as described in 7.2.1, but calibration is not required. A suitable compression device is shown in Fig. 2.

7.3 *Plates*—The plates between which the test specimen is compressed shall be made of steel of sufficient thickness to withstand the compressive stresses without bending. The surfaces against which the specimen is held shall have a highly polished chromium-plated finish and shall be cleaned thoroughly and wiped dry before each test.

7.4 *Oven,* conforming to the specification for a Type IIB laboratory oven given in Specification E 145.

8. Procedure

8.1 *Original Thickness Measurement*—Measure the original thickness of the specimen to the nearest 0.02 mm (0.001 in.). Place the specimen on the anvil of the dial micrometer so that the presser foot will indicate the thickness at the central portion of the top and bottom faces.

8.2 *Application of Compressive Force*—Assemble the specimens in the compression device, using extreme care to place them exactly in the center between the plates to avoid tilting. If the calibrated spring device (Fig. 1) is used, apply the compressive force by tightening the screw until the deflection as read from the scale is equivalent to that shown on the calibration curve for the spring corresponding to a force of 1.8 kN (400 lbf). With the external loading device (Fig. 2), apply this force to the asembly in the compression machine or by adding required masses, but in the latter case, take care to add the mass gradually without shock. Tighten the nuts and bolts just sufficiently to hold the initial deflections of the specimen and spring. It is imperative that no additional force be applied in tightening the bolts.

8.3 *Test Time and Test Temperature*—Choose a suitable temperature and time for the compression set, depending upon the conditions of the expected service. In comparative tests, use identical temperature and heating periods. It is suggested that the test temperature be chosen from those listed in Practice D 1349. Suggested test periods are 22 h and 70 h. The specimen shall be at room temperature when inserted in the compression device. Place the assembled compression device in the oven within 2 h after completion of the assembly and allow it to remain there for the required test period in dry air at the test temperature selected. At the end of the test period, take the device from the oven and remove the specimens immediately and allow it to cool.

8.4 *Cooling Period*—While cooling, allow the specimens to rest on a poor thermally conducting surface, such as wood, for 30 min before making the measurement of the final thickness. Conduct the cooling period at a standard laboratory temperature of 23 ± 2°C (73.4 ± 3.6°F). Specimens whose compression set property is affected by atmospheric moisture shall be cooled in an atmosphere controlled to 50 ± 5 % relative humidity.

8.5 *Final Thickness Measurement*—After the rest period, measure the final thickness at the center of the specimen in accordance with 8.1.

9. Calculation

9.1 Calculate the compression set as a percentage of the original thickness as follows:

$$C_A = [(t_o - t_i)/t_o] \times 100 \quad (1)$$

where:

C_A = compression set (Method A) as a percentage of the original thickness,
t_o = original thickness (8.1), and
t_i = final thickness (8.5).

10. Report

10.1 The report shall include the following:

10.1.1 Original dimensions of the test specimen, including the original thickness, t_o,

10.1.2 Actual compressive force on the specimen as determined from the calibration curve of the spring and spring deflection reading (7.2.1) or as applied by an external force (7.2.2),

10.1.3 Thickness of the test specimen 30 min after removal from the clamp, t_i,

10.1.4 Type of test specimen used, together with the time and temperature of test,

10.1.5 Compression set, expressed as a percentage of the original thickness,

10.1.6 Method used (Method A), and

10.1.7 Number of specimens tested.

METHOD B—COMPRESSION SET UNDER CONSTANT DEFLECTION IN AIR

11. Apparatus

11.1 *Dial Micrometer*—A dial micrometer, for measuring the specimen thickness, in accordance with Practice D 3767, Method A1.

NOTE 4—For vulcanizates having a hardness below 35 IRHD, the force on the presser foot should be reduced to 0.2 ± 0.05 N (0.04 ± 0.01 lbf).

11.2 *Spacer Bars*, to maintain the constant deflection required under Method B.

11.2.1 Spacer bars for Type 1 samples shall have a thickness of 9.5 ± 0.02 mm (0.375 ± 0.001 in.).

11.2.2 Spacer bars for Type 2 samples shall have a thickness of 4.50 ± 0.01 mm (0.1770 ± 0.0005 in.).

11.3 *Compression Device*, consisting of two or more flat steel plates between the parallel faces of which the specimens may be compressed as shown in Fig. 3. Steel spacers for the required percentage of compression given in 12.2 shall be placed on each side of the rubber specimens to control their thickness while compressed. The steel surfaces contacting the rubber specimens shall be ground to a maximum roughness of 250 μm (10 μin.) and then chromium plated and polished.

11.4 *Oven,* conforming to the specification for a Type IIB laboratory oven given in Specification E 145.

11.5 *Plates*—The plates between which the test specimen is compressed shall be made of steel of sufficient thickness to withstand the compressive stresses without bending. The surfaces against which the specimen is held shall have a highly polished chromium-plated finish and shall be cleaned thoroughly and wiped dry before each test.

12. Procedure

12.1 *Original Thickness Measurement*—Measure the original thickness of the specimen to the nearest 0.02 mm (0.001 in.). Place the specimen on the anvil of the dial micrometer so that the presser foot will indicate the thickness at the central portion of the top and bottom faces.

12.2 *Application of Compressive Force*—Place the test specimen between the plates of the compression device with the spacers on each side, allowing sufficient clearance for the bulging of the rubber when compressed (Fig. 3). Where a lubricant is applied, it shall consist of a thin coating of a lubricant having substantially no action on the rubber. For most purposes, a silicon or fluorosilicon fluid is suitable. Tighten the bolts so that the plates are drawn together uniformly until they are in contact with the spacers. The amount of compression employed shall be approximately 25 %. A suitable mechanical or hydraulic device may be used to facilitate assembling and disassembling the test fixture.

12.3 *Test Time and Temperature*—Choose a suitable temperature and time for the compression set, depending upon the conditions of the expected service. In comparative tests, use identical temperature and test periods. It is suggested that the test temperature be chosen from those listed in Recommended Practice D 1349. Suggested test periods are 22 h and 70 h. The test specimen shall be at room temperature when inserted in the compression device. Place the assembled compression device in the oven within 2 h after completion of the assembly and allow it to remain there for the required test period in dry air at the test temperature selected. At the end of the test period, take the device from the oven and remove the test specimen immediately and allow them to cool.

12.4 *Cooling Period*—While cooling, allow the test specimen to rest on a poor thermally conducting surface, such as wood, for 30 min before making the measurement of the final thickness. Maintain the conditions during the cooling period in accordance with 8.4.

12.5 *Final Thickness Measurement*—After the rest period, measure the final thickness at the center of the test specimen in accordance with 12.1.

13. Calculation

13.1 Calculate the compression set expressed as a percentage of the original deflection as follows:

$$C_B = [(t_o - t_i)/(t_o - t_n)] \times 100 \qquad (2)$$

where:

C_B = compression set (Method B) expressed as percentage of the original deflection,
t_o = original thickness of specimen (12.1),
t_i = final thickness of specimen (12.5), and
t_n = thickness of the spacer bar used.

NOTE 5—Lubrication of the operating surfaces of the compression device is optional while giving more reproducible results, lubrication may somewhat alter the compression set values.

14. Report

14.1 The report shall include the following:

14.1.1 Original dimensions of the test specimen including the original thickness, t_o,

14.1.2 Percentage compression of the specimen actually employed,

14.1.3 Thickness of the test specimen 30 min after removal from the clamp, t_i,

14.1.4 Type of test specimen used, together with the time and temperature of test,

14.1.5 Whether or not the surfaces of the compression device are lubricated. If they are, what type lubrication was used,

14.1.6 Compression set, expressed as a percentage of the original deflection,

14.1.7 Method used (Method B), and

14.1.8 Numbe... specimens tested.

15. Precision[5] and B...

15.1 These precision statements have been prepared in accordance with Practice D 3040-81. Please refer to this practice for terminology and other testing and statistical concepts.

15.2 Prepared test specimens of two rubbers, Methods A and B, were supplied to five laboratories. These were tested in duplicate each day on two separate testing days. A test result, therefore, is the average of two test specimens, for both Methods A and B.

15.3 One laboratory did not run the Method A testing, therefore the precision for Method A is derived from four laboratories.

15.4 The precision results are given in Tables 1 and 2.

15.5 *Bias*—In test method statistical terminology, bias is the difference between an average test value and the reference or true test property value. Reference values do not exist for this test method since the value or level of the test property is exclusively defined by the test method. Bias, therefore, cannot be determined.

[5] Supporting data are available from ASTM Headquarters. Request RR: D-11-1138.

TABLE 1 LQC Precision Data Compression Set—Method A

Material	Mean Level	Within Laboratories		Among Laboratories	
		S	*CV*	*S*	*CV*
A	1.73 (%)	0.0500	0.0277	0.190	0.1096
B	26.1	0.898	0.0336	2.37	0.0908
Average or Pooled Values		0.636	0.0308	1.681	0.1006

	Repeatability	Reproducibility
Standard Deviation, $(S)^A$	0.636	1.743
Coefficient or Variation, (CV)	0.0308	0.103
Least Significant Difference, $(LSD)^{B,C}$	8.8 %	29.1 %

[A] An average value, the value of *S* varies with mean level.
[B] *LSD* based on 95 % confidence level; two results are considered significantly different if their difference, expressed as a percentage of their average, exceeds the stated percent value.
[C] The *LSD* values are relative percent, that is, a percent of the "percent" values used to measure the tested property.

TABLE 2 LQC Precision Data Compression Set—Method B

Material	Mean Level	Within Laboratories		Among Laboratories	
		S	*CV*	*S*	*CV*
A	13.7 (%)	0.591	0.0420	1.543	0.113
B	52.8	0.567	0.0110	5.924	0.112
Average or Pooled Values		0.579	0.0307	4.329	0.1124

	Repeatability	Reproducibility
Standard Deviation, $(S)^A$	0.579	4.348
Coefficient or Variation, (CV)	0.0307	0.114
Least Significant Difference, $(LSD)^{B,C}$	8.7 %	32.4 %

[A] An average value, the value of *S* varies with mean level.
[B] *LSD* based on 95 % confidence level; two results are considered significantly different if their difference, expressed as a percentage of their average, exceeds the stated percent value.
[C] The *LSD* values are relative percent, that is, a percent of the "percent" values used to measure the tested property.

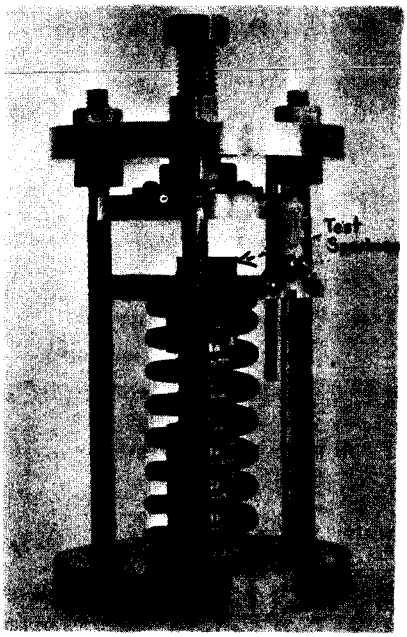

FIG. 1 Device for Compression Set Test, Using Calibrated Spring Loading, Method A

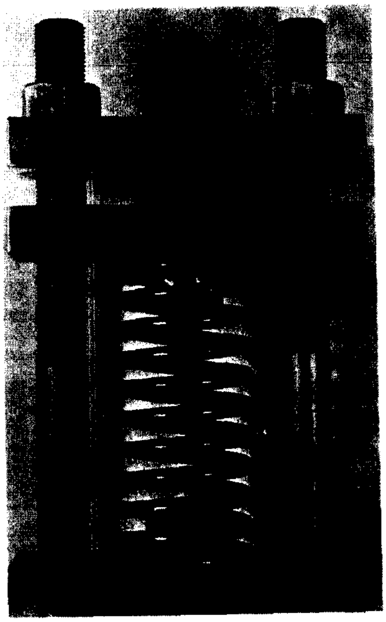

FIG. 2 Device for Compression Set Test, Using External Loading, Method A

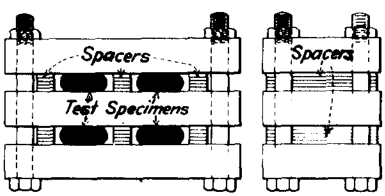

FIG. 3 Device for Compression Set Test Under Constant Deflection, Method B

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This standard is subject to revision at any time by the responsible technical committee and must be reviewed every five years and if not revised, either reapproved or withdrawn. Your comments are invited either for revision of this standard or for additional standards and should be addressed to ASTM Headquarters. Your comments will receive careful consideration at a meeting of the responsible technical committee, which you may attend. If you feel that your comments have not received a fair hearing you should make your views known to the ASTM Committee on Standards, 1916 Race St., Philadelphia, Pa. 19103.

VII-B

NASA

National Aeronautics and
Space Administration

Washington, D.C.
20546

JUL 17 1985

to Attn of: MPS

TO: M/Associate Administrator for Space Flight

FROM: MPS/Irv Davids

SUBJECT: Case to Case and Nozzle to Case "O" Ring Seal Erosion Problems

As a result of the problems being incurred during flight on both case to case and nozzle to case "O" ring erosion, Mr. Hamby and I visited MSFC on July 11, 1985, to discuss this issue with both project and S&E personnel. Following are some important factors concerning these problems:

A. Nozzle to Case "O" ring erosion

There have been twelve (12) instances during flight where there have been some primary "O" ring erosion. In one specific case there was also erosion of the secondary "O" ring seal. There were two (2) primary "O" ring seals that were heat affected (no erosion) and two (2) cases in which soot blew by the primary seals.

The prime suspect as the cause for the erosion on the primary "O" ring seals is the type of putty used. It is Thiokol's position that during assembly, leak check, or ignition, a hole can be formed through the putty which initiates "O" ring erosion due to a jetting effect. It is important to note that after STS-10, the manufacturer of the putty went out of business and a new putty manufacturer was contracted. The new putty is believed to be more susceptible to environmental effects such as moisture which makes the putty more tacky.

There are various options being considered such as removal of putty, varying the putty configuration to prevent the jetting effect, use of a putty made by a Canadian Manufacturer which includes asbestos, and various combination of putty and grease. Thermal analysis and/or tests are underway to assess these options.

Thiokol is seriously considering the deletion of putty on the QM-5 nozzle/case joint since they believe the putty is the prime cause of the erosion. A decision on this change is planned to be made this week. I have reservations about doing it, considering the significance of the QM-5 firing in qualifying the FWC for flight.

PC 098840

It is important to note that the cause and effect of the putty varies. There are some MSFC personnel who are not convinced that the holes in the putty are the source of the problem but feel that it may be a reverse effect in that the hot gases may be leaking through the seal and causing the hole track in the putty.

10

Considering the fact that there doesn't appear to be a validated resolution as to the effect of putty, I would certainly question the wisdom in removing it on QM-5.

B. Case to Case "O" Ring Erosion

There have been five (5) occurrences during flight where there was primary field joint "O" ring erosion. There was one case where the secondary "O" ring was heat affected with no erosion. The erosion with the field joint primary "O" rings is considered by some to be more critical than the nozzle joint due to the fact that during the pressure build up on the primary "O" ring the unpressurized field joint secondary seal unseats due to joint rotation.

The problem with the unseating of the secondary "O" ring during joint rotation has been known for quite some time. In order to eliminate this problem on the FWC field joints a capture feature was designed which prevents the secondary seal from lifting off. During our discussions on this issue with MSFC, an action was assigned for them to identify the timing associated with the unseating of the secondary "O" ring and the seating of the primary "O" ring during rotation. How long it takes the secondary "O" ring to lift off during rotation and when in the pressure cycle it lifts are key factors in the determination of its criticality.

