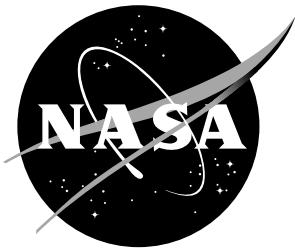


A History of Aerospace Problems, Their Solutions, Their Lessons

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TECHNICAL PAPER

A HISTORY OF AEROSPACE PROBLEMS, THEIR SOLUTIONS, THEIR LESSONS

I. INTRODUCTION

The author, along with the support of the Structures and Dynamics Laboratory, has made a major thrust to develop criteria, design guidelines, and “lessons learned” as a part of an effort to provide mentorship and to transfer knowledge to new NASA employees. A study of the problems experienced is a part of that effort. These criteria and guidelines should not become the layering of old laws or additional laws to protect against a failure or problems experienced, but should be overarching principles and guidelines that allow for creativity, innovation, and accountability by each and every member of a design and operations group. The study of problems serves as a basis for developing these high-level principles that guide the development of future programs. Problem study is not to be the negative finger pointing “look what you have done wrong,” but the beacon to illuminate “what to know and understand in order to build low-cost, robust aerospace systems of the future.”

This paper is an attempt to establish part of this needed knowledge foundation. It is based on many years of experience in both the early and present years of space exploration. This study is an expansion of information published in references 1, 2, 3, and 4. It is presented with the faith that some will be challenged to launch out into the deep. Only to the degree that the individuals of the aerospace community launch out into the depth of their specialties, yet maintain a bend toward being a generalist, an integrator, or a systemist, will the technology of the future aerospace (and possibly our Nation’s) be served.

The information presented has many sources. Much of it was developed by the various vehicle or spacecraft contractors. The information was obtained in general from briefings; however, some came from reports and papers (see reference list). The remainder of the information was produced by NASA engineers as they performed their tasks of managing the various project contracts, or conducting research and technology development. Given the vast scope of this paper, it is impossible to credit all the individuals who contributed; however, it is they, and not the author, who did the work, had the insight, solved the problems. The author’s task has been to collect the information and draw lessons from it. The author has been involved in all the projects presented, as well as most of the specific problems discussed. Therefore, the selection is generally limited to the author’s experiences and does not cover all NASA or Department of Defense (DOD) aerospace experience. In addition, the solutions set forth do not cover all the many disciplines of any given project.

The format used is first a discussion of the characteristics of each project, followed by a discussion of the technology development/creativity and innovation brought to bear as a result of design challenges and problems, along with highlights of key problems. This portion of the report is, therefore, a summary providing the reader with insight without the details. Second, the appendix contains a detailed listing of all problems/documents and a discussion of problems not previously documented. A selected few of previously documented major problems are included for completeness.

The approach for applying the study of problems to current projects is straightforward, but important. First, study the documented history of problems, their solutions, facilities, technology, and the resulting knowledge (theory, knowledge, guidelines) to understand the physics of the problem and the solution approach. Second, tailor the results to the individual project under consideration. A natural outcome of this tailoring is the further development and enhancement of models, simulations, test facilities, etc., that are keys to good design and are the basis for solving the complex problems that develop during the new product’s design and operation. The importance of having this knowledge, these tools, and facilities going into a new product cannot be overstated. Numerous examples can be given

where the program would have been canceled had not these technologies, tools, skills, and facilities existed. The advanced turbopump design (ATD) liquid oxygen (lox) pump for the space shuttle main engine (SSME) is a classic example. In this case, a deadline was given for the solution of the bearing wear problem or the program would be canceled. The skills of Government/industry/universities assuming a concurrent engineering team, the MSFC flow facilities, Pratt & Whitney's E-8 test stand and manufacturing unit, and the silicon nitride ball bearing technology developed at MSFC, were the contributing factors that ensured a solution in time to save the program. Future programs, therefore, require from the past and present:

- Human skills and knowledge
- Tools (analytical and computational)
- Facilities
- Enabling technologies.

From these come knowledge and design guidelines, etc., that serve as the foundation of future programs.

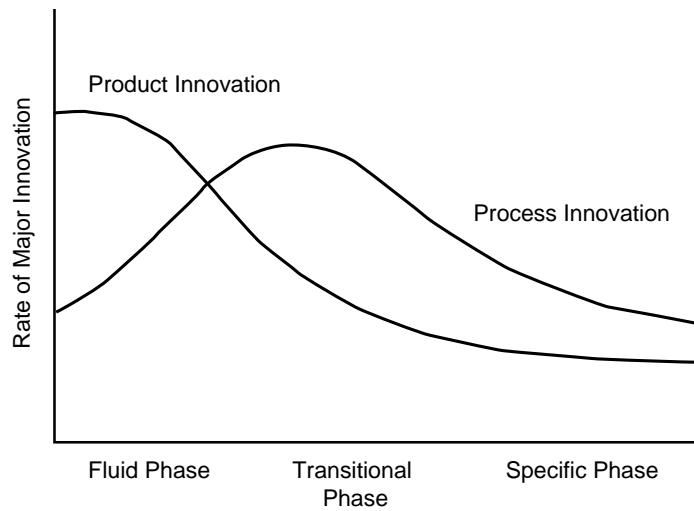
I would like to thank all who have helped pull together the information contained in this report. Each of the laboratory's project lead engineers put together the problem matrixes. Others have provided lessons learned and other key comments. I would like to thank Karen Spanner for a significant effort in pulling together the SSME flight readiness review data base and write-ups of some of the key SSME problems. Werhner Dahm and Harold Scofield made significant contributions to the sections on innovations and creativity.

II. GENERAL

A. Why Study Problems

The study of problems and their solutions is the foundation of the success of future space programs. The design and operation of these space programs are very complex and challenging, pushing the state of the art. They are made up of space vehicles, spacecraft, payloads, manufacturing facilities, and operations facilities (including communication systems) that continuously push state-of-the-art technology in engineering, management, and leadership. The systems being developed are highly tuned, balanced between competing requirements. This is true whether one is designing a turbopump, balancing the hydro output (pressure, flow) with vibration, stability, life, weight, and cost, or a launch vehicle's performance versus operational robustness, cost, schedule, etc. In general, these systems are high-performance driven, which leads to a "high-q" system (sensitive to small changes and highly tuned).

The role that problem analysis/study plays in a products success depends on the time phase the product is in. An excellent discussion of product time phasing by Utterback is in "The Dynamics of Innovation" in reference 5. There are two major observations in the book. The first has to do with shifting of emphasis from the product development (technology) to the process (fig 1). The second deals with the impacts of the invading radical, innovative technologies on the established products (fig. 2). Invading technologies cause the established product to accomplish bursts of improvement to counteract the invasion. For a period of time, this works; however, if the invading technologies are strong, they can win out and replace the product. As figure 2 indicates, parallel with this development is the shift from a technology to a process improvement and product standardization. As the product life is in its early phase where technology development/implementation is the focus, problem history and



Product	From high variety, to dominant design, to incremental innovation on standardized products
Process	Manufacturing progresses from heavy reliance on skilled labor and general-purpose equipment to specialized equipment tended by low-skilled labor
Organization	From entrepreneurial <i>organic</i> firm to hierachial <i>mechanistic</i> firm with defined tasks and procedures and few rewards for radical innovation
Market	From fragmented and unstable with diverse products and rapid feedback to commodity-like with largely undifferentiated products
Competition	From many small firms with unique products to an oligopoly of firms with similar products

Figure 1. The dynamics of innovation.

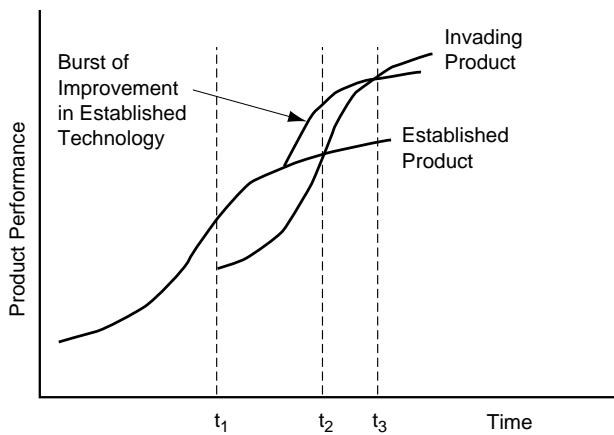


Figure 2. Performance of an established and an invading product (burst of improvement in established product).

understanding is very important, as it guides the technology development focus, including the determination of the technologies and systems needs. The problems experienced in past programs drive the technology program. The process of moving a product into an established phase requires technology focus by leadership and managers in order to understand problems, their source, their systems impacts, and the final product. As the product moves into the process-controlled phase, the emphasis is on fine tuning the technology/processes to achieve better operations, higher reliability, and lower cost. This process focus of leadership must be keenly aware of problem prevention. Problem history plays a major role also in this problem prevention of the product. The transition period between technology focus and process focus is also very critical in that during this time the product either makes it, or it does not. The role of problem history analysis here is clear.

Space exploration is certainly in the transition period where much of the products technology is at the standardized phase. This means that the emphasis is shifting to the process focus with cost, operations, and reliability playing key roles, yet space exploration requires high performance. High-performance systems require attention and care or problems erode the gain required for cost and operational efficiency. In order to maintain the high performance/low cost, new technologies must be developed to meet these goals.

Thinking about this product evolution and the role problem analysis/history plays, a key interrelationship emerges, that is required to solve development, operating, and new technologies inclusion. The key interrelationships are, in general, between people development (learning organization), tool development (facilities are included here), and technology development. Senge, in "The Fifth Discipline,"⁶ makes two observations. First, it is the tendency of a team to perform at the level of the member with the lowest IQ (i.e., skills); and second, with proper leadership and synergy, the team can perform higher than the member with the highest IQ (skills). The conclusion is that people skills, both technical and leadership, must be constantly developed. Style of organizational leadership starts at the top, technical leadership starts at the bottom. First line supervisors must necessarily be style and technical leaders. Style without technology is like an English teacher who writes eloquently but has nothing to say. Most of the time this learning is best accomplished by working and developing technology options first during the product emerging phase and then during the sustaining process controlled phase. It is also clear that if this second leg, technology, is not developed parallel with the product development, a time will come when the product will be lost without supporting technological solutions. The third interrelationship leg is tool development and maintenance. The tool leg is multifaceted, covering analysis, testing, information networks, facilities, and management. There are numerous examples in space exploration when simultaneous existing of these three legs saved a program (product). Figure 3 tries to symbolically capture these relationship organizational characteristics. The bottom line is that the organization must have two parallel emphases. One works on fine tuning, improving the current product. The second explores new technologies that can drastically alter the product to accomplish the task. For example, the typewriter replaced by the computer word processor. These, in general, are two specific groups of people who are not competing but working different roles. Space exploration must follow this path.

How does this relate to the study of problem history/analysis? Problem history/analysis plays at least two major roles. (1) It serves as a source of knowledge that relates in a way no academic process can, showing the depth and subtleties of a problem and system interactions/sensitivities. (2) It provides the road map/information for developing the technology gap analysis and, thus, the priorities and programs that develop the supporting technologies. Clearly, it could help to identify the competing technologies and to provide the basis for a product shift; however, history has shown that the competing technologies usually emerge from the outside.