The present consensus is that if the primary "O" ring seats during ignition, and subsequently fails, the unseated secondary "O" ring will not serve its intended purpose as a redundant seal. However, redundancy does exist during the ignition cycle, which is the most critical time.

It is recommended that we arrange for MSFC to provide an overall briefing to you on the SRM "O" rings, including failure history, current status, and options for correcting the problems.

Ir. Davids
Irving Davids

cc:
M/Mr. Weeks
M/Mr. Hamby
ML/Mr. Harrington
MP/Mr. Winterhalter

VII-C

National Aeronautics and
Space Administration

George C. Marshall Space Flight Center
Marshall Space Flight Center, Alabama
35812

NASA

August 1985

Reply to Attn of: SA42-349-85

TO: NASA Headquarters
Attn: Code M/W. H. Hamby

FROM: SA41/L. B. Mulloy

SUBJECT: SRM Joint/O-Ring Erosion

On July 11, 1985 you and Irv Davids were briefed by Jim Thomas of my office on the history and the effort underway to resolve the issues and concerns of the above subject. During this briefing the following information was requested:

1. Question: If the field joint secondary seal lifts off the metal mating surfaces during motor pressurization, how soon will it return to a positon where contact is re-established?

   Answer: Bench test data indicates that the o-ring resiliency (its capability to follow the metal) is a function of temperature and rate of case expansion. MTI measured the force of the o-ring against Instron plattens, which simulated the nominal squeeze on the o-ring and approximated the case expansion distance and rate.

   At 100°F the o-ring maintained contact. At 75°F the o-ring lost contact for 2.4 seconds. At 50°F the o-ring did not re-establish contact in ten minutes at which time the test was terminated.

   The conclusion is that secondary sealing capability in the SRM field joint cannot be guaranteed.

2. Question: If the primary o-ring does not seal, will the secondary seal seat in sufficient time to prevent joint leakage?

   Answer: MTI has no reason to suspect that the primary seal would ever fail after pressure equilibrium is reached, i.e., after the ignition transient. If the primary o-ring were to fail from 0 to 170 milliseconds, there is a very high probability that the secondary o-ring would hold pressure since the case has not expanded appreciably at this point. If the primary seal were to fail from 170 to 330 milliseconds, the probability of the secondary seal holding is reduced. From 330 to 600 milliseconds the chance of the secondary seal holding is small. This is a direct result of the o-ring's slow response compared to the metal-case segments as the joint rotates.

3. Question: Headquarters was not aware that the secondary o-ring may not seat due to joint rotation, when was this incorporated into the FMEA/CIL?

Answer: MTI submitted TWR-13520 "Retention Rationale, SRM Simplex Seal" to MSFC on 12/1/82. The SRB CIL requirement change was approved by Level III CCB on 1/21/83. Level II authorized submittal of Level I change request S22106L on 3/2/83. On 5/2/83 Level II issued PRCBD S22106LR1 to implement approved Level I change request S22106L and S22106M.

L. B. Mulloy
Manager, SRB Project

cc:
SA41/Messrs. Wear/Thomas

VIII-A

MAR 12 '86 12:11 NASA HQ WASH. D.C. PAGE.04

NASA
Space
Flight

SFO-PD- 710.5A
DATE- 9/26/83

IN EFFECT FOR
STS 51-L

PROGRAM DIRECTIVE

Responsible Office: SF/Space Shuttle Operations, Integration Division

SUBJECT: Space Shuttle Flight Readiness Reviews

1. PURPOSE

This Directive defines the responsibilities, requirements, and procedures to insure effective planning for and conduct of Flight Readiness Reviews for the Space Transportation System (STS) missions.

2. SCOPE

This Directive is applicable to all STS missions.

3. POLICY

It is the policy of the Associate Administrator for Space Flight (AA-SF) to make an assessment of mission readiness prior to each flight. This will be accomplished by a consolidated Flight Readiness Review (FRR) of all activities/elements necessary for safe and successful conduct of the launch, flight, and post-landing operations. The review will be supported by the NASA Chief Engineer and the Center Directors from JSC, MSFC, and KSC.

The FRR will be preceded by detailed readiness reviews (pre-FRR's) on individual elements, including cargo, under the cognizance of the responsible Managers.

4. RESPONSIBILITIES

a. The conduct of the FRR is the responsibility of the AA-SF or his designated representative.

b. The Director, Space Shuttle Operations, is responsible for FRR planning and requirements, coordinating the FRR agenda, FRR action items and action item closeouts, and preparing the readiness assessment.

c. The JSC Level II National Space Transportation Systems (NSTS) Program Manager is responsible for implementing readiness review plans and requirements, organizing the FRR, and for the program assessment of flight readiness.

e. The Project/Element Managers will conduct pre-FRR's to develop their readiness assessment and are responsible for the FRR briefing content

Att 1-1

45

in their particular area.

5. <u>REVIEW REQUIREMENTS</u>

a. <u>Review Concept</u>: The FRR will employ a delta review concept from prior reviews (eg. Acceptance, Orbiter Rollout/Pre-Stack, Cargo Readiness) and previous STS missions. The review will be conducted by telecon using the NASA Teleconference Network.

b. <u>Schedule</u>: The FRR will be held between one and two weeks prior to launch.

c. <u>Agenda</u>: The major agenda items and responsibilities are:

(1) Introduction ———— Headquarters/AA-SF

(2) Integrated System ———— JSC/Manager, NSTS Program

(3) Orbiter/Crew GFE ———— JSC/Manager, Space Shuttle Projects

(4) SSME/ET/SRB ———— MSFC/Manager, Space Shuttle Projects

(5) Cargo ———— JSC/Manager, Mission Integration

(6) Launch & Landing Operations ———— KSC/Director, Shuttle Projects Management

(7) Flight Operations ———— JSC/Director, Mission Operations

(8) Safety, Reliability & Quality Assurance ———— JSC/SR & QA Manager, Space Shuttle

(9) Action Items/Readiness Poll ———— Headquarters/AA-SF

d. <u>Presentation Emphasis</u>: The presentation of agenda items will normally include a brief status summary with appropriate supporting detail on significant items and conclude with a readiness assessment. The presentation topics and scope should be developed from the pre-FRR's and should:

(1) be that required to provide the AA-SF with the information needed to make a judgment as to flight readiness;

(2) review recent significant resolved problems and prior flight anomalies when necessary to establish confidence;

(3) cover all problems, open items, and constraints remaining to be resolved before the mission;

(4) establish the mission baseline configuration in terms of all significant changes since the last STS mission (changes to be considered include hardware, software, vehicle servicing/checkout, launch commit criteria, flight plans.

flight rules, and crew procedures);

Within the above guidelines, the scope of the review should cover status and issues in areas such as: vehicle checkout, shortages and open work, unexplained anomalies, hardware failures, prior flight anomalies, certification/verification, as-built hardware configuration versus certified hardware list, Critical Item List (CIL), development, qualification, and reliability testing, waivers and deviations, limited life components, launch critical spares, sneak circuits, system safety/hazards, and flight margins. In addition, the review should also cover readiness of ground systems/facilities (e.g., launch site, landing sites, network, Mission Control Center), operations teams (e.g., launch, recovery, flight crew, flight control), and support operations (e.g., DOD, contingency, weather, medical, security).

6. DOCUMENTATION REQUIREMENTS

a. Presentations: Vugraphs will be used for FRR presentations, and a paper copy will be provided to the reviewing officials. A copy of the presentation material shall be provided to telecon locations 24 hours in advance of the review.

b. Certificate of Flight Readiness (CoFR): CoFR endorsements, including a complete listing of exceptions, will be executed and submitted at the conclusion of the review (Reference: Space Shuttle Program Procedure for Certification of Flight Readiness (CoFR), JSC 08117, current revision).

c. FRR Assessment: A FRR assessment letter will be issued by OSF within two working days after the review and will include the following:

(1) Action items;

(2) Significant decisions;

(3) CoFR exceptions and endorsement summary.

7. FRR ACTION ITEM/OPEN ITEM CLOSEOUT REPORTING REQUIREMENTS

a. Status: The NSTS Program Manager will report periodically on the closeout status of action items/open items identified at the FRR. This may be accomplished by the Daily Special Level II PRCB.

Significant items occurring subsequent to the FRR will also be reported to the AA-SF. Actions that can be easily accomplished without safety, mission, or launch impact and do not violate flight vehicle or launch complex configuration integrity or cause basic changes to launch commit criteria, flight rules, flight plan, or abort and alternate mission plans, need not be reported.

b. Action Item Closeouts: Closeouts to FRR action items will be submitted to the AA-SF by Level III/II in writing or TWX and will state fully the basis for closeout, that is: action taken, results obtained, and determinations made. An action will not be closed until signoff is completed by the AA-SF or his designated representative.

c. CoFR Exceptions: The resolution of all exceptions identified in the CoFR endorsement will be completed by two days prior to launch, and a copy of the endorsement will be submitted to the Director, Space Shuttle Operations.

8. PROCEDURES

a. Single Points of Contact: FRR planning and procedures will be coordinated through single points of contact established at the following locations:

Headquarters - Space Shuttle Operations, Integration Division

JSC - Level II NSTS Program Office

MSFC - Space Shuttle Projects Office

KSC - Space Shuttle Projects Management Directorate

Individuals will be designated by the responsible organizations and the names provided to the Director, Space Shuttle Operations. Duties will be defined by separate correspondence issued by the Director, Space Shuttle Operations.

b. Guidelines: Approximately three weeks prior to the FRR, guidelines will be issued by the Director, Space Shuttle Operations, establishing the review date and any special requirements not covered by this directive.

c. FRR Agenda: Proposed agenda items/times will be submitted to Level I/II by the single points of contact at the respective Centers two weeks prior to the FRR. Responsible Space Shuttle Operations Directors will coordinate detailed presentation items with their Project and will accept items determined to be completed and not requiring the AA-SF review at the FRR. The final FRR agenda will be coordinated by the Director, Space Shuttle Operations, and issued about one week prior to the FRR.

9. ACTION

This Directive supersedes STS-PD 710.5 dated May 13, 1981, and shall be implemented by the JSC/NSTS Program Manager effective STS-9 and subsequent missions.

James A. Abrahamson
JAMES A. ABRAHAMSON
Lieutenant General, USAF
Associate Administrator for
Space Flight

VIII-B

TWR-15380

STS-51L (STS-33) SOLID ROCKET MOTOR (SRM-25)
FLIGHT READINESS REVIEW

11 DECEMBER 1985

MORTON THIOKOL, INC.

WASATCH DIVISION
P.O. BOX 524, BRIGHAM CITY, UTAH 84302 (801) 863-3511

AGENDA

SECTION		
1.0	o	STS-61C (STS-32) (SRM-24) PERFORMANCE
2.0	o	PROBLEM SUMMARY
3.0	o	STS-51L (STS-33) (SRM-25) CHANGES
		o CHANGE SUMMARY
		o DESIGN
		o HARDWARE CHANGEOUT
		o PROCESS CHANGES
		o FIRST TIME CONFIGURATION REUSED
		o OMRSD
		o OMD
4.0	o	CERTIFICATION/VERIFICATION STATUS
5.0	o	STS-51L (STS-33) (SRM-25) PERFORMANCE PREDICTIONS
6.0	o	TECHNICAL ISSUES
7.0	o	FLIGHT READINESS REVIEW

STS-61C (STS( ) (SRM-24)
PERFORMANCE
(CHART NO. 1-1)

PROBLEM	CONCERN	RESOLUTION	TO BE DISCUSSED
TBD			

PC 053077

VIII-C

Organizational Issuance

NASA
National Aeronautics and Space Administration

George C. Marshall Space Flight Center
Marshall Space Flight Center, Alabama 35812

Effective Date: December 29, 1983	Number: SOP 8000.1
Organization: Shuttle Projects Office	

Subject: SHUTTLE PROJECT FLIGHT READINESS REVIEW

1. PURPOSE

This procedure defines the responsibilities, requirements, and procedures to insure effective planning for and conduct of the Shuttle Projects Flight Readiness Review.

2. SCOPE

This procedure is applicable to all STS missions.

3. POLICY

a. It is the policy of the Manager, Shuttle Projects to make an assessment of flight readiness of the Shuttle projects prior to each flight. This will be accomplished by a consolidated Flight Readiness Review (FRR) of all MSFC Shuttle Projects Office elements necessary for safe and successful conduct of the launch, flight, and post-landing operations. The review will be supported by all MSFC organizations which participate in MSFC Shuttle Projects activities.

b. The Shuttle Project FRR will be preceded by separate detailed readiness reviews (pre-FRR's) of individual elements by the prime contractor's and the element project offices, under the cognizance of the responsible Managers.

4. RESPONSIBILITIES

a. The conduct of the Shuttle Projects FRR is the responsibility of the Manager, Shuttle Projects or his designated representative.

b. The Program Plans and Management Systems Office is responsible for FRR scheduling, planning and requirements, coordinating the FRR agenda, FRR action items and action item closeouts, and preparing the readiness assessment and maintenance of all records associated therewith. The Program Plans and Management Systems Office will be the focal point with the JSC Level II National Space Transportation System (NSTS) Program Manager and the Level I FRR under the cognizance of the Director, Space Shuttle Operations.

PC 056354

73

c. The Project Managers will assure the effectivity of a detailed prime contractor pre-FRR.

d. The Project Managers will conduct FRR's to develop their readiness assessment and are responsible for the Shuttle Projects FRR briefing content in their particular area.

5. PRIME CONTRACTOR PRE-FRR

Project Managers will assure that each prime contractor conducts a pre-FRP in preparation for the Project Offices FRR. The contractors review shall be chaired by a level of management at least one level above the contractor Project Manager.

6. SHUTTLE ELEMENT FRR

a. The Project will conduct a FRR in preparation for the Shuttle Projects FRR. The respective Project Manager of element under review will serve as Chairman. The membership will consist of representatives from the following:

Shuttle Projects Office

S&E Directorate

Reliability and Quality Assurance

Safety

Contractors
USBI, MMC, Thiokol, Rocketdyne

b. Each Project Office will make the necessary conference arrangements, notify review members, designate secretary, prepare presentations, record and track action items, closures and retain a copy of the presentation material in the Project Office record file.

7. SHUTTLE PROJECTS FRR REQUIREMENTS

a. Review Concept: The Shuttle Projects FRR will employ a delta review concept from prior reviews and previous STS missions.

b. Schedule: The Shuttle Projects FRR will be held prior to the Center FRR.

c. Agenda: The major agenda items and responsibilities are:

(1)	Introduction	Programs Plans and Management Systems Office
(2)	Systems	Manager, Systems Management Office
(3)	External Tank	Manager, External Tank Project Office

PC 056335

74

(4)	Solid Rocket Booster	Manager, Solid Rocket Booster Project Office
(5)	Flight Engine	Manager, Flight Engine Project Office
(6)	Action Items/Readiness Poll	Program Plans and Management Systems Office

d. Presentation Emphasis: The presentation of agenda items will normally include a brief status summary with appropriate supporting detail on significant items and conclude with a readiness assessment. The presentation topics and scope should be developed from the Project FRR's and should:

(1) be that required to provide the Shuttle Projects Manager and Review Team with the information needed to make an independent judgement as to flight readiness;

(2) review recent significant resolved problems and prior flight anomalies when necessary to establish confidence;

(3) cover all problems, technical issues, open items, and constraints remaining to be resolved before the flight;

(4) establish the flight baseline configuration in terms of all significant changes since the last flight and/or applicable STS flight.