Complications also arise because of the U.S. funding approach. Many problems are introduced as a result. Predominating is the stretching of funds and schedules that transfers issues from design to operations, greatly increasing operations cost and decreasing flexibility, etc. This means that the design in the end is always a balance between the multitude of conflicting parameters and requirements.

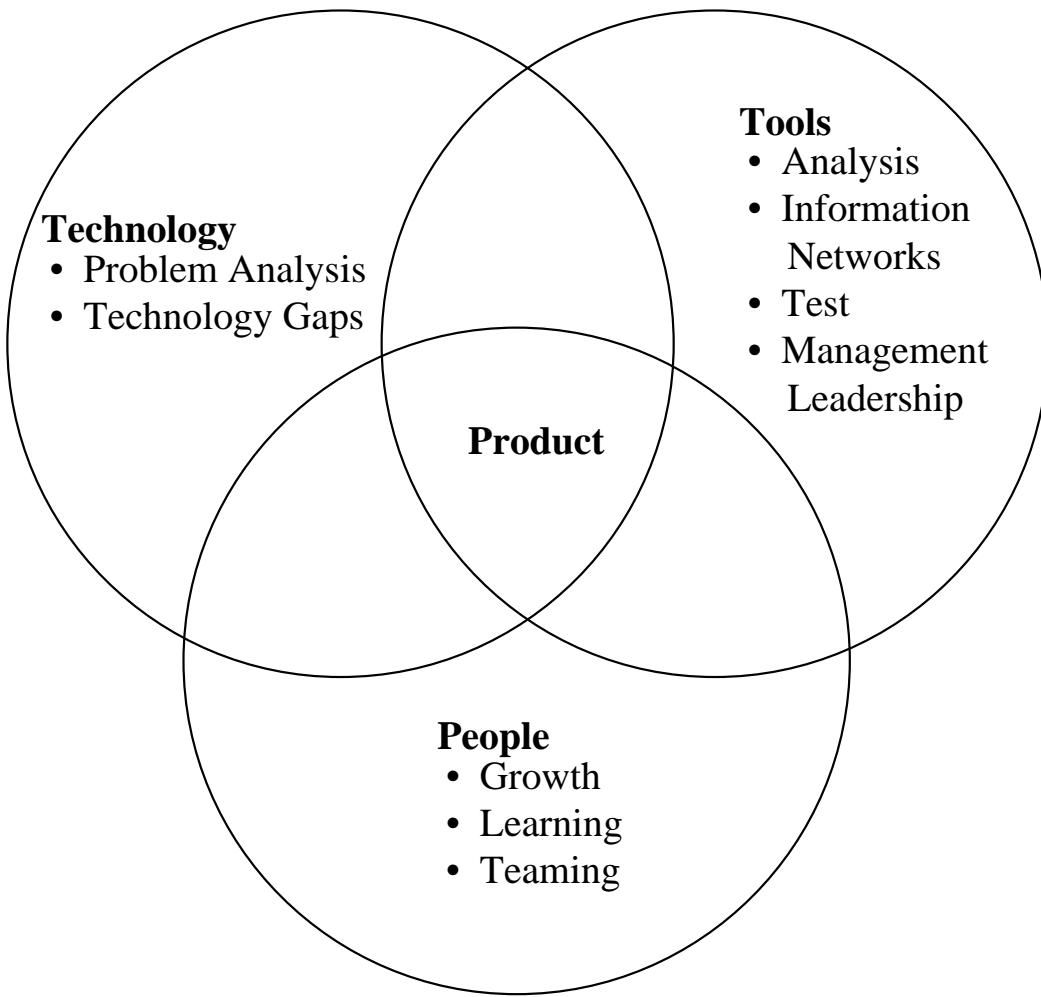


Figure 3. The organizational requirement for product success.

Gordon, in “Structures,” says: “All structures will be broken or destroyed in the end. Just as all people will die in the end. It is the purpose of medicine and engineering to postpone these occurrences for a decent interval; the question is: what is to be regarded as a decent interval.”⁷

Pye, in “The Nature of Design,”⁸ discusses the source of problems and these compromises. He talks about the source of problems dealing with the manifestation and transfer of energy. He says: “Any of these forms of energy is capable of producing changes, changes in things; more exactly, redistribution of matter. . . Now whenever a change is made by the passage of energy and a result is left, this event takes place in a group of things. Things are always together. They do not exist separately and they cannot act separately. . . When you put energy into a system you can never choose what kind of changes shall take place and what kind of results shall remain. . . All you can do, and that only within limits, is to regulate the amounts of the various changes. This you do by design.” He talks about the compromises in the design for failures. “The requirements for design conflict and cannot be reconciled. All designs for devices are in some degree failures. . . The designer or his client has to choose to what degree and where these shall be failures . . .”

Pugh, in “Total Design,”⁹ discusses in detail how the design problem is approached. He separates the task into parts where the concept selection and the design activities are the lynch pins of success. He states that outstanding design engineering can never right the shortcomings of poor concept selection, while the best concept selection can be destroyed by poor design engineering.

Petroski, in his book “Design Paradigms,”¹⁰ relates that design usually starts to analyze a problem in the middle based on past experiences and scaling, forgetting to start at the beginning. “Every solution of every design problem begins, no matter how tacitly, with a conception of how to obviate failure in all its potential manifestations.” He further discusses how changes are made without considering other effects that solving the immediate problem creates. Any design change can introduce new failure modes or bring into play latent failure modes. Changes must be analyzed with the objective of the original design, including all potential failure modes. Senge, in his notable work on systems,⁶ says that “If any small change is made to a system, the whole system must adjust to that change.”

As space exploration progresses into the future, the compromises will, in all likelihood, be magnified. Design of these systems will be based on processes that include much of what is embodied in the principles of total quality management (TQM), concurrent engineering, or their equivalent. This means not only a change in attitude, leadership, management, but also a change in methods such as the application of quality function deployment (QFD),¹¹ Taguchi,¹² robust design,¹³ simultaneous engineering,¹⁴ and just-in-time manufacturing.¹⁵ The key is the change in management and leadership, the others are merely tools to help the process. The challenge of the future, plus the promise of these new attitudes, new leadership, tools, and principles, creates excitement and interest as well as challenge and motivation. It should be seared into one’s mind, however, that the design can be no better than the knowledge and skills of the people brought to these systems. No organizational structures, tools, or facilities can supplant that. This means that even with concurrent engineering teams, TQM, etc., the team membership is critical, as well as both the team and institutional leadership. Leadership serves at least two functions: (1) to bring out the best creativity, knowledge, and design the team is capable of; and (2) to lead institutional setting of mission/vision, not the least of which is the development of the “total person” of every employee. Therefore, training and development of the key resource people is the answer. The strategic bomb of the future is the leveraging of knowledge. Aerospace and aeronautical engineering has the edge here.

Deming, the “guru” of quality, emphasizes that knowledge and theory are key ingredients to a better product. “You cannot see anything without knowledge . . . need theory to interpret data and learn . . . cannot observe what is right or wrong without theory and knowledge . . . what organizations need is not just good people, but people who are improving with education . . . knowledge increased advances in competitive positions have their roots in knowledge . . . theory is the basis for interpreting data, etc., and is the foundation of an organization . . . there is no substitute for knowledge of the system . . . tampering without knowledge creates problems (PS solving).”¹⁶

Drucker says in reference 17, “Management is about human beings. Its task is to make people capable of joint performance, to make their strengths effective and their weaknesses irrelevant. Management must also enable the enterprise and each of its members to grow and develop as needs and opportunity change. Every enterprise is a learning and teaching institution. Training and development must be built in on all levels . . . training and development that never stops.”

Depree writes in reference 18, “Leaders owe people space, space in the sense of freedom. Freedom in the sense of enabling our gifts to be exercised. We need to give each other the space to grow, to be ourselves, to exercise our diversity.”

As a project unfolds from concept through development, problems develop during the design, the development, and the operations of space systems. These problems have both a real-time and a long-term effect. Many times, in fact most of the time, analysis techniques and tools, testing techniques and tools, materials development and characterization, and manufacturing techniques and tools are not adequate to understand and solve the problem. Large efforts are expended to expand these technologies, understand the problems, and design fixes so that a successful program occurs. The next program assumes this same complexity, building more tools, criteria, and requirements instead of simplicity and robustness. We need to use this technology to emphasize robustness in the future systems.¹⁹ One of the most interesting aspects of the evolving culture (still present in aerospace engineering) was the

innovation/creativity engineers brought to the understanding and solution to problems that occur. This was particularly true before high-speed, large-capacity computers, when they could not brute force the computational analysis, but had to rely on sound physical principle and innovative ways of dealing with the problems. Numerous examples of creativity and innovation will be given throughout the paper.

As was discussed in other reports and papers by the author, because of high-performance requirements and programmatic constraints, the design, development, and implementation of these requirements have led to many development and operations problems. In addition, operational costs have sky-rocketed as decisions were pushed downstream. The question that arises is: "is the high cost due to major design process errors?" Probably not in the early days, since space was a complex evolving technology. Space exploration by its nature involves risks, costs, failures, etc. However, the industry has matured to the point where things can be done to reduce (not eliminate) costs. A better approach would consider all cost/program phases concurrently in order to reduce total cost. Figure 4 illustrates this concept that should be striven for.

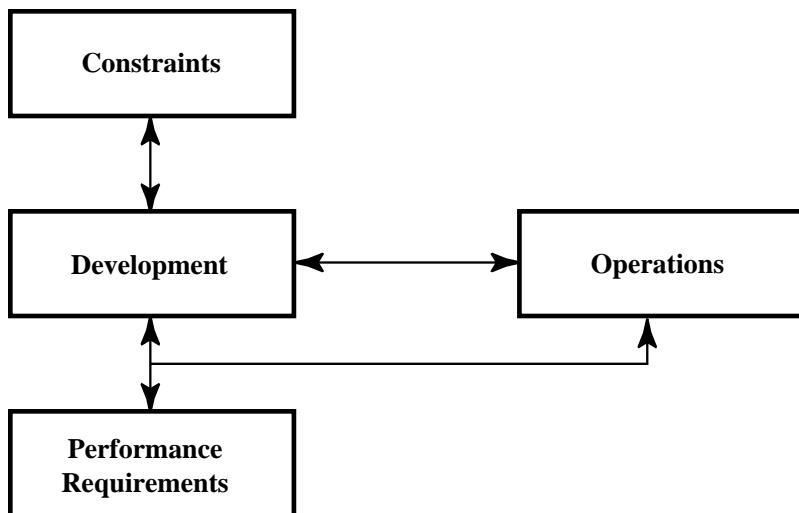


Figure 4. Systems approach.

Looking back at the history of aerospace, the evolution of the technologies commensurate with the evolution of space systems is apparent. Electronics have made quantum jumps in capability increase, size reduction, and reliability. Materials have evolved with high strength, fracture toughness, etc.. Manufacturing has made major advances through castings, composites, robotics etc. Structural analysis has moved from equivalent beam modal analysis to finite element models augmented by substructuring and modal coupling techniques. Computational fluid dynamics (CFD) and computational mechanics (CM) was necessary to solve complex flow and structural problems. Cross-discipline-integrated analysis tools have evolved, as well as have complex probabilistic approaches for controlling quality and assessing reliability. Fatigue and fracture control has made quantum jumps, as have thermal heat transfer analysis, pointing control techniques, structural control interaction, smart structures, and induced environment definitions. Criteria and standards have followed (in many cases becoming excessive and over constraining). Testing techniques from modal, vibration, electrical, thermal vacuum, flow, and aerodynamics have kept pace with many unique facilities developed. In the individual project sections, portions of this development will be discussed. As any new project moves through the development phase, new problems will occur. How these occurrences are handled determines success or failure. The tool that has evolved for coping with failures/problems is the use of concurrent engineering teams. Using formalized fault trees, logic diagrams, and action item closure has proven best.^{9 21 22} The authors have been members or leaders of several of these failure teams and can attest to the merits of the approach. It and the use of failure effects modes analysis (FEMA) in design are two common-sense innovations growing out of space exploration, and should be the working knowledge of every engineer.