Within the above guidelines, the scope of the review should cover status, changes and issues in areas such as:

(1) Hardware/Software Anomalies, Failures including development and acceptance test failures
(2) Launch Commit Criteria
(3) Flight Plans/Rules
(4) Vehicle Checkout
(5) Shortages and Open Work
(6) Prior Flight Anomalies
(7) As-built Hardware Configuration versus Certified Hardware List
(8) Critical Item List (CIL)/Hazards
(9) Development, Qualification, and Reliability Testing (Certification/Verification)
(10) Waivers and Deviations
(11) Limited Life Components
(12) Launch Critical Spares
(13) Sneak Circuits
(14) Flight Margins
(15) PAS Assessment
(16) Safety
(17) Process Changes: Design, manufacturing, checkout and launch processing

PC 055836

e. Shuttle Projects FRR Membership: The Manager, Shuttle Projects will establish review membership and serve as Chairman. Membership will comprise representation from the following organizations:

Program/Project Offices

Science & Engineering Directorate

Reliability & Quality Assurance

Safety

Contractors
USBI, MMC, Thiokol, Rocketdyne

f. Documentation Requirements

(1) Presentations: Vugraphs will be used for FRR presentations and paper copies will be provided to the reviewing officials.

(2) Statement of Flight Readiness: Statement of Flight Readiness will be executed by all Project Managers and submitted at the conclusion of the review.

g. FRR Action Item/Open Item Closeout Reporting Requirements

(1) Subsequent to the conclusion of the Shuttle Projects FRR, a copy of assigned action items will be provided to each actionee by the Program Plans and Management Systems Office.

(2) The FRR secretary will track all action items and provide status to the Shuttle Projects Manager.

(3) Closeouts to FRR action items will be submitted to the Program Plans and Management Systems Office in writing and will state fully the basis for closeout, that is: action taken, results obtained, and determinations made. The Program Plans and Management Systems Office will submit closures to the Manager, Shuttle Projects or his designated representative for signature.

h. Procedures

(1) Single Points of Contact: FRR planning and procedures will be coordinated through a single point of contact in the Program Plans and Management Systems Office.

Individual element for points of contact will be designated by the responsible Project Manager and the names provided to the Program Plans and Management Systems Office. These individuals will be responsible for these duties outlined in paragraph 6.b.

(2) Guidelines: Approximately three weeks prior to the MSFC Shuttle Projects FRR, guidelines will be prepared by Program Plans and Management Systems Office and issued by the Manager, Shuttle Projects, establishing Projects FRR date, Shuttle Projects FRR date, with applicable membership, and any special requirements not covered by this procedure.

8. EFFECTIVE DATE

This procedure is effective on date of issue.

Robert E. Lindstrom
Manager, Shuttle Projects

SOP 8000.1

cc:
SA01/Dr. Luvingood
SA01/Mr. Askew
SA04/Mr. Harden
SA11/Mr. Lombardo
SA31/Mr. Bridwell
SA41/Mr. Mulloy
SA51/Mr. Taylor
SA71/Mr. Boze
SA81/Mr. Zoller
EA01/Mr. Kingsbury
EA01/Dr. Thomson
EE01/Mr. Hardy
EE11/Mr. Horton
EE21/Mr. Thomason
EE31/Mr. Nichols
EE51/Mr. Goetz
EG01/Mr. Brooks
EG03/Mr. Bunn
EB01/Mr. Bradford
EH01/Mr. Schwinghamer
ES01/Dr. Dessler
EP01/Mr. McCool
EP43/Mr. Worlund
EL01/Mr. Hopson
ED01/Dr. McDonough
ET01/Mr. Taylor
PD01/Mr. Marshall
TA01/Mr. Odom
NA01/Mr. Thomas
JA01/Mr. Downey
DA01/Dr. Lucas
DD01/Mr. Lee
DE01/Mr. Bethay
DR01/Mr. Sneed

PC 056889

VIII-D

National Aeronautics and
Space Administration

George C. Marshall Space Flight Center
Marshall Space Flight Center, Alabama
35812

NASA

Reply to Attn of: DA01

JAN 7 1986

TO: Distribution

FROM: DA01/W. R. Lucas

SUBJECT: MSFC Flight Readiness Review (FRR) Board for MSFC Elements for Mission 51-L

An MSFC FRR Board of senior MSFC management personnel will convene at 8:30 a.m. on January 13, 1986, in Building 4663, HOSC/MCR, to review and assess the readiness status of the MSFC Mission 51-L elements for flight.

This meeting will be held at the SECRET level and all documentation will be handled and presented in accordance with the NASA Security regulations. Attendance will be restricted and all Projects are to coordinate this activity through the Program Planning and Management Systems Office, Tom Staples, 3-0338.

The Center Board is composed of the following:

DA01/W. R. Lucas:	Chairman
DD01/T. J. Lee	Vice Chairman
EA01/J. E. Kingsbury	
DS01/F. A. Speer	
EG01/J. P. Madole	
JA01/J. A. Downey	
PA01/W. R. Marshall	
CS01/J. C. Walker	
SA76/S. G. Henderson:	Secretariat

Each project manager must certify the flight readiness of his hardware and present supporting rationale and data so the Board can independently assess the flight readiness. The Shuttle Projects Office manager is responsible for preparation and coordination of the meeting, presenting an overall assessment of flight readiness, recording of minutes and action items, and tracking action items for closure by the Review Board.

ATTACHMENT 2-9

PC 014983

Emphasis will be placed on safety of flight and mission success, including potential impact of prior flight anomalies; ground test anomalies; revisions to hardware, software, launch commit criteria, or redlines which have not been flight verified; revisions to SRM recovery risks since the previous flight; any waiver which has not been flight verified or which requires external approval; and any revisions to hazard or critical item lists. Issues, concerns, and risks should be clearly identified as well as methods of closure.

In an effort to minimize administrative control requirements associated with the dissemination of classified data, an effort is to be made to present classified information only through viewgraphs. If Projects elect to incorporate classified data within their handouts, it is the Project's responsibility to assure that the handouts are marked and handled in accordance with the NASA Security regulations. If any assistance is needed in this matter, please contact the MSFC Security Division at 3-4310.

A preliminary agenda is enclosed.

W. R. Lucas
Director

Enclosure

Distribution:
See Page 3

NSTS SPACE TRANSPORTATION SYSTEM PROJECTS

Copy

STS 51-L

LEVEL I

FLIGHT READINESS REVIEW

ARNOLD D. ALDRICH
JANUARY 15, 1986

ATTACHMENT 2-7

PC 014660

MSFC SPACE TRANSPORTATION SYSTEM PROJECTS

STS 51-L FLIGHT READINESS REVIEW

SOLID ROCKET BOOSTER

L. B. MULLOY
MANAGER, SOLID ROCKET BOOSTER
JANUARY 15, 1986

PC 014737

AGENDA

- 0 STS 61-C FLIGHT ANOMALIES
- 0 MAJOR ISSUES/PROBLEMS
- 0 MAJOR CONFIGURATION DIFFERENCES
- 0 CERTIFICATION/VERIFICATION STATUS
- 0 READINESS STATEMENT

SRB-2

ANOMALIES/PROBLEMS

- O NO 61-C FLIGHT ANOMALIES
- O NO MAJOR PROBLEMS OR ISSUES

SRB-3

PC 014740

STS 51-L

SIGNIFICANT CONFIGURATION DIFFERENCE

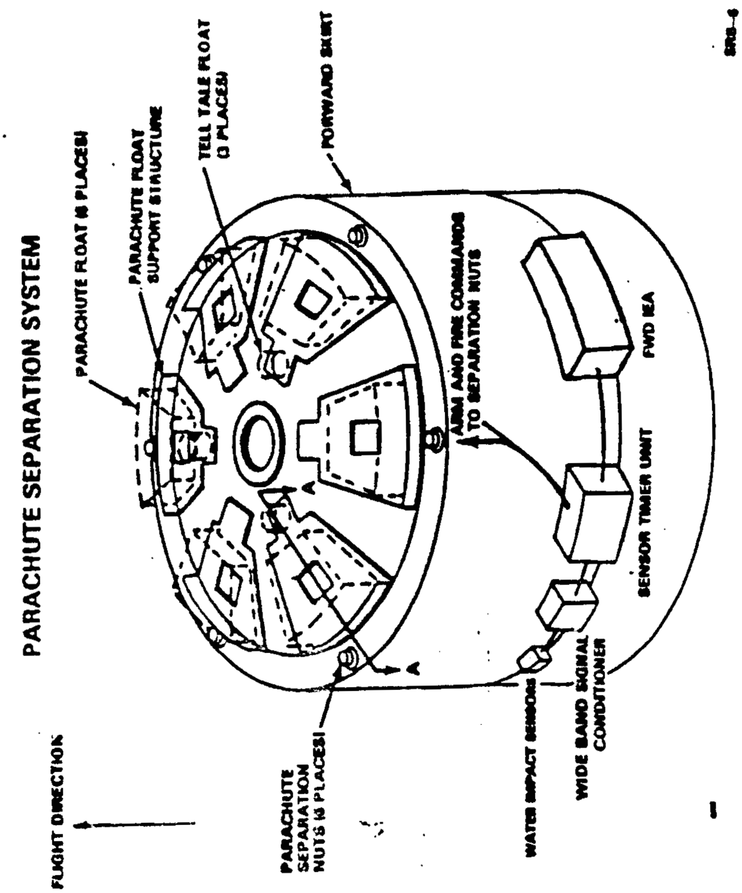
CHANGE	REASON	BASIS FOR CERTIFICATION
IMPLEMENTS CAPABILITY TO SEPARATE SRM NOZZLE AT APOGEE AND TO SEPARATE MAIN PARACHUTES AT WATER IMPACT	O NOZZLE SEPARATION O PRECLUDE FRAGMENTS FROM NOZZLE EXTENSION DESTROYING DROGUE PARACHUTE	O ANALYSIS - STRUCTURAL - DYNAMIC - FMEA
	O REDUCE DESIGN LOAD EXCEEDANCES ON MAIN PARACHUTE SYSTEM	O QUALIFICATION TEST
	O MAIN PARACHUTE SEPARATION	O FORMAL PDR/CDR
	O SRB RETRIEVAL PERSONNEL SAFETY	O DCR

SRB-4

PC 014741

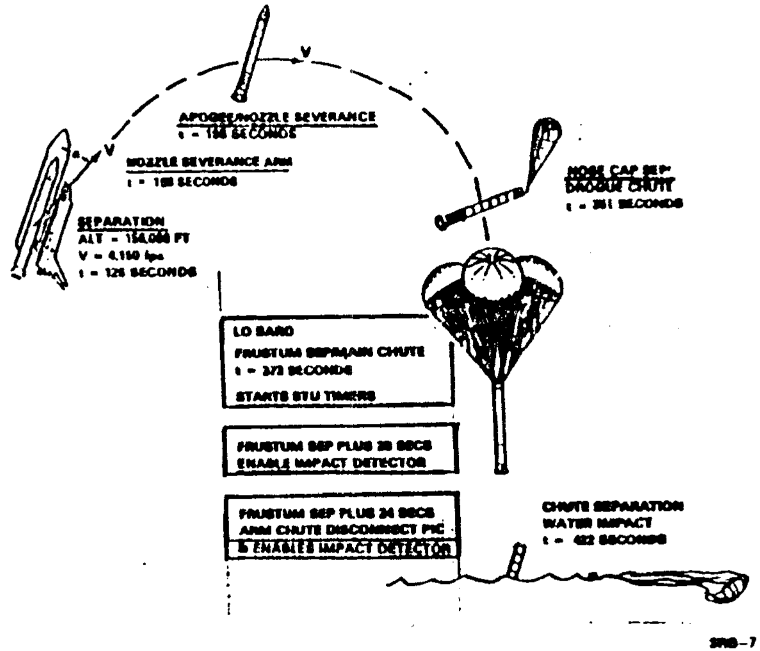
PC 014742

PARACHUTE SEPARATION SYSTEM



JAN 14 '86 11:16 AL-826/MSFC 4282 345A PAGE.05

NOMINAL SRB REENTRY PROFILE
(136 FOOT MAINS)



PC 014743

SUMMARY OF DESIGN CERTIFICATION

- O DESIGN ASSURES SINGLE FAILURE WILL NOT CAUSE PREMATURE SEPARATION EVENTS:

SEQUENCE	PROTECTION
SHUTTLE ENGINE/MOTOR IGNITION	STU NOT POWERED UP
STS BOOST PHASE	STU NOT POWERED UP
SRB SEPARATION	STU NOT POWERED UP
NOZZLE JETTISON	STU POWERED - PARACHUTE PIC'S NOT ARMED WATER IMPACT SENSORS NOT ENABLED, PARACHUTE SEP TIMER NOT STARTED
DROGUE DEPLOYMENT	STU POWERED - PARACHUTE PIC'S NOT ARMED, WATER IMPACT SENSORS NOT ENABLED, PARACHUTE SEP TIMER NOT STARTED
FRUSTUM SEPARATION	STU POWERED - PARACHUTE PIC'S NOT ARMED WATER IMPACT SENSORS NOT ENABLED, PARACHUTE SEP TIMER NOT STARTED

- O FMEA VERIFIES
  - O TWO OR MORE FAILURES REQUIRED FOR MALFUNCTION
  - O ALL PYRO EVENTS OCCUR PRIOR TO ARMING PIC'S FOR PARACHUTE SEPARATION
  - O ALL OTHER FUNCTIONS OCCUR PRIOR TO ENABLING SEPARATION CIRCUITS
  - O BOTH WATER IMPACT SENSORS MUST INITIATE SEPARATION SIGNAL
- O QUALIFICATION TESTS AND SYSTEMS CHECKOUT VERIFY PROPER SYSTEM OPERATION

SRB-8

PC 014745

CERTIFICATION/VERIFICATION STATUS

- NO OPEN CERTIFICATION OF QUALIFICATION
- NO OPEN HAZARDS/CIL'S
- NO OPEN WAIVERS
- NO OPEN UNPLANNED KSC WORK

PC 014746

READINESS STATEMENT

PENDING SATISFACTORY COMPLETION OF NORMAL OPERATIONS FLOW (OMRSD) AND OPEN ITEMS IDENTIFIED, WE CERTIFY THE SRB FLIGHT HARDWARE READY TO SUPPORT MISSION 51-L.