Not only have engineers been innovative in solving technical problems, their innovation and creativity has taken many forms. It behooves management, as well as engineers, to utilize and enhance all these various expressions. The major one obviously is the innovative and creative engineering solutions just discussed and will comprise much of the rest of this report; however, the other expressions of creativity not only add humor but are many times better forms of communication than our formal, logical, analytical, scientific formats. To be associated with these expressive, atypical engineers who choose to use humor or descriptive titles and acronyms is one of the great highlights of our careers. Rich Holmen at McDonnell Douglas used humor through cartoons to capture personalities as well as engineering concepts during the Apollo and *Skylab* projects. McDonnell Douglas published some of them in a booklet titled “The First Five Years Are the Hardest or It Only Hurts At Resonance,” nicely drawn by Rich Holmen. The cartoon (fig. 5), “It Only Hurts at Resonance,” captures a problem we had on the Saturn rocket third stages (SIV B), as well as all subsequent projects in dynamics and control in a way we cannot forget. With dynamic resonances, you either understand and solve them or the price is very high. Design compromises to solve problems are a constant struggle. Holman’s cartoon (fig. 6) shows a design complication in order to get rid of a feedback pot, pictorially giving us the message.

In the Saturn Apollo days, Honeywell published yearly calendars with Bill Eddy’s cartoons highlighting each month. There are two bound books of these cartoons, which are not only humorous but give great messages in unforgettable ways. Two of my favorites deal with specifications (fig. 7) and accuracy (fig. 8). “So that decimal point was a fly speck!” brings home a poignant point, as did all the other calendars that not only made us better engineers technically, but also made us laugh a little, which relieved work pressures. Remember, each of these lessons was displayed on our office walls for a whole month, speaking to us. Then came the next month. Today we see planes, ships, electronics, etc., trying to sell us on a product.

The other area of creativity that could be either humorous or illustrative, or both, has been the naming of parts or acronyms. Figure 9 lists a few selected for illustration. In general, the name either describes the geometry or the physical characteristics of the phenomenon. The rest of the creative illustrations of this paper will deal with the creative engineering contributions made while addressing major design problems.



Figure 5. It only hurts at resonance.

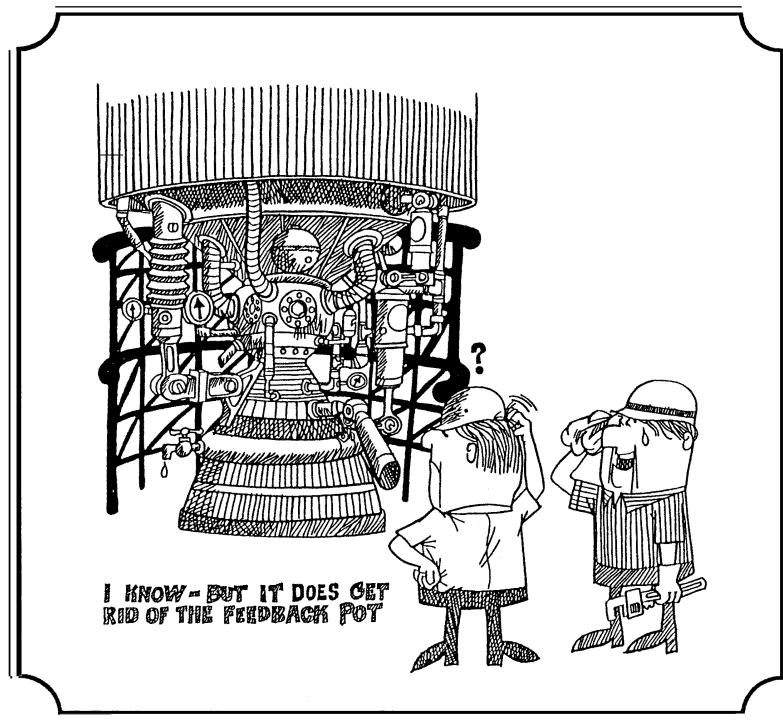


Figure 6. Design compromises.

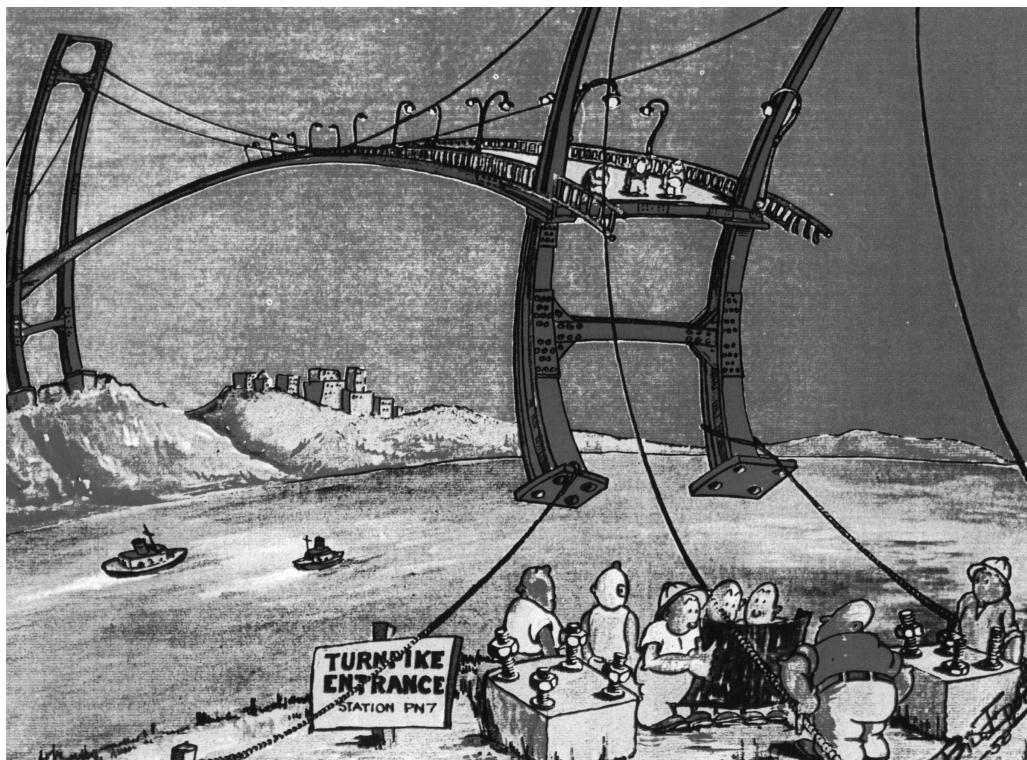
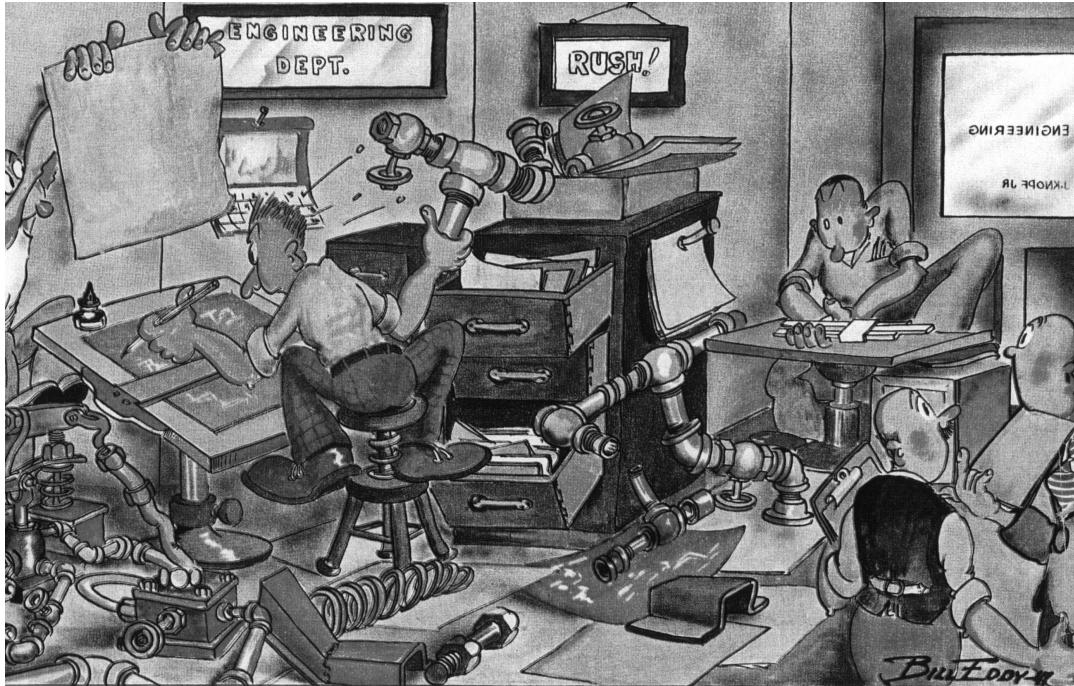


Figure 7. So that decimal point was a fly speck!



"I haven't the heart to tell him that the specs have changed"

Figure 8. Change in what specs?

Lessons in Propulsion System Design SSME Buzz Words

- "Steerhorn"-- Nozzle down comers
- "Baby pants"-- Heat exchanger bifurcation joint
- "Sea world"-- EDNI deposit bath for MCC's
- "Ski slope"-- HPOTP turbine end
- "Pop"-- Type of engine detonation in oxidizer and fuel preburner
- "Fir tree"-- Region of disk which blades are mounted, also region at base of blades
- "Kaiser hat"-- HPFTP turbine end entrance to turbine blades
- "Pony foot"-- Bottom of down comers
- "Coolie hat"-- HPOTP turbine end
- "Band weld"-- Nozzle jacket repair (EDNI)
- "Fish mouth"-- Seals in the pump
- "Fish mouth"-- Repair of the nozzle hot wall coolant tubes
- "Mixture bowl"-- Nozzle
- "Hot dog"-- Lox inlet manifold for lox dome
- "FRI"-- Flow restriction inhibitor for G15 seal
- "Turnaround duct"-- HPFTP turbine exit duct
- "Belly band"-- MCC throat shell
- "Pogo"-- Longitudinal vehicle oscillation due to propulsion system coupling
- "DOLIU"-- Day of launch I-loads update
- "Hex"-- Heat exchanger
- "T-bolts"-- Bolts to hold shields on lox post
- "Whirl"-- Rotary instability in turbo machinery
- "Z-baffle"-- Baffle in pogo accumulator
- "Buzz"-- High frequency vibration in SSME system
- "Beer cans"-- Jupiter anti-slosh devices
- "Crossbow seal"-- Seal in SSME turbo pumps
- "Grub worm"-- External Tank to launch vehicle connection

Figure 9. SSME "buzz" words.