/S/ MR. KILMINSTER	/S/ MR. MURPHY
----------------------	----------------------
J. KILMINSTER, MTI VICE PRESIDENT SPACE BOOSTER PROGRAMS	G. MURPHY, USBI-BPC EXECUTIVE VICE PRESIDENT
/S/ MR. SMITH	/S/ MR. MULLOY
----------------------	----------------------
J. D. SMITH CHIEF ENGINEER, SRB PROJECT	L. B. MULLOY MANAGER, SRB PROJECT

SRB-10

PC 015243

VIII-F

NSTS
NATIONAL SPACE
TRANSPORTATION
SYSTEM OFFICE

STS 51-L

LEVEL II

FLIGHT READINESS REVIEW

ARNOLD D. ALDRICH
JANUARY 14, 1986

ATTACHMENT 2-10

MSFC
SHUTTLE
PROJECTS
OFFICE

STS 51-L FLIGHT READINESS REVIEW

SOLID ROCKET BOOSTER

L. B. MULLOY
MANAGER, SOLID ROCKET BOOSTER
JANUARY 15, 1986

AGENDA

- 0 STS 61-C FLIGHT ANOMALIES
- 0 MAJOR ISSUES/PROBLEMS
- 0 MAJOR CONFIGURATION DIFFERENCES
- 0 CERTIFICATION/VERIFICATION STATUS
- 0 READINESS STATEMENT

SRB-2

ANOMALIES/PROBLEMS

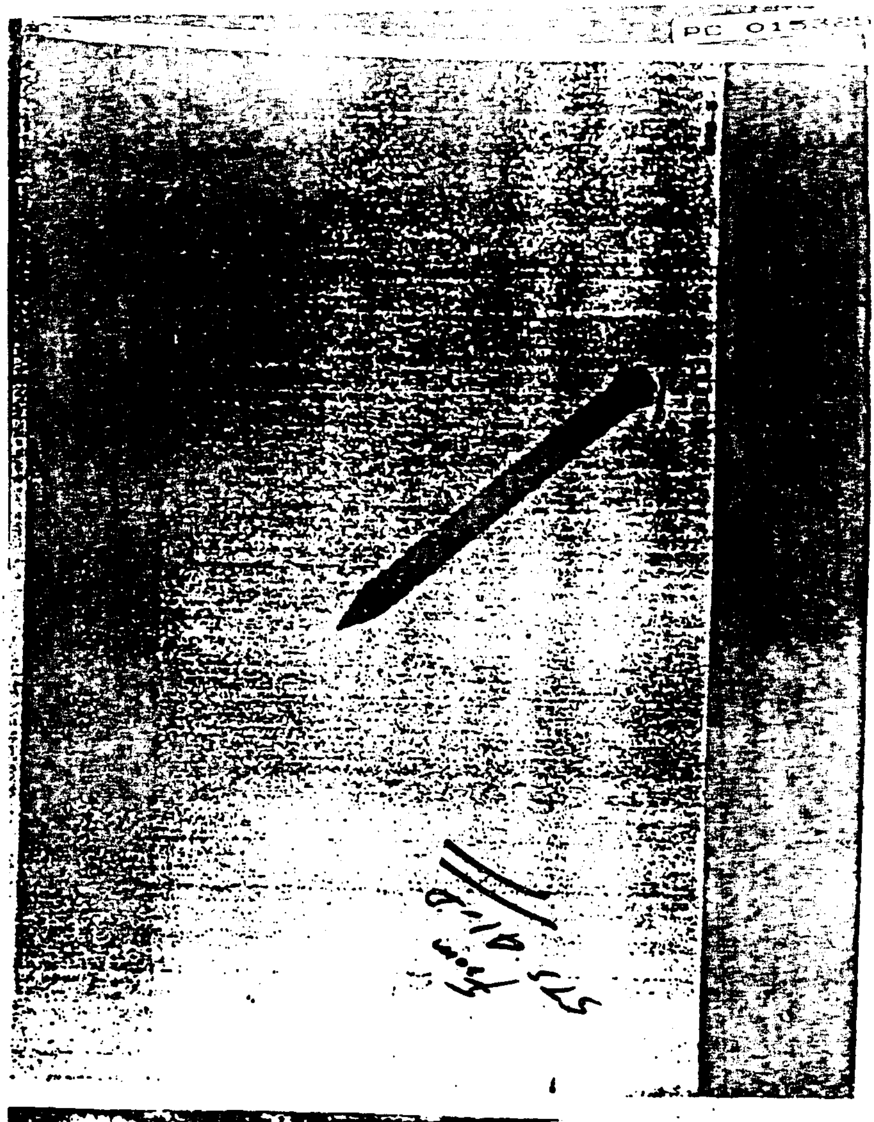
- NO 61-C FLIGHT ANOMALIES
- NO MAJOR PROBLEMS OR ISSUES

SRB-3

STS 51-L

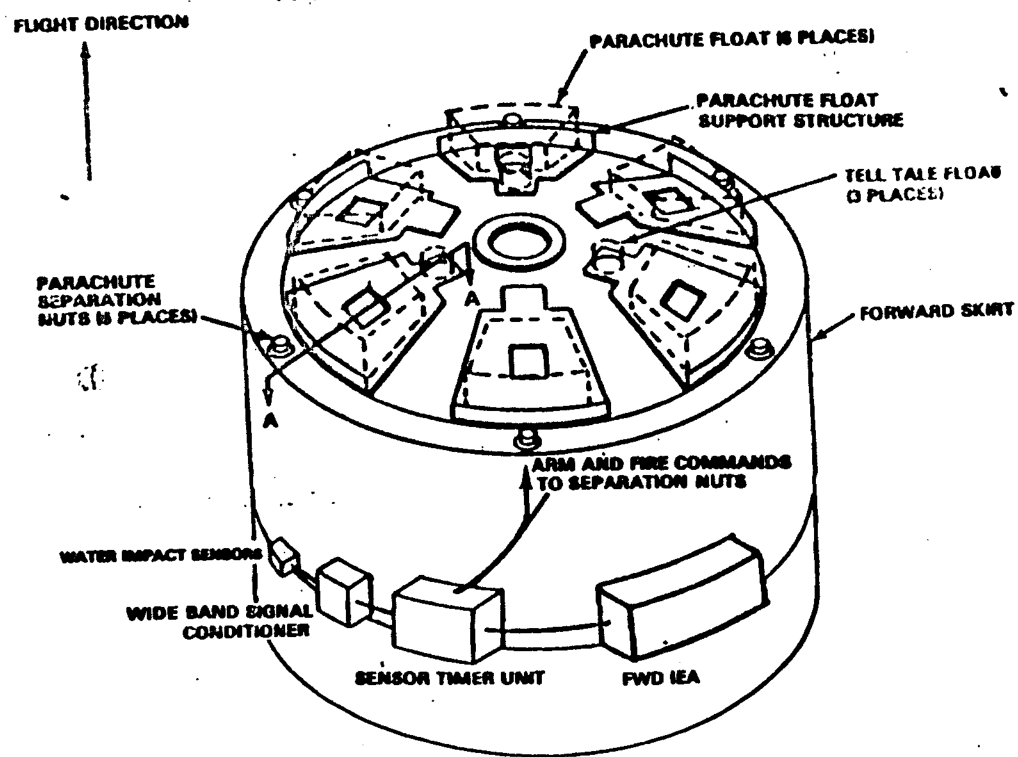
SIGNIFICANT CONFIGURATION DIFFERENCE

CHANGE	REASON	BASIS FOR CERTIFICATION
IMPLEMENTS CAPABILITY TO SEPARATE SRM NOZZLE AT APOGEE AND TO SEPARATE MAIN PARACHUTES AT WATER IMPACT	O NOZZLE SEPARATION O PRECLUDE FRAGMENTS FROM NOZZLE EXTENSION DESTROYING DROGUE PARACHUTE	O ANALYSIS - STRUCTURAL - DYNAMIC - FMEA
	O REDUCE DESIGN LOAD EXCEEDANCES ON MAIN PARACHUTE SYSTEM	O QUALIFICATION TEST
	O MAIN PARACHUTE SEPARATION	O FORMAL PDR/CDR
	O SRB RETRIEVAL PERSONNEL SAFETY	O DCR



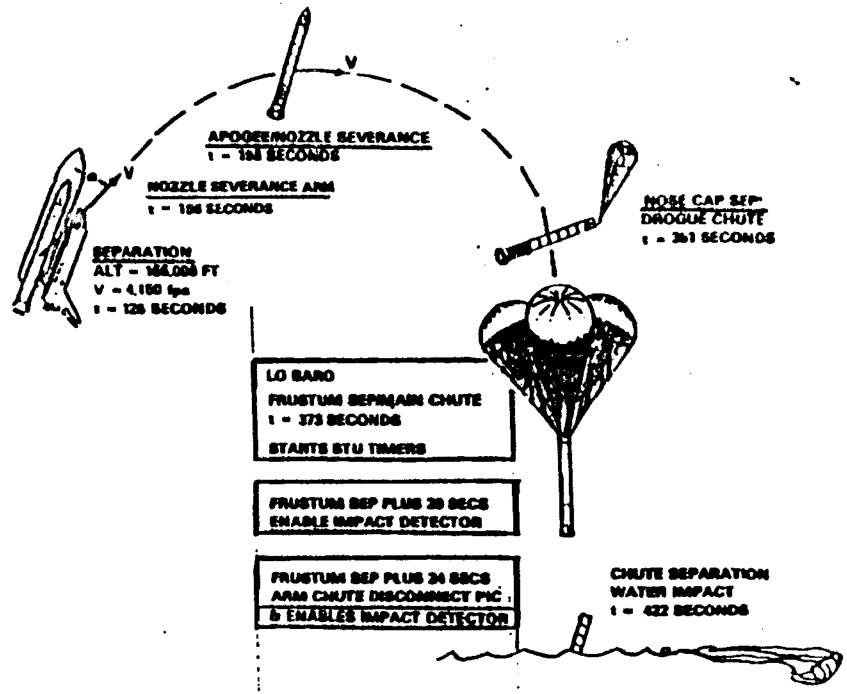
PC 015321

PARACHUTE SEPARATION SYSTEM



PC 015327

NOMINAL SRB REENTRY PROFILE
(136 FOOT MAINS)



SRB-7

SUMMARY OF DESIGN CERTIFICATION

O DESIGN ASSURES SINGLE FAILURE WILL NOT CAUSE PREMATURE SEPARATION EVENTS:

SEQUENCE	PROTECTION
SHUTTLE ENGINE/MOTOR IGNITION	STU NOT POWERED UP
STS BOOST PHASE	STU NOT POWERED UP
SRB SEPARATION	STU NOT POWERED UP
NOZZLE JETTISON	STU POWERED - PARACHUTE PIC'S NOT ARMED WATER IMPACT SENSORS NOT ENABLED, PARACHUTE SEP TIMER NOT STARTED
DROGUE DEPLOYMENT	STU POWERED - PARACHUTE PIC'S NOT ARMED, WATER IMPACT SENSORS NOT ENABLED, PARACHUTE SEP TIMER NOT STARTED
FRUSTUM SEPARATION	STU POWERED - PARACHUTE PIC'S NOT ARMED WATER IMPACT SENSORS NOT ENABLED, PARACHUTE SEP TIMER NOT STARTED

O FMEA VERIFIES
- O TWO OR MORE FAILURES REQUIRED FOR MALFUNCTION
- O ALL PYRO EVENTS OCCUR PRIOR TO ARMING PIC'S FOR PARACHUTE SEPARATION
- O ALL OTHER FUNCTIONS OCCUR PRIOR TO ENABLING SEPARATION CIRCUITS
- O BOTH WATER IMPACT SENSORS MUST INITIATE SEPARATION SIGNAL

O QUALIFICATION TESTS AND SYSTEMS CHECKOUT VERIFY PROPER SYSTEM OPERATION

SRB-8

CERTIFICATION/VERIFICATION STATUS

- NO OPEN CERTIFICATION OF QUALIFICATION
- NO OPEN HAZARDS/CIL'S
- NO OPEN WAIVERS
- NO OPEN UNPLANNED KSC WORK

SRB-9

READINESS STATEMENT

PENDING SATISFACTORY COMPLETION OF NORMAL OPERATIONS FLOW (OMRSD) AND OPEN ITEMS IDENTIFIED, WE CERTIFY THE SRB FLIGHT HARDWARE READY TO SUPPORT MISSION 51-L.

/S/ MR. KILMINSTER	/S/ MR. MURPHY
----------------------	----------------------
J. KILMINSTER, MTI	G. MURPHY, USBI-BPC
VICE PRESIDENT	EXECUTIVE VICE PRESIDENT
SPACE BOOSTER PROGRAMS	

/S/ MR. SMITH	/S/ MR. MULLOY
----------------------	----------------------
J. D. SMITH	L. B. MULLOY
CHIEF ENGINEER, SRB PROJECT	MANAGER, SRB PROJECT

SRB-10

VIII-G

TWR-15284C

FLIGHT READINESS REVIEW
SRM-24 (STS-61C)
02 DECEMBER 1985
L. O. WEAR

FLIGHT READINESS REVIEW
SRM-24 (STS-61C)

1.0	INTRODUCTION	MSFC
2.0	ELEMENT BOARD REVIEW	MSFC
	0 PROBLEM SUMMARY	
3.0	SRM-24 REVIEW	MTI
	0 STS-61B (SRM-23) PERFORMANCE	
	0 CHANGE SUMMARY	
	0 PERFORMANCE PREDICTION	
	0 CERTIFICATION/VERIFICATION	
	0 PROBLEM SUMMARY TOPICS	
	0 FLIGHT READINESS CERTIFICATION	

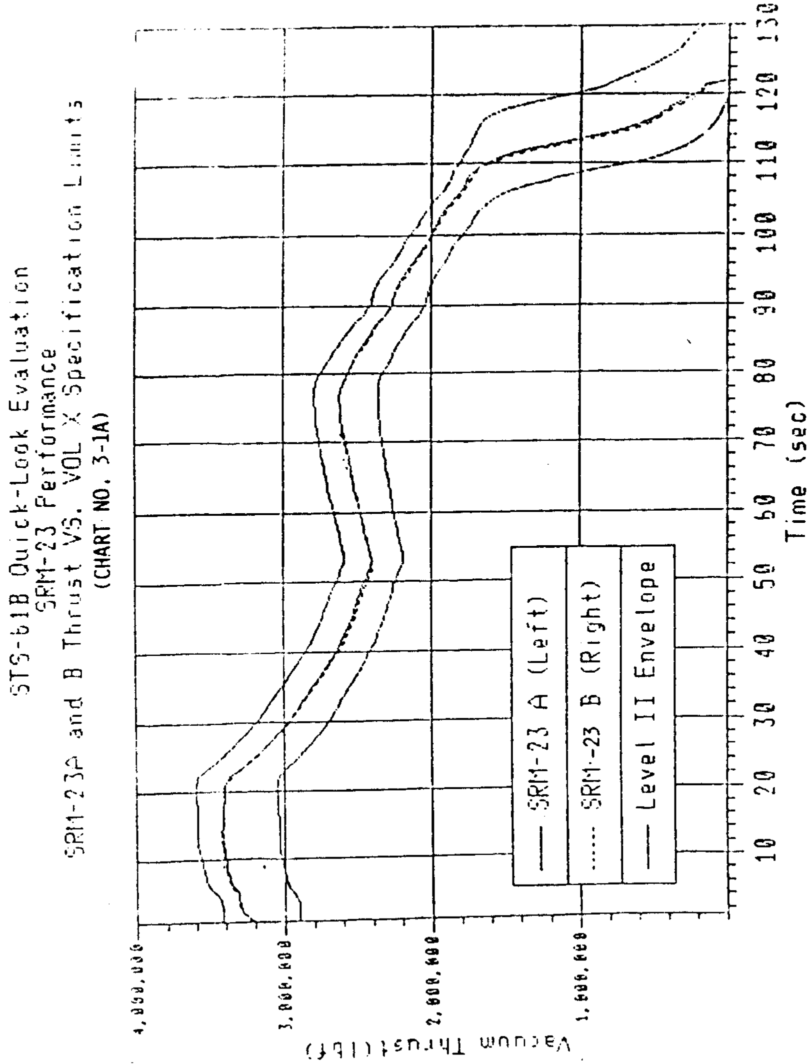
STS-61B QUICKLOOK EVALUATION
SRM-23 PERFORMANCE
(CHART NO. 3-1)

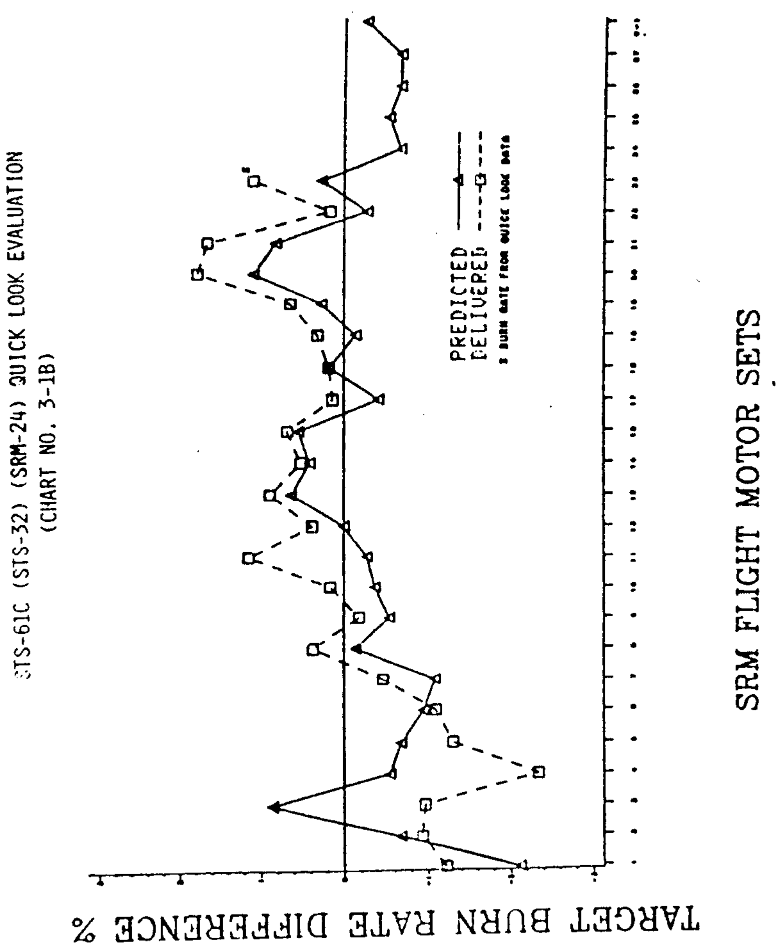
	PREDICTED (73°F)	EVALUATION (73°F)
o AVERAGE HEAD PRESSURE (5-20 SEC, PSIA)		
o LEFT MOTOR	901.0	926.2
o RIGHT MOTOR	899.7	928.7
o WEB TIME (SEC)		
o LEFT MOTOR	109.7	108.4
o RIGHT MOTOR	109.7	108.0
o ACTION TIME (SEC)		
o LEFT MOTOR	121.4	121.6
o RIGHT MOTOR	121.5	120.6
o SEPARATION COMMAND TIME BASED ON 50 PSIA ON LAST SRM + 5.7 SEC	124.8	123.9
o PROP. BURN RATE (IPS)		
o LEFT MOTOR	.372 (.369)*	.374 (.371)*
o RIGHT MOTOR	.372 (.369)*	.376 (.373)*

* ADJUSTED TO A TEMPERATURE OF 60°F AND PRESSURE OF 625 PSIA

INFORMATION ON THIS PAGE WAS PREPARED TO SUPPORT AN ORAL PRESENTATION

PC 05-53


STS-61B Quick-Look Evaluation
SRM-23 Performance
SRM-23A and B Thrust VS. VOL X Specification Limits
(CHART NO. 3-1A)
SRM-23 A (Left)
SRM-23 B (Right)
Level II Envelope
Time (sec)
Vacuum Thrust (lbf)
4,000,000
3,000,000
2,000,000
1,000,000
10
20
30
40
50
60
70
80
90
100
110
120
130


STS-61C (STS-32) (SRM-24) QUICK LOOK EVALUATION
(CHART NO. 3-1B)
PREDICTED
DELIVERED
BURN RATE FROM QUICK LOOK DATA
SRM FLIGHT MOTOR SETS
TARGET BURN RATE DIFFERENCE %

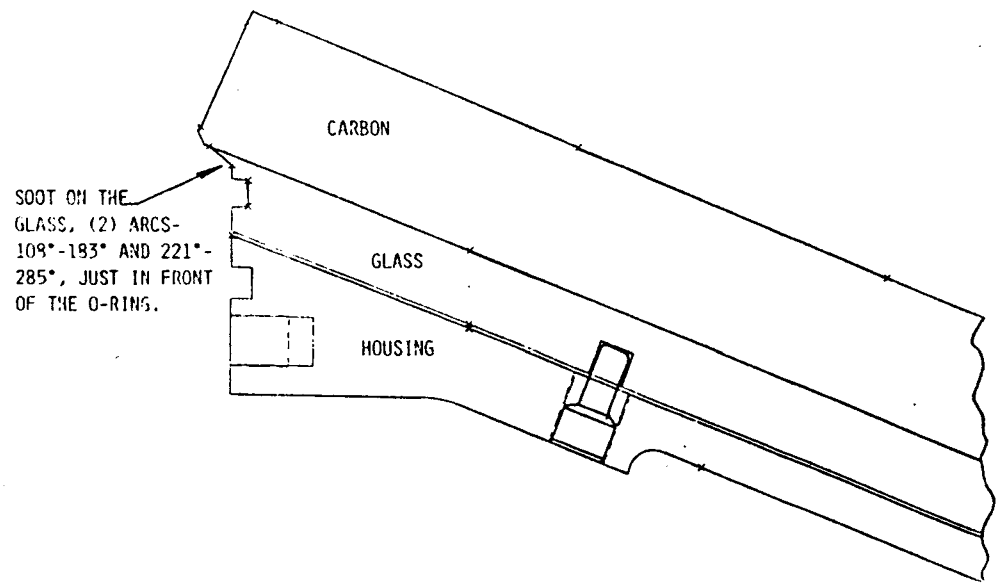
STS-61B (STS-31) (SRM-23) PERFORMANCE
(CHART NO. 3-2)

PROBLEM	CONCERN	RESOLUTION	TO BE DISCUSSED
ALL THREE STIFFENER RINGS ON BOTH SRM'S EXPERIENCED WATER IMPACT DAMAGE. CRACKS AND BUCKLES OBSERVED IN ALL THE RINGS. NINE OF THE 18 STIFFENER RING SEGMENTS WILL HAVE TO BE REPLACED.	REUSE OF SRM HARDWARE	ACCEPTABLE	NO
NO REPORTABLE POCKETS, GOUGES, OR WASHES IN THE INLET RINGS FOR BOTH NOZZLES.	SRM NOZZLE PERFORMANCE	ACCEPTABLE	NO
ALL ABLATIVE RINGS IN PLACE BUT SOME DISPLACEMENT FORWARD. APPARENTLY DUE TO WATER IMPACT, HAS CAUSED LOCAL RAISED AREAS AT SOME OF THE RING INTERFACES.	SRM NOZZLE PERFORMANCE	ACCEPTABLE	NO
SOOT OBSERVED ON SRM-23A (LH) AFT EXIT CONE SEALING SURFACE BUT THERE WAS NO APPARANT DAMAGE TO O-RING.	SRM NOZZLE PERFORMANCE	ACCEPTABLE	NO
SRM-23A (LH) HAD NO GAS PATHS THROUGH THE PUTTY AT ANY FIELD JOINT. (SEE SKETCH ON SUBSEQUENT CHART)	SRM PERFORMANCE	ACCEPTABLE	NO

PC 30051

STS-61B (STS-31) (SRM-23) POST FLIGHT
(CHART NO. 3-2A)

AFT EXIT CONE



STS-61B (STS-31) (SRM-23) PERFORMANCE
(CHART NO. 3-3)

PROBLEM	CONCERN	RESOLUTION	TO BE DISCUSSED
SRM-23B (RH) HAD ONE GAS PATH AT THE AFT/AFT CENTER JOINT AT 58° BUT IT DID NOT PROGRESS TO THE PRIMARY O-RING.	SRM PERFORMANCE	ACCEPTABLE	NO
LH MOTOR NOZZLE JOINT GAS PATH AT 340°. SOOT ON ALL SIDES OF O-RING GROOVE FROM 210.6° - 0° - 176.4° (316.8° TOTAL ARC). SOOT PAST PRIMARY SEAL FROM 325.8° - 354.6° (28.8° TOTAL ARC). SOOT PENETRATED ONLY ¼ INCH BEYOND PRIMARY SEAL. IT DID NOT TURN CORNER ONTO THE POLAR BOSS. PRIMARY O-RING DAMAGED TO UNKNOWN EXTENT SECONDARY O-RING NOT DAMAGED.	SRM PERFORMANCE	ACCEPTABLE	NO
RH MOTOR NOZZLE JOINT. GAS PATH AT 0°. SOOT BETWEEN PUTTY AND PRIMARY SEAL FROM 281° - 0° - 212° (291° TOTAL ARC) NO SOOT PAST PRIMARY SEAL. PRIMARY O-RING DAMAGED TO UNKNOWN RADIAL DEPTH. CIRCUMFERENTIAL LENGTH OF MAJOR ERODED REGION IS ABOUT 1 IN. SECONDARY O-RING NOT DAMAGED.	SRM PERFORMANCE	ACCEPTABLE	NO

PC 065514

Launch Constraint (2)

PC 037710

VIII-H

MSFC PROBLEM ASSESSMENT SYSTEM
FEBRUARY 26, 1986

PAGE 1 OF 3
MSFC RECORD NUMBER
A09288
CONTRACTOR REPORT NUMBER
DR4-5/49

PROBLEM TITLE: O-RING EROSION IN THE CASE TO NOZZLE JOINT

ELEMENT	CONTRACTOR	SYSTEM	CRIT	SYS LEVEL	MISC CODE	ABCDEFGHIJKLMNO
1	THIOKOL	STRUCTURE	1R	YES		2 X A67

PART IDENTIFICATION	NOMENCLATURE	PART NUMBER	SERIAL/LOT	MANUFACTURER
FAILURE ITEM	PRIMARY O-RING	1052557-00	SRM-L016A	HYDRAPAC
HIGHER ASSY	AFT SEGMENT	1050454-01	SRM-L016A	THIOKOL
ARTICLE	SRB ASSY	10101-0001	BI014L	THIOKOL

TEST / OP	PREV COND	LOCATION	SYMPTOM	FAIL/UNSAT	FAIL MODE	CAUSE	FMEA NUMBER
F	PT-FLT INS	KSC-K	123	UC	123	W20	10-01-01

LAUNCH CONSTRAINTS						CONCUR	TIME CYCLES	WUC	DATE OCCURED	DATE REC
STS - 51FL	51IL	51JL	61AL	61BL	61CL	JWT	N/A	A. MCDONALD	04/29/85	07/10/85

VEHICLE EFFECTIVITY	R/C	EST RES DATE	CONT CLOSURE REC.
51Y & SUBS	M20		12/10/85

PROBLEM DESCRIPTION:

REF: SIMILAR REPORTS - A07934, A08299, A08014, A08615, A08607, A08939, A09260
PRIMARY O-RING EROSION WAS EXPERIENCED ON THE AFT SEGMENT OF BOTH SRM 16A & B. SRM-16A HAD THREE DEEP EROSION POCKETS, THE GREATEST BEING 0.171 IN., WITH SOOT BETWEEN THE PRIMARY & SECONDARY O-RING & SOME EROSION ON THE SECONDARY O-RING. SRM-16B HAD MINOR EROSION, THE GREATEST POCKET BEING .005 IN., WITH NO SOOT BETWEEN THE PRIMARY & SECONDARY O-RINGS.

INVESTIGATION / RESOLUTION:

7/11/85 - POST FLT INSPECTION OF SRM 16A REVEALED A GAS PATH THROUGH THE VACUUM PUTTY AT 54 DEGS. THE PRIMARY O-RING HAD 12 IN. OF CIRCUMFERENTIAL (HOOP) LENGTH WHICH EXPERIENCED HEAT DEGRADATION AT THIS LOCATION. MAJOR EROSION OCCURRED ADJACENT TO THE HOLE IN THE PUTTY FOR 1.50 IN. IN THE CIRCUMFERENTIAL (HOOP) DIRECTION. WITHIN THE 1.52 IN. OF MAJOR EROSION THERE WERE THREE DEEP EROSION POCKETS. THE GREATEST AMOUNT OF EROSION WAS 0.171 IN. & THE GREATEST AMOUNT OF CROSS SECTIONAL PERIMETER ERODED WAS 332 DEGS. BETWEEN THE PRIMARY & SECONDARY O-RING THERE WAS HEAVY DEPOSIT OF SOOT FOR 90 DEGS TO EACH SIDE OF THE GAS PATH WITH A LIGHTER DEPOSIT FOR THE REMAINING 180 DEGS.

AT 54 DEGS THE SECONDARY O-RING EXPERIENCED 0.032 IN. OF EROSION & THE CROSS SECTIONAL PERIMETER WAS ERODED FOR 145 DEGS. THE CIRCUMFERENTIAL EROSION WAS 3.1 IN. WITH NO ADDITIONAL HEAT AFFECTED LENGTH. THERE WAS NO SOOT PAST THE SECONDARY O-RING.

POST FLT INSPECTION OF SRM 16B REVEALED A GAS PATH THROUGH THE VACUUM PUTTY AT 14.4 DEGS. AT THIS LOCATION THE PRIMARY O-RING WAS ERODED 0.005 IN. DEEP WITH 124 DEGS OF CROSS SECTIONAL PERIMETER ERODED. MAJOR EROSION EXTENDED FOR 3.4 IN. THE CIRCUMFERENTIAL DIRECTION WITHIN A HEAT AFFECTED LENGTH OF 4.7 IN. THE SECONDARY O-RING WAS UNAFFECTED.

BASED ON THE AMOUNT OF CHARRING, THE EROSION PATHS ON THE PRIMARY O-RING & WHAT IS UNDERSTOOD ABOUT THE EROSION PHENOMENON: IT IS BELIEVED THAT THE PRIMARY O-RING OF SRM 16A NEVER SEATED.

ORIG. / EVAL.	R. MCINTOSH DESIGN	O. NEWMAN R & Q.A.	L. WEAR PROJECT	J. FLETCHER PAC	PAC REVIEW COMPLETE JP
				MSFC CLOSURE DATE	STATUS OPEN-M

7/15/85 - THIS PROBLEM IS NOT CONSIDERED A CONSTRAINT TO 51F. A 100 PSIG LEAK CHECK IS PERFORMED WHICH CONFIRMS SEATING OF THE SECONDARY O-RING. NOZZLE O-RINGS HAVE BEEN SHOWN TO SURVIVE EROSION DEPTHS OF 125 MILS IN HOT SUBSCALE TESTS. THE MATH MODEL PREDICTS THE MAXIMUM EROSION DEPTH FOR THE SECONDARY O-RING TO BE 75 MILS IN THE WORST CASE CONDITION. THE EROSION ON THE SECONDARY O-RING WAS 32 MILS, WELL BELOW THE DEMONSTRATED SEALING CAPABILITY. ANALYTICAL RESULTS INDICATE THAT A WORST CASE EROSION CONDITION WOULD NOT ERODE THE SECONDARY O-RING BEYOND ITS SEALING CAPABILITY. SEAL CAVITY LEAK CHECK AT 100 PSIG CAN BE MASK BY THE VACUUM PUTTY, HOWEVER, AT 200 PSIG THE VACUUM PUTTY CANNOT MASK THE LEAK CHECK. STS-51F (SRM 19) HAS THE 200 PSIG LEAK CHECK AND THEREFORE SHOULD EXHIBIT BEHAVIOR MORE LIKE STS-51D (SRM-19) AND STS-51B (SRM 18) WHICH EXHIBITED PRIMARY O-RING EROSION WHICH WAS WITHIN EXPERIENCE BASE WITH NO SECONDARY O-RING EROSION. ALL JOINTS WITH VACUUM PUTTY WILL BE LEAK CHECKED TO 200 PSIG IN THE FUTURE.

REF. TWR 15080 "STS-51F SOLID ROCKET MOTOR (SRM 19) FLT READINESS REVIEW," DATED 7-1-85, SRM SPECIAL TOPIC, NOZZLE O-RING ANOMALY OBSERVED ON STS-51B (SR-16A) - ___JWT 7-19-85....