B. Classification of Problems

Space vehicles and spacecraft are very complex; many are one of a kind. This complexity and uniqueness produces a variety of problems that occur during development and operations. Many of these problems have been published in technical papers or failure reports, with some given at professional conferences. Design reviews and flight readiness reviews also contain many examples. Reusability further complicates the design concerns; therefore, vehicles such as the space shuttle not only had development problems, but must deal with lifetime, maintenance, and operations problems that occur because of lifetime issues and other factors.² In attempting to summarize all these different problem sources, taking into account the project individual characteristics, it was decided to group the problems by project instead of by problem cause, as was done in reference 1. The problem causes or manifestations can be generally classified as one of the following:

1. Aeroelasticity
2. Instabilities (control, flow, structural, Pogo, etc.)
3. Fatigue, fracture mechanics, and wear
4. Environments (winds, contamination, temperature, etc.)
 - a. Natural
 - b. Induced
5. Response (forced and transient)
6. Modeling discrepancies
7. Acoustical tuning
8. Structural, control, fluid, modal tuning
9. Manufacturing/quality (process control)
10. Procedures.

The projects covered are:

1. Saturn Apollo, Saturn V Apollo
2. *Skylab*
3. Titan Viking
4. Jupiter—military and aerospace
5. Redstone—military and aerospace
6. Space shuttle
 - a. External tank (ET)

- b. Solid rocket booster/solid rocket motor (SRB/SRM)
 - c. Advanced solid rocket booster/advanced solid rocket motor (ASRB/ASRM)
 - d. SSME
1. Baseling
 2. ATD
7. Hubble Space Telescope (HST)
 8. Tether Satellite System (TSS)
 9. High Energy Astro Observatory (HEAO)
10. Spacelab
 - a. Multipurpose Experiment Support Structure (MPESS)
 - b. Office of Space Technology Applications Experiment No. 1 (OSTA-1)
 - c. Spacelab-1
 - d. Spacelab-2
 - e. Spacelab-3
 - f. D-1
 - g. Astro-1
 - h. Office of Aeronautics and Space Technology Experiment No. 1 (OAST-1)
 - I. Orbiting Materials Science Laboratory Experiment No. 1 (OMSL-1)
11. Upper stages
 - a. Inertial upper stage (IUS)
 - b. Transfer orbit stage (TOS)
12. Advanced X-Ray Astrophysics Facility (AXAF)
 - a. AXAF-I
 - b. AXAF-S
13. Space Station *Freedom* (S.S. *Freedom*)
 14. National Launch System (NLS)
 15. Furnaces (experiments)

16. IPS

17. Burst and Transient Source Experiment (BATSE).

C. Pictorial of Problems

1. Pictorial of Projects. NASA has had an evolution of projects since its inception in 1959. MSFC has been involved in most of these efforts. Figure 10 shows the Marshall project involvement, while figure 11 depicts the project evolution. Marshall's experience base involves science, upper stages, satellites, tethers, launch vehicles, propulsions systems, telescopes, manned stations, etc. These are encompassed by the outline given in section B. In the following sections, more information will be given on each of the projects.

2. Problem List. The appendix lists, by project and title, the various problems collected and studied. This should serve as a handy reference. Not all problems listed will be summarized and discussed in this document because they were documented in references 1 through 4. Some will be repeated in order to provide continuity and clarity.

It should be pointed out that not all problems, which are important to study and become lessons learned for the development phases, are failures. Many were discovered in the analytical and trade phases. In fact, it is better to find them in these phases and thus design out the problem. Some problems of this type are included. Also, the process of design provides lessons in terms of philosophy, approaches, and criteria inherent in successful projects. In general, it is left to the reader to develop his or her own list in these categories.

III. AEROSPACE PROJECT/PROGRAM DEVELOPMENT

A. Redstone and Jupiter

1. Characteristics. The Redstone was a military vehicle designed to deliver a nuclear warhead 200 mi downrange. It burned kerosene and lox and was designed to be a significant improvement over the German rocket V-2 technology, yet was basically a derivative of the V-2 designed by the German team transported to the United States after World War II. Dr. Wernher von Braun was the famed leader of this team, which contributed greatly to the U.S. Moon race by developing the Saturn launch vehicle. The airframe was integral with the propellant tanks. Figures 12a and 12b are pictures of the basic vehicle showing its general characteristics. Control was accomplished using jet vanes in the thrust exhaust stream, augmented with movable fins or rudders during the aerodynamic portion of flight.

The basic vehicle was modified to accept an upper stage for launching the first U.S. satellite, and later, to launch man in a lob flight as a part of the Mercury program. This program was called the Mercury Redstone. Also, this vehicle, in a flight testing reentry nose cone, launched a nose cone over 3,000 mi downrange. This launch created a furor and led to the Army being restricted to 200-mi vehicles. Therefore, the vehicle was very robust and successful.

The Jupiter was an intermediate range ballistic missile (IRBM) designed to deliver a warhead to a target 200 mi away. Figure 13 shows the basic vehicle and its characteristics. The Air Force developed comparable vehicles, namely Thor and Atlas. The Jupiter was modified and flew several satellite missions under the code name Juno. The technology of the Redstone and Jupiter was combined to build the first Saturn vehicles, which is discussed in the next chapter. Propulsion, tankage, tooling, etc., was adapted to the Saturn I and IB vehicles.