7/23/85 UPDATED FROM 7-11-85 PRB - RATIONALE FOR ACCEPTANCE WAS DISCUSSED AS INDICATED BY THE 7-2-85 UPDATE. NO OTHER SIGNIFICANT INFO WAS PROVIDED.

8/9/85 - STS-51F (SRM-19) FIELD JOINT O-RING INSPECTION REVEALED SEVERAL GAS PATHS THROUGH THE PUTTY BUT NO O-RING EROSION WAS FOUND ON ANY OF THE CASE JOINTS. THE NOZZLE TO CASE O-RING JOINT REVEALED A GAS PATH AT 21.6 DEGS WITH THE PRIMARY O-RING EXPERIENCING EROSION OF APPROX. 0.010 IN. THIS IS WITHIN THE FLT EXPERIENCE DATA BASE & IS NOT A FLT CONSTRAINT TO STS-51I (SRM-20) - ....JAMES W. THOMAS 8-11-85....

8/19/85 UPDATED FROM 8-8-85 PRB - NOTHING NEW TO REPORT AT THIS MEETING.

8/20/85 UPDATED FROM 21 DAY REPORT - RESULTS OF THE TEAM INVESTIGATION ARE THAT THE PRIMARY O-RING FAILED TO SEAT AT 54 DEGS WHEN THE GAS JET IMPINGED UPON IT. A PORTION OF THE GAS JET THEN PASSED THE PRIMARY O-RING & CONTINUED UNTIL IT IMPINGED ON THE SECONDARY O-RING. THE SECONDARY O-RING SEATED & SEALED THE JOINT. THE PRIMARY O-RING CONTINUED TO ERODE DUE TO A COMBINATION OF HOT GAS JET IMPINGEMENT & CONVECTION HEATING UNTIL THE IGNITION TRANSIENT WAS COMPLETED & THE MOTOR REACHED OPERATING PRESSURE APPROX. 600 MILLISECONDS AFTER IGNITION.

9/20/85 - ON STS-51I NO O-RING EROSION WAS EXPERIENCED THAT EXCEEDED THE CURRENT EXPERIENCE BASE. SOME PRIMARY O-RING EROSION WAS NOTED ON THE LH NOZZLE TO CASE JOINT. NO SOOT WAS FOUND BEHIND THE PRIMARY O-RING. THIS PROBLEM IS NOT A CONSTRAINT TO STS-51J - ....J.S.W.....

10/16/85 - ON STS-51J NO O-RING EROSION WAS EXPERIENCED THAT EXCEEDED THE CURRENT EXPERIENCE BASE. THIS PROBLEM IS NOT A CONSTRAINT TO STS-61A - ....JAMES W. THOMAS 10-17-85....

11/6/85 - ON STS-61A NO O-RING EROSION WAS EXPERIENCED THAT EXCEEDED THE CURRENT EXPERIENCE BASE. THIS PROBLEM IS NOT A CONSTRAINT TO STS-61B - ___ JWT 11-14-85....

12/10/85 - ON STS-61B NO O-RING EROSION WAS EXPERIENCED ON THE FIELD JOINTS. HOWEVER, MINOR O-RING EROSION WAS EXPERIENCED ON BOTH THE LH AND RH NOZZLE TO CASE JOINTS. SOOT BLOW-BY WAS NOTED ON THE LH NOZZLE JOINT. THE EROSION EXPERIENCED DURING THE STS-61B LAUNCH WAS WELL WITHIN THE CURRENT EXPERIENCE BASE. THIS PROBLEM IS NOT A CONSTRAINT TO STS-61C - ....JWT 12-12-85....

1/23/86 RESOLUTION - SRM FIELD AND NOZZLE JOINTS HAVE EXPERIENCED EROSION OF THE PRIMARY O-RINGS DURING SEVERAL MISSIONS AND STATIC TEST AS DETERMINED BY POST FLIGHT INSPECTION.

PC 037711

MSFC PROBLEM ASSESSMENT SYSTEM
(C O N T I N U A T I O N)

PAGE 3 OF 3
MSFC RECORD NUMBER
A09288
CONTRACTOR REPORT NUMBER
DR4-5/249

THE PRIMARY O-RING IN A NUMBER OF INSTANCES WAS SUFFICIENTLY DAMAGED AND/OR DISPLACED TO ALLOW BLOWBY. THEREFORE EXPOSING THE SECONDARY O-RING TO HOT COMBUSTION GASES. SOME EROSION WAS EXPERIENCED ON THE SECONDARY O-RING AT THE NOZZLE JOINT BUT NOT TO THE POINT WHERE THE O-RING SEALING CAPABILITY WAS JEOPARDIZED.

THE CAUSE OF THE O-RING EROSION HAS BEEN ATTRIBUTED TO HOT GAS PATHS THROUGH THE VACUUM PUTTY ALLOWING THE HOT GASES TO IMPINGE UPON THE O-RING. IMPROPER SEATING OF THE PRIMARY O-RING ALLOWED BLOWBY AND MINOR HOT GAS IMPINGEMENT ON THE SECONDARY O-RING. THE MAGNITUDE OF THE O-RING EROSION IS A FUNCTION OF ORIFICE SIZE AND FREE VOLUME (AREA BETWEEN THE PRIMARY O-RING AND THE PUTTY).

PRIMARY O-RING EROSION IS EXPECTED TO CONTINUE SINCE NO CORRECTIVE ACTION HAS BEEN ESTABLISHED THAT WILL PREVENT HOT GASES FROM REACHING THE PRIMARY O-RING CAVITY. STEPS HAVE BEEN TAKEN TO ASSURE THAT THE SECONDARY O-RING WILL BE SEATED AND ANALYTICAL ANALYSIS HAVE INDICATED THAT UNDER A WORST CASE SITUATION, EROSION OF THE SECONDARY O-RING WILL NOT BE SEVERE ENOUGH TO ALLOW A LEAK PATH PAST THE SECONDARY O-RING. STW7-2747 WAS CHANGED EFFECTING AN INCREASE IN THE LEAK CHECK PRESSURE FROM 100 PSIG TO 200 PSIG. WITH THE 100 PSIG LEAK TEST THE SECONDARY O-RING WOULD BE SEATED BUT ANY GAS PATHS THROUGH THE PUTTY WOULD BE UNDETECTABLE. BY INCREASING THE LEAK TEST PRESSURE TO 200 PSIG, ANY LEAK PATHS THROUGH THE PUTTY WOULD BE DETECTABLE. THE 200 PSIG LEAK TEST WAS IMPLEMENTED ON SRM-19 AND SUBSEQUENT MOTORS.

THE STACKING PROCEDURES HAVE BEEN CLARIFIED TO ASSURE MOTOR-TO-MOTOR CONSISTENCY IN JOINT MATING. THE OPERATIONAL MAINTENANCE DOCUMENT CHANGES WERE EFFECTIVE ON SRM L024 AND SUBS..

ANALYTICAL STUDIES BASED ON BOTH IMPINGEMENT EROSION AND BLOWBY EROSION SHOW THAT THIS PHENOMENON HAS AN ACCEPTABLE CEILING SINCE IMPLEMENTING THE ABOVE CHANGES. RECENT EXPERIENCE HAS BEEN WITHIN THE PROGRAM DATA BASE.

THIS SEAL IMPROVEMENT PROGRAM PLAN WILL CONTINUE UNTIL THE PROBLEM HAS BEEN ISOLATED AND DAMAGE ELIMINATED TO THE SRM SEALS. STATUS WILL CONTINUE TO BE PROVIDED IN THE FLIGHT READINESS REVIEWS AND IN FORMAL TECHNICAL REVIEWS AT MTI AND MSFC. AT THE CONCLUSION OF THE PROGRAM, A COMPREHENSIVE REPORT WILL BE WRITTEN TO CONSOLIDATE THE RESULTS, CONCLUSIONS, AND RECOMMENDATIONS.

THIS PROBLEM IS CONSIDERED CLOSED BASED ON MTI REPORT TWR-14359, REV. A, "IMPROVEMENTS OF SPACE SHUTTLE SRM MOTOR SEALS" DATED 8-30-85 AND MTI LETTER E150/BGR-86-114 "RATIONALE FOR CLOSURE OF THE O-RING EROSION PROBLEM", A07934, DR4-5/301 DATED 1-2-86.

constraint can be lifted by
Project Manager

PC 037712

VIII-I

PC 037596

PAGE 1 OF
MSFC RECORD
A09314
CONTRACTOR REF
DR4-5/35

MSFC PROBLEM ASSESSMENT SY
March 7, 1986

PROBLEM TITLE: SEGMENT JOINT PRIMARY O-RING CHARRED

ELEMENT	CONTRACTOR	SYSTEM	CRIT	SYS LEVEL	MISC CODE	ANOMALY
SRM	THIOKOL	STRUCTURE	1R	YES		O

HARDWARE IDENTIFICATION	NOMENCLATURE	PART NUMBER	SERIAL/LOT	MANUFACTURE
FAILED/ORIGINAL ITEM	PRIMARY O-RING	1U50228-24	SRM 0010A	THIOKOL
NEXT HIGHER ASSY	SRB ASSY	10101-0001	SRM 0010A	THIOKOL
TEST ARTICLE	SRB ASSY	10101-0001	BI010L	THIOKOL

TEST / OP	PREV COND	LOCATION	SYMPTOM	FAIL/UNSAT	FAIL MODE	CAUSE	FMEA NO
FLT	PI-FLT INS	KSC-K	1C1	UC	134	U20	10-01-C

LAUNCH CONSTRAINTS	CONCUR	TIME CYCLES	UOC	DATE OCCURED	DATE REC
N/L	JOB	N/A	R. LANEY	02/03/84	02/17/84

VEHICLE EFFECTIVITY	R/C	EST RES DATE	CONT CLOSUR
BI019 & SUBS	M20		12/18/85

PROBLEM DESCRIPTION:

REF: SIMILAR REPORTS; A00014 & A00299, A00615, A00607, A00939, A09360, A09208
SLIGHT CHAR CONDITION ON PRIMARY O-RING SEAL IN FWD FIELD JOINT ON SRM A57 OF STS-11 FLT, MISSION 41B.

INVESTIGATION / RESOLUTION:

REMEDIAL ACTION - NONE REQ'D; PROB OCCURRED DURING FLT. THE PRIMARY O-RING SEAL IN THE FWD FIELD JOINT EXHIBITED A CHARRED AREA APPROX 1 IN. LONG. 0.030-0.050 IN. DEEP, & 0.100 IN. WIDE. THIS WAS DISCOVERED DURING POST-FLT SEGMENT DISASSY AT KSC.

3/8/84 - POSSIBILITY EXIST FOR SOME O-RING EROSION ON FUTURE FLTS. ANALY INDICATES MAX. EROSION POSSIBLE IS 0.090 IN. ACCORDING TO THE FLT READINESS REVIEW FINDINGS FOR STS-13. LABORATORY TEST SHOWS SEALING INTEGRITY AT 3000 PSI USING AN O-RING WITH A SIMULATED EROSION DEPTH OF 0.095 IN., THEREFORE, THIS IS NOT A CONSTRAINT TO FUTURE LAUNCHES. ADDRESSED PRB ON 3-8-84.

3/27/84 UPDATED FROM 5 DAY REPORT - THE FAILURE DEFINED ABOVE WAS A SECONDARY PROB. THE VACUUM PUTTY LAY-UP IS CONSIDERED THE PRIME PROB. THE EFFECT ON THE O-RING SEAL BEING CHARRED HAS BEEN REVISED TO APPROX 3 IN. IN LENGTH, 0.040 IN. DEEP & AFFECTED 217 DEGS OF THE O-RING CROSS-SECTION PERIMETER. THE O-RINGS IN THE SRM SEGMENT ASSY JOINTS ARE DESIGNED AS PRESS. SEALS & ARE NOT INTENDED TO BE EXPOSED TO HOT GASES. STW4-2047 VACUUM SEAL PUTTY IS LAID UP IN THE JOINT AREA DURING MOTOR SEGMENT MATING TO PROVIDE A HEAT/HOT GAS BARRIER TO PROTECT THE O-RINGS. THE PRIME PROB IS THE LAY UP OF THE VACUUM PUTTY ALLOWING A HOT GAS PATH TO THE O-RING SEAL. THE O-RING IS USED IN THE FWD FIELD JOINT WHICH JOINED THE FWD MOTOR SEGMENT (P/N 1U52216-04, S/N 0000001) & FWD CENTER MOTOR SEGMENT (P/N 1U52217-03, S/N 0000001).

4/17/84 UPDATED FROM 4-12-84 PRB & MARCH MONTHLY SUMMARY REPORT - TEST FIXTURE & TEST PLANS HAVE BEEN PREPARED & TEST STARTED TO SIMULATE THE EFFECTS OF THE HOT GAS ON THE O-RING. THESE TESTS ARE EXPECTED TO CONFIRM THAT HOT GAS WILL NOT ERODE THE O-RING TO A POINT WHERE SEAL INTEGRITY WOULD FAIL. THE PUTTY LAY-UP IS STILL TO BE STUDIED.

CONCURRENCE	R. MCINTOSH	D. NEWMAN	L. WEAR	J. FLETCHER	PAC REVIEW COM
	DESIGN	R & Q.A.	PROJECT	PAC	JF
APPROVAL				MSFC CLOSURE DATE	STATUS OPEN-A

MSFC PROBLEM ASSESSMENT SYSTEM
(c o n t i n u a t i o n)

PAGE 2 OF 6
MSFC RECORD NUMB
A07934
CONTRACTOR REPOR
DR4-5/30

5/1/84 UPDATED FROM 21-DAY REPORT - THE FORWARD FIELD JOINT DESIGN IS BEING EVALUATED. IT HAS BEEN DETERMINED THAT THE FREE VOLUME, THAT PORTION OF THE JOINT FWD OF THE PRIMARY O-RING NOT FILLED WITH EITHER VACUUM PUTTY OR GREASE, IS WHERE THE PROB LIES. IT IS BELIEVED THAT THE O-RING CHAR IS CAUSED BY A SHORT TIME DURATION IMPINGEMENT OF A HIGH ENERGY JET WHICH IS INDUCED DURING IGNITION PRESSURIZATION BY A COMBINATION OF VOIDS IN THE PROTECTIVE VACUUM PUTTY & THE FILLING OF AVAILABLE FREE VOLUMES CREATED BY THE TOL. OF MATING PARTS & THE O-RING SLOT. THE TEST BEING CONDUCTED SHOULD VERIFY THAT THIS CONDITION WILL NOT CAUSE A SERIOUS PROB.

5/23/84 UPDATED FROM 5-10-84 PRB - INVESTIGATIONS ARE CONTINUING. O-RING DAMAGE IS NOT TO THE POINT WHERE SEAL INTEGRITY WOULD FAIL (LESS THAN 0.090 IN. POSSIBLE). O-RING CHAR HAS ALSO BEEN FOUND ON STS-13.

5/30/84 UPDATED FROM MAY MONTHLY SUMMARY REPORT - TWR-14336 "PERFORMANCE CHARACTERISTICS OF THE SRM O-RING ASSY TEST PLAN" HAS BEEN RELEASED & THE 5 IN. CP HOT TESTING INITIATED. DATA FROM THE TESTS IS BEING ANALYZED & WILL BE IN NEXT MONTH'S REPORT.

6/20/84 UPDATED FROM 6-14-84 PRB & JUNE MONTHLY SUMMARY REPORT - PRELIMINARY FINDINGS FROM THE O-RING TEST CONFIRM THE HOT JET IMPINGEMENT SCENARIO & THE MAJOR CONCLUSIONS DRAWN ARE:

1. O-RING EROSION IS A FUNCTION OF ORIFICE SIZE & FREE VOLUME.
2. GREASE ON THE O-RINGS MODERATELY REDUCED EROSION.
3. NO DAMAGE TO THE O-RING RESULTS WHEN THE ORIFICE IS ABSENT, SIMULATING THE ABSENCE OF VACUUM PUTTY.

A METHOD OF SOLUTION HAS NOT BEEN DETERMINED. THIOKOL WILL HAVE RECOMMENDED CHANGES IN JULY 1984.

A07934, A08014 & A08299 DOCUMENT DIFFERENT OCCURRENCES OF THE SAME GENERAL PROB. ALL ANALYSES DONE & CONCLUSIONS MADE WILL APPLY TO ALL 3 OCCURRENCE & WILL BE REPORTED UNDER A07934 HENCEFORTH.

7/10/84 ADDRESSED AT PRB ON 7-12-84 - TWO SPECS WERE RELEASED CHANGING THE LAYUP OF THE VACUUM PUTTY & GREASE APPLICATION. THESE CHANGES WERE TESTED SATISFACTORILY, HOWEVER ADDITIONAL TEST ARE REQ'D.

9/24/84 UPDATED FROM AUG. MONTHLY SUMMARY REPORT & 9-13-84 PRB - VACUUM SEAL PUTTY LAYUP SPECS STW7-3193 & STW7-3220 WITH NEW LAYUP PROCEDURES HAVE BEEN RELEASED. STW7-3194 IS STILL IN THE SIGN OFF PROCESS. INVESTIGATIONS ARE CONT'D. PLANS ARE TO PERFORM FULL SCALE NOZZLE & JOINT TESTS. ALSO LOOKING AT NEW PUTTY. THIS PROBLEM OCCURRED AGAIN ON FLT 41D, SEE PROBLEM REPORT A08615.

10/2/84 UPDATED FROM SEPT. MONTHLY SUMMARY REPORT - VACUUM SEAL PUTTY LAYUP SPEC STW7-3220 HAS BEEN REVISED TO REDUCE THE AMOUNT OF GREASE & STW7-3194 IS STILL IN SIGN OFF.

11/5/84 UPDATED FROM OCT. MONTHLY SUMMARY REPORT - THE JOINT FILLER MTRL TEST PLAN IS RELEASED & FULL SCALE JOINT ASSY TESTING WAS INITIATED 10-10-84 AT MORTION THIOKOL'S H-7 CLEARFIELD FACILITY.

11/14/84 UPDATED FROM PRB MTG 11-8-84 - THE TEST PLAN SHOULD BE REVISED & TEST STARTED WITHIN APPROX. 2 WKS.

1/2/85 UPDATED FROM PRB MTG 12-13-84 - A NEW PROGRAM WAS REQUESTED BY MSFC FOR EVALUATION & TESTING. THIS IS NOW IN WORK & THE ESTIMATED COMPLETION DATE IS 1-4-85. ANY DESIGN CHANGES ARE PENDING THE

PC 037597

TEST RESULTS. A COPY OF THE 40-LB & 5 IN. CP MOTOR TEST SCHEDULE IS INCLUDED IN THIS PAGE. THE O-RING CHAR PROBLEM IS A REVIEW ITEM FOR EACH FLT READINESS REVIEW (FRR).

1/22/85 UPDATED FROM PRB MTG ON 1-10-85 - ALL SCALE MODEL TESTING WILL BE COMPLETED IN JAN. FULL SCALE TESTS ARE TO FOLLOW.

2/15/85 UPDATED FROM 2-7-84 PRB & JAN. MONTHLY SUMMARY REPORT - TESTING OF VACUUM SEAL PUTTY LAYUP CONFIG. USING 5 IN. CP MOTOR IS CONTINUING. TORCH TESTS HAVE BEEN DONE ON ASBESTOS FILLED PUTTY MFR'D BY RANDOLPH & INMONT & A NONASBESTOS PUTTY MFR'D BY INMONT. BOTH ASBESTOS PUTTIES PERFORMED COMPARABLY. THE HEAT TRANSFER RATE OF THE NONASBESTOS PUTTY WAS MORE THAN TWICE THAT OF THE ASBESTOS FILLED PUTTY. FORTY LB. CHAR TEST MOTORS ARE SCHEDULED TO FURTHER TEST THE PUTTY TYPES.

EROSION WAS AGAIN EXPERIENCED ON FLT 51C. THE O-RING BURNS WERE AS BAD OR WORSE THAN PREVIOUSLY EXPERIENCED, SEE PROBLEM REPORT A09939/CONT. NO. 4-5/41. DESIGN CHANGES ARE PENDING TEST RESULTS. CHANGES BEING CONSIDERED ARE: MODIFYING THE O-RINGS, DOUBLE BARRIER VENTING & ADDING GREASE AROUND O-RINGS TO FILL THE VOID LEFT BY THE PUTTY.

3/18/85 UPDATED FROM 3-7-84 PRB - TESTING & EVALUATION PROGRAM IS CONTINUING. DESIGN CHANGES ARE PENDING TEST RESULTS.

3/29/85 UPDATED FROM FEB. MONTHLY SUMMARY REPORT - DISASSY OF THE SRM 15A & 15B NOZZLE TO AFT CASE JOINTS REVEALED GAS PATHS THRU THE PUTTY AT 109 DEG & 64 DEG, RESPECTIVELY. BOTH PRIMARY O-RINGS HAD HEAT AFFECTED AREAS (NO EROSION) AT THE GAS PATH LOCATIONS. SOOT WAS FOUND BETWEEN THE PRIMARY & SECONDARY O-RINGS IN EACH JOINT BUT THE SECONDARY O-RINGS WERE UNTOUCHED. THE SOOT WAS MOST LIKELY DEPOSITED DURING IGNITION PRESSURIZATION WHEN GAS PASSED THE PRIMARY O-RINGS BEFORE THEY SEATED.

A DR4-5 PROBLEM REPORT WAS NOT WRITTEN FOR THIS OCCURRENCE FOR THE FOLLOWING REASONS:

1) THE PRIMARY O-RINGS SEATED & THE PRESSURE INTEGRITY OF THE JOINTS WAS MAINTAINED.
2) THE O-RINGS WERE NOT ERODED.
3) GAS PATHS & O-RING EROSION HAVE OCCURRED ON PREVIOUS FLTS & ARE EXPECTED TO OCCUR RANDOMLY ON FUTURE FLTS.

IT HAS BEEN REPORTED HERE TO DOCUMENT THE OCCURRENCE.

4/17/85 UPDATED FROM PRB MEETING 4-11-85 - O-RING INVESTIGATION & PUTTY TESTING IS CONTINUING. MSFC'S FINDING THAT THE PUTTY DISSOLVED IN GREASE COULD NOT BE DUPLICATED BY THIOKOL.

4/24/85 UPDATED FROM APRIL MONTHLY SUMMARY REPORT - AT NASA REQUEST, A SOLUTION FOR O-RING EROSION WILL NOT INVOLVE A RADICAL DESIGN CHANGE. THEREFORE, THE POSSIBLE SOLUTIONS UNDER CURRENT INVESTIGATION ARE LIMITED TO:

1. NEW O-RING MTRL AND/OR DIAMETER.
2. NEW VACUUM PUTTY AND/OR LAYUP PROCEDURE.

6/6/85 UPDATED FROM MAY MONTHLY SUMMARY REPORT - DISASSY OF SRM 17A & 17B REVEALED GAS PATHS & EROSION IN BOTH NOZZLE TO CASE JOINTS. THE GAS PATH IN THE NOZZLE TO AFT CASE JOINT OF SRM 17A WAS CENTERED AT 14.4 DEGS & THE O-RING HAD A .010 IN. OF EROSION. SRM 17B'S NOZZLE JOINT HAD A GAS PATH AT 116 DEGS & THE O-RING HAD A .068 IN. OF EROSION. THERE WAS SOOT DEPOSITED IN THE PRIMARY

O-RING GROOVES OF EACH JOINT, HOWEVER, NO SOOT PASSED THE PRIMARY O-RING.

6/25/85 UPDATED FROM 6-13-85 PRB - PLAN TO SHAVE O-RING PAST THE .095 IN. TO DETERMINE WHEN A LEAK WILL OCCUR. HOT FIRE JOINT TESTS ARE PLANNED FOR AUG., 1985.

7/16/85 UPDATED FROM JUNE MONTHLY SUMMARY REPORT DATED 6-17-85 - ONE POSSIBLE EXPLANATION FOR THE GREATER EROSION DEPTHS OBSERVED ON SRB 17B (60 MILS) & NOW DM-7 (80 MILS) RELATES TO AN ASPECT OF NOZZLE JOINT ASSEMBLY: AN ECCENTRIC NOZZLE JOINT CAN OCCUR MORE EASILY THAN AN ECCENTRIC FIELD JOINT DUE TO A) ITS HORIZONTAL VS VERTICAL POSITION DURING ASSEMBLY, B) SMALL JOINT CLEARANCES, AND C) THE ABSENCE OF SHIMS. CONSEQUENTLY, IT IS POSTULATED THAT A REDUCED JOINT GAP (& THERE-FORE GAS JET ORIFICE) OCCURRED AT THE CIRCUMFERENTIAL LOCATION OF THE BLOWHOLE, THEREBY CAUSING ENHANCED EROSION. THE VOLUME FILLING MODEL PREDICTS THAT AN ECCENTRICITY OF ONLY 7 MILS WILL ROUGHLY DOUBLE THE 40 MILS EROSION PREDICTED FOR CONCENTRIC NOZZLE JOINTS. THUS AN O-RING EROSION EXCEEDING 80 MILS CAN BE CAUSED BY JOINT ECCENTRICITY IF THE REDUCED GAP ALIGNS WITH THE BLOWHOLE. INVESTIGATION OF THIS SCENARIO IS CONTINUING.

7/23/85 UPDATED FROM 7-11-85 PRB - THIS PROBLEM WAS SCHEDULED FOR STATUSING, BUT NOT DISCUSSED SINCE NO SIGNIFICANT PROGRESS HAS BEEN MADE SINCE THE LAST UPDATE.

8/19/85 UPDATED FROM 8-8-85 PRB & JULY 1985 MONTHLY SUMMARY REPORT - THE ANALYTICAL MODEL OF O-RING EROSION HAS BEEN USED TO PREDICT A WORST CASE EROSION OF 75 MILS ON THE SECONDARY O-RING SHOULD THE PRIMARY O-RING FAIL TO SEAL. PRESSURE TESTS OF O-RINGS WITH MTRL REMOVED TO SIMULATE EROSION & SUBSCALE MOTOR FIRINGS HAVE DEMONSTRATED THAT THIS IS WELL BELOW THE SEAL CAPABILITY OF ERODED O-RINGS. EVALUATION OF THE RESULTS OF 819-51F ARE CONTINUING.

9/3/85 UPDATED FROM JULY MONTHLY SUMMARY REPORT DATED 8-15-85 - DR4-5/48 (A09260) REPORTED .088 INCH OF EROSION OBSERVED IN THE NOZZLE TO CASE JOINT ON DEVELOPMENT MOTOR NO. 7.

DR4-5/49 (A09280) REPORTED .171 IN. OF EROSION ON PRIMARY O-RING & .032 IN. OF EROSION ON THE SECONDARY O-RING IN THE NOZZLE TO CASE JOINT OF SRM 16A.

THE INVESTIGATIONS ARE CONTINUING & WILL BE REPORTED HERE AS THEY ARE THE SAME GENERIC PROBLEM.

INSPECTION OF SRM 18A FIELD JOINTS REVEALED NO EROSION HOWEVER, THE NOZZLE TO CASE JOINT HAD PRIMARY O-RING EROSION AT 151.2 DEGS WITH A MAX. DEPTH OF .013 IN. SRM 18B HAD NO FIELD JOINT EROSION. THE NOZZLE TO CASE JOINT EXPERIENCED GAS PATHS AT 63, 93.6, 270 & 342 DEGS. THE PRIMARY O-RING WAS ERODED ONLY AT 270 & 342 DEGS WITH DEPTHS OF .009 & .023 IN., RESPECTIVELY. A PROBLEM REPORT WAS NOT WRITTEN FOR THESE OCCURRENCES SINCE THEY WERE WITHIN HISTORICAL LEVELS. THEY ARE REPORTED HERE FOR INFORMATION.

10/2/85 UPDATED FROM AUGUST MONTHLY SUMMARY REPORT DATED 9-16-85 - SRM 19B (STS-51F) EXPERIENCED A GAS PATH AT 21.6 DEGS. AT THIS LOCATION THE O-RING WAS ERODED APPROX. 0.001 IN., WELL WITHIN THE THE EXPERIENCE BASE. QM-5 WILL BE DELAYED TO ALLOW FOR DEVELOPMENT OF O-RING SOLUTION.

11/4/85 UPDATED FROM OCTOBER MONTHLY SUMMARY REPORT & 10-17-85 PRB - ANALYSIS OF DATA FROM STS-51J & QM-5 TEST ARE IN WORK.

11/19/85 UPDATED FROM 11-7-85 PRB MEETING. 81U-61A EXPERIENCED NO FIELD JOINT EROSION, HOWEVER, SOOT BLOWBY WAS FOUND ON 3 OF THE 6 PRIMARY O-RINGS. 818-61A R.H. NOZZLE EXPERIENCED EROSION OF THE PRIMARY O-RING. THE EROSION WAS WELL WITHIN THE CURRENT DATA BASE.

MSFC PROBLEM ASSESSMENT SYSTEM
(c o n t i n u a t i o n)

BASED ON THE JOINT PUTTY & O-RING PROBLEM, A PROGRAM WAS INITIATED TO ESTABLISH ACCEPTABLE MATERIALS & CONFIGURATION THAT COULD BE TESTED ON QM-5 & ULTIMATELY INCORPORATED IN FLIGHT MOTORS.

12/17/85 - UPDATE FROM NOVEMBER MONTHLY SUMMARY REPORT.
THE O-RING TASK FORCE HAS STARTED ASSEMBLY AND COLD FLOW TESTING OF NOZZLE PUTTY AND JOINT FILLERS. TESTING IS CONTINUING.

SRM 21 - NO DAMAGE
SRM 22 - RIGHT HAND (B) NOZZLE

THERE WAS EROSION AT 97.2 DEGREE. IT WAS .075 INCH DEEP AND HAD 162 DEGREE OF CROSS SECTIONAL PARIMETER AFFECTED WITH 13 INCHES OF HEAT DAMAGE LENGTH AND 23 INCHES OF HEAT AFFECTED LENGTH. THIS IS NOT THE WORST CASE TO DATE. THERE WAS NO OTHER REPORTED O-RING DAMAGE.

12/20/85 UPDATED FROM 12-12-85 PRB MEETING - THIOKOL IS CONTINUING STUDIES RELATIVE TO THE O-RING EROSION AND PUTTY LAYUP ON AN ON-GOING BASIS. A TOTAL SOLUTION TO THIS PROBLEM IS LONG RANGE; HOWEVER, AN ACCEPTABLE DATA BASE HAS BEEN ESTABLISHED AND SUFFICIENT ENHANCEMENT HAS BEEN MADE TO PROVIDE THE NECESSARY CONTROLS TO PREVENT RECURRENCE OF THE PROBLEM OUTSIDE THE ESTABLISHED DATA BASE. CLOSURE PAPER HAS BEEN PREPARED AND IS BEING EVALUATED.

1/20/86 UPDATED FROM DECEMBER MONTHLY SUMMARY REPORT. THE TASK FORCE HAD MADE ONE HOT GAS TEST. THE PRELIMINARY RESULTS INDICATE THAT THE TEST CHAMBER NEEDS TO BE REDESIGNED.

SPACE SHUTTLE SRM FLIGHT SET 23 WAS LAUNCHED NOVEMBER 27. INSPECTION OF THE FIELD JOINTS REVEALED NO DAMAGE. WHILE BOTH OF THE NOZZLE-TO-CASE JOINTS HAD EROSION:

	23A	23B
EROSION LENGTH (INCHES)	3.0	7.0
HEAT AFFECTED LENGTH (INCHES)	22.0	25.0
EROSION DEPTH (INCHES)	.017	.037
DEGREE LOCATION	346 DEGREES	0 DEG

1/23/86 RESOLUTION - SRM FIELD AND NOZZLE JOINTS HAVE EXPERIENCED EROSION OF THE PRIMARY O-RINGS DURING SEVERAL MISSIONS AND STATIC TEST AS DETERMINED BY POST FLIGHT INSPECTION. THE PRIMARY O-RING IN A NUMBER OF INSTANCES WAS SUFFICIENTLY DAMAGED AND/OR DISPLACED TO ALLOW BLOWBY. THEREFORE EXPOSING THE SECONDARY O-RING TO HOT COMBUSTION GASES. SOME EROSION WAS EXPERIENCED ON THE SECONDARY O-RING AT THE NOZZLE JOINT BUT NOT TO THE POINT WHERE THE O-RING SEALING CAPABILITY WAS JEOPARDIZED.
THE CAUSE OF THE O-RING EROSION HAS BEEN ATTRIBUTED TO HOT GAS PATHS THROUGH THE VACUUM PUTTY ALLOWING THE HOT GASES TO IMPINGE UPON THE O-RING. IMPROPER SEATING OF THE PRIMARY O-RING ALLOWED BLOWBY AND MINOR HOT GAS IMPINGEMENT ON THE SECONDARY O-RING. THE MAGNITUDE OF THE O-RING EROSION IS A FUNCTION OF ORIFICE SIZE AND FREE VOLUME (AREA BETWEEN THE PRIMARY O-RING AND THE PUTTY).

PRIMARY O-RING EROSION IS EXPECTED TO CONTINUE SINCE NO CORRECTIVE ACTION HAS BEEN ESTABLISHED THAT WILL PREVENT HOT GASES FROM REACHING THE PRIMARY O-RING CAVITY. STEPS HAVE BEEN TAKEN TO ASSURE THAT THE SECONDARY O-RING WILL BE SEATED AND ANALYTICAL ANALYSIS HAVE INDICATED THAT UNDER A WORST CASE SITUATION, EROSION OF THE SECONDARY O-RING WILL NOT BE SEVERE ENOUGH TO ALLOW A LEAK PATH PAST THE SECONDARY O-RING. STW7-2747 WAS CHANGED EFFECTING AN INCREASE IN THE LEAK CHECK PRESSURE FROM

100 PSIG TO 200 PSIG. WITH THE 100 PSIG LEAK TEST THE SECONDARY O-RING WOULD BE SEATED BUT ANY GAS PATHS THROUGH THE PUTTY WOULD BE UNDETECTABLE. BY INCREASING THE LEAK TEST PRESSURE TO 200 PSIG, ANY LEAK PATHS THROUGH THE PUTTY WOULD BE DETECTABLE. THE 200 PSIG LEAK TEST WAS IMPLEMENTED ON SRM-15 AND SUBSEQUENT MOTORS.

THE STACKING PROCEDURES HAVE BEEN CLARIFIED TO ASSURE MOTOR-TO-MOTOR CONSISTENCY IN JOINT MATING. THE OPERATIONAL MAINTENANCE DOCUMENT CHANGES WERE EFFECTIVE ON SRM LO24 AND SUBS.

ANALYTICAL STUDIES BASED ON BOTH IMPINGEMENT EROSION AND BLOWBY EROSION SHOW THAT THIS PHENOMENON HAS AN ACCEPTABLE CEILING SINCE IMPLEMENTING THE ABOVE CHANGES. RECENT EXPERIENCE HAS BEEN WITHIN THE PROGRAM DATA BASE.

THIS SEAL IMPROVEMENT PROGRAM PLAN WILL CONTINUE UNTIL THE PROBLEM HAS BEEN ISOLATED AND DAMAGE ELIMINATED TO THE SRM SEALS. STATUS WILL CONTINUE TO BE PROVIDED IN THE FLIGHT READINESS REVIEWS AND IN FORMAL TECHNICAL REVIEWS AT MTI AND MSFC. AT THE CONCLUSION OF THE PROGRAM, A COMPREHENSIVE REPORT WILL BE WRITTEN TO CONSOLIDATE THE RESULTS, CONCLUSIONS, AND RECOMMENDATIONS.

THIS PROBLEM IS CONSIDERED CLOSED BASED ON MTI REPORT TWR-14359, REV. A, "IMPROVEMENTS OF SPACE SHUTTLE SRM MOTOR SEALS" DATED 8-30-85 AND MTI LETTER E150/SSR-86-114 "RATIONALE FOR CLOSURE OF THE O-RING EROSION PROBLEM", A07934, DR4-5/30) DATED 1-2-86.

PC 037601

VIII-J

George C. Marshall Space Flight Center
Marshall Space Flight Center, Alabama
35812

SEP 1 5 1980

Reply to Attn of: SA01

TO: Distribution

FROM: SA01/Mr. Lindstrom

SUBJECT: Assigning Launch Constraints on Open Problems Submitted to MSFC PAS

The Shuttle Projects Office has established a requirement for identification of launch constraints for problems being reported to the MSFC Problem Assessment System (PAS) by element contractors. Each element contractor (Rocketdyne, MMC, USBI, and Thiokol) has been directed to support this requirement by providing launch constraint information on each new problem submitted to the MSFC PAS. The launch constraint information provided by the contractor is based upon their preliminary technical evaluation and will require final concurrence by the responsible element project manager.

a. The following guidelines have been established to aid in making constraint decisions on open problems and are limited to recurrence control determination only. In accordance with practices established on past programs, remedial actions (e.g. removal and replacement of defective hardware, etc.) for correcting discrepancies on the vehicle to be launched are considered launch constraints and are tracked by the KSC system.

(1) All open problems coded criticality 1, 1R, 2, or 2R will be considered launch constraints until resolved (recurrence control established and its implementation effectivity determined) or sufficient rationale, i.e., different configuration, etc., exists to conclude that this problem will not occur on the flight vehicle during prelaunch, launch, or flight.

(2) Problems coded criticality 3 will not be considered launch constraints unless (a) the potential exists of leading to a criticality 1 or 2 failure mode; or (b) the failed component has multiple use on the element and more than one occurrence could lead to a criticality 2 condition; or (c) the failure could result in multiple loss of flight instrumentation channels. If a criticality 3 is determined to be a launch constraint, it will be treated the same as a.(1) above.

b. To assure that each reported problem is reviewed for correct criticality and constraint assignment by the appropriate MSFC personnel, the following procedure will be followed:

(1) The responsible S&E design actionee and the element project office actionee will review each problem upon receipt to assure that the criticality and constraint assignments meet with their approval. Exceptions to criticality or constraint assignment will be coordinated with the Problem Assessment Center (PAC) actionee within two working days from receipt of the problem report.

(2) The Problem Assessment Center will prepare a weekly constraints list by element. This list will be submitted to the Shuttle R&QA Support Office, EG03, for input to the Shuttle Projects Manager, SA01. Copies of the constraint list will be furnished concurrently to each Shuttle Element Project Manager.

(3) Launch constraints will be reviewed at each Problem Review Board (PRB) meeting.

The Problem Assessment Center will be responsible for coordinating all launch constraint activity and assuring that information is properly documented in the Problem Assessment System (PAS) data base and transmitted to MSFC management.

Robert E. Lindstrom
Manager
Shuttle Projects Office

Distribution:
See page 3

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R. Jackson/J. Gross/R. Riemer
EG23/E. Hoard/D. Herda/D. Gladish/A. Torruella
EG24/E. Lowe/J. Davis/E. Baker/B. Nunnelley
EH01/R. Schwinghamer
EL01/T. Isbell
EP01/A. McCool
EP13/C. Gaines/E. Engler/W. Harrison
EP14/H. Nein/R. White/R. Beck/R. Runkle/R. Dotson/W. Bell
EP21/O. Goetz/G. Smith/E. Jacobs
EP22/R. Rodgers
EP23/F. Pitsenberger/H. Stinson/C. Ratliff/D. Pryor/R. Counts
EP25/J. Hester/J. Niblett
EP31/G. Geller
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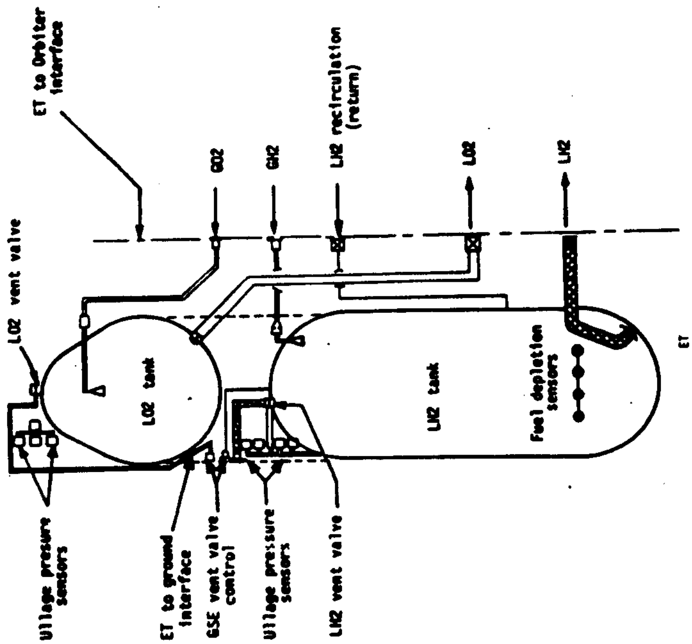
VIII-K

SECTION 5.1 EXTERNAL TANK SUBSYSTEMS

MML ID Nos. T41, T56, and Ground Measurement Nos. GGD, GLH, GLO

5.1-3 REV C



PC 001354

MEAS. NO.	MEASUREMENT DESCRIPTION	CAT.	REDLINE MINIMUM	REDLINE MAXIMUM	REDLINE UNITS	CODE	TIME PERIOD, ACTION, AND NOTES
T41T1[illegible]	Nose cone comp temp	1 of 2 N	45	135	°F	[illegible]	From Prior to Start of Chilldown to T-3 Min
T41T1[illegible]	Nose cone comp temp		45 (1)	135	°F	[illegible] (A)	(1) Minimum may be exceeded for up to 15 minutes.
							For confirmed instrumentation loss switch to backup and continue (A).

DRAWING, SCHEMATIC, OR FLOW DIAGRAM:

BIT VALUE	PCM RANGE LOW	PCM RANGE HIGH	METER RANGE LOW	METER RANGE HIGH	C AND W MIN	C AND W MAX	SM MIN	SM MAX
	-4	150						

REDLINE DERIVATION:
Redlines established for following reasons:
1. Minimum
   A. LO2 ullage pressure transducers calibrated to 40°F
   B. Avoid ice build-up at nose cone fairing exit
2. Maximum
   A. Protect CDF/LSC in cable tray area
   B. LO2 ullage pressure transducers Calibrated to 140°F
   C. SLA long term bond line exposure is 200°F

Min	Max	
40	140	Basic Redline
5	-5	ET Instr Error
45	135	Redline

CONSEQUENCES OF EXCEEDING REDLINE:
1. Ice build-up and possible impact to Orbiter
2. Inaccurate ullage pressure readings
3. Possible detonation of CDF/LSC and loss of vehicle
4. Debonding of SLA

CAUSES OF EXCEEDING REDLINE:
1. Heater malfunction.
2. Regulator malfunction.
3. Improper purge flow.
4. Pneumatic valve is closed.
5. Clogged filter.

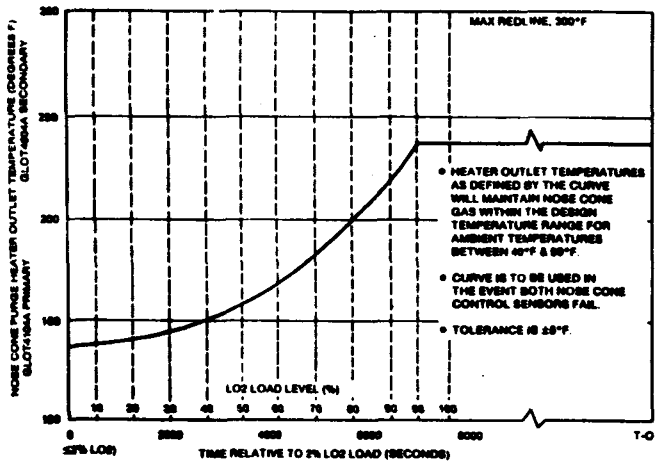
NOTES:
(A) The following ground purge temps are backup measurements
GLOT4100A PRI nose cone heated purge temp — See curve on
GLOT4004A SEC nose cone heated purge temp — page 5.1-4

SPACE SHUTTLE SYSTEMS HANDBOOK DWG NO. TBD SHEET ZONE TBD
DWG NO. SHEET ZONE
DWG NO. SHEET ZONE
INTEGRATED SYSTEMS SCHEMATIC: DWG NO. SHEET ZONE
EFFECTIVITY ELEMENT: ET MISSION: STS 51-G & SUBS. CONTACTS CONTRACTOR: [illegible], J. K. [illegible] CENTER: JSC, W. [illegible], KSC, H. [illegible] LAUNCH COMMIT CRITERIA AND BACKGROUND CHANGE NO., REV. & AMEND [illegible] PAGE 5.1-4

01424/0013A:

AAAAAAAAAAA

PC 001855



ET NOSE CONE PURGE HEATER OUTLET TEMPERATURE REQUIREMENT

PC 00185

VIII-L

SHUTTLE REQUIREMENTS EXCEPTION/WAIVER LIST STS —381 DATE:

NUMBER: WK0407
ELEMENT: ET (CLOH)
OMRSD TITLE: LCC PAGE 5.1-4 / VERIFY ET NOSE CONE PGE (OMRSD T00F00.552
EFFECTIVITY: 51 L & SUBS

REQT. NO.	DESCRIPTION	TECHNICAL RATIONALE
(PR # 35V-0247)	NOSE CONE TEMP. EXCEEDS THE LCC LOW LIMIT TEMP. S/B 45°F. TEMP INDICATES 33°F. [redacted].	NO VISABLE ICE BUILDUP ON THE NOSE CAP FAIRING EXIT AREA. TEMPERATURE IS 12°F BELOW REDLINE. FOR STS-33 MIN. LCC ACCEPTABLE IS 29°F (WAS 45°F) ULLAGE TRANSDUCERS ARE ACCEPTABLE DOWN TO 29°F (WAS 40°F) REFER. NOTE A LCC 5.1-4

PART NAME	DRAWING NUMBER	QTY & SERIAL/NUMBER
ET NOSE CONE HEATER (GT ANTI-ICE PANEL) 972-0694-17	79K07194	1

5686

APPROVALS

1/27/86

REQUESTED BY: PE 1/27/86 S. Reinartz 1/27/86

LSS 1/27/86 · DESIGN CENTER 1-27-86

R. Wahler 1/27/86 LSOC SYS ENGR · NASA SYS ENGR 1-27-86

LSOC SYS MGR 1/27/86 · NASA DEP DIR 1-27-86

LSOC PE 1/27/86 · NASA/PE (EXCEPTION ONLY)

SUSPENSE DATE: ________

CONSTRAINT: ________

KSC MAILBOX:
LEVEL III PROJECT OFFICE CONCURRENCE:
LEVEL II APPROVAL:

PC 000347