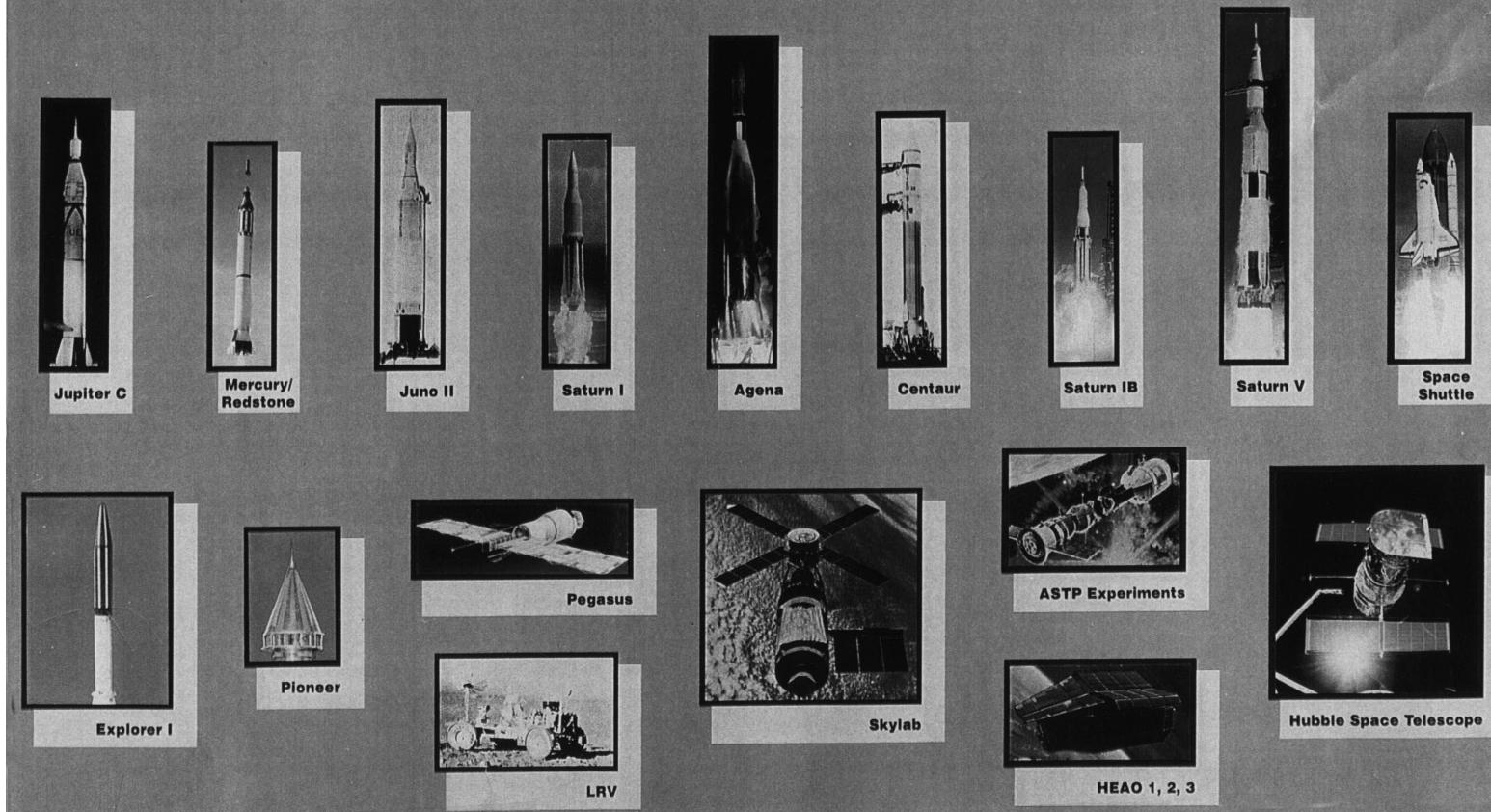
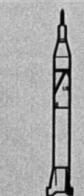
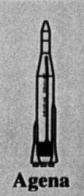
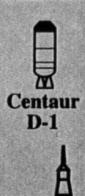
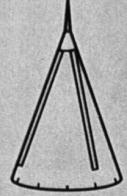
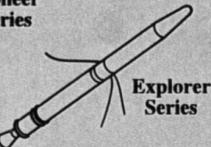
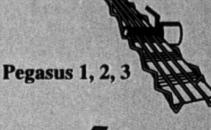
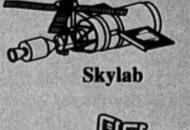
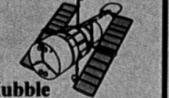
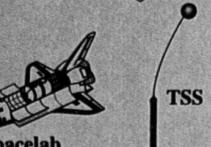
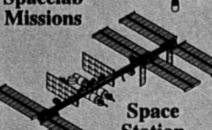
LESSONS IN PROPULSION SYSTEM DESIGN**Marshall Space Flight Center
Heritage**

Figure 10. Heritage.

LESSONS IN PROPULSION SYSTEM DESIGN

MSFC Program Evolution					
	1950's	1960's	1970's	1980's	1990's and Projected
Booster and Space Transportation Vehicles	 Jupiter C  Mercury  Juno I Engines: - A Series - S3D - RL 10 - H-1 - F-1 - J-2	 Agena  Saturn IB  Saturn V	 Space Shuttle: - Solid Rocket Boosters - External Tank - Main Engines  IUS	 Space Shuttle Operations  IUS Ops  RSRM  PAM A, F  TOS	Chemical Engines: - HLLV - Lunar - Mars  Automated Rendezvous & Capture  Upper Stages  Reusable Launch Vehicle
Payloads	 Pioneer Series  Explorer Series	 Pegasus 1, 2, 3  Skylab  LRV	 ASTP Experiments  Skylab Ops  Spacelab Development  HEAO's 1, 2, 3  Hubble Space Telescope	 AXAF  TSS  Spacelab Missions  Space Station	 Space Station Ops  Space Station Payloads  AXAF Ops  Geo. Earth Observatory  Spacelab Missions
Energy			 Solar Heating and Cooling  Mining Technology		

MSFC PROGRAM EVOLUTION

Figure 11. Project evolution.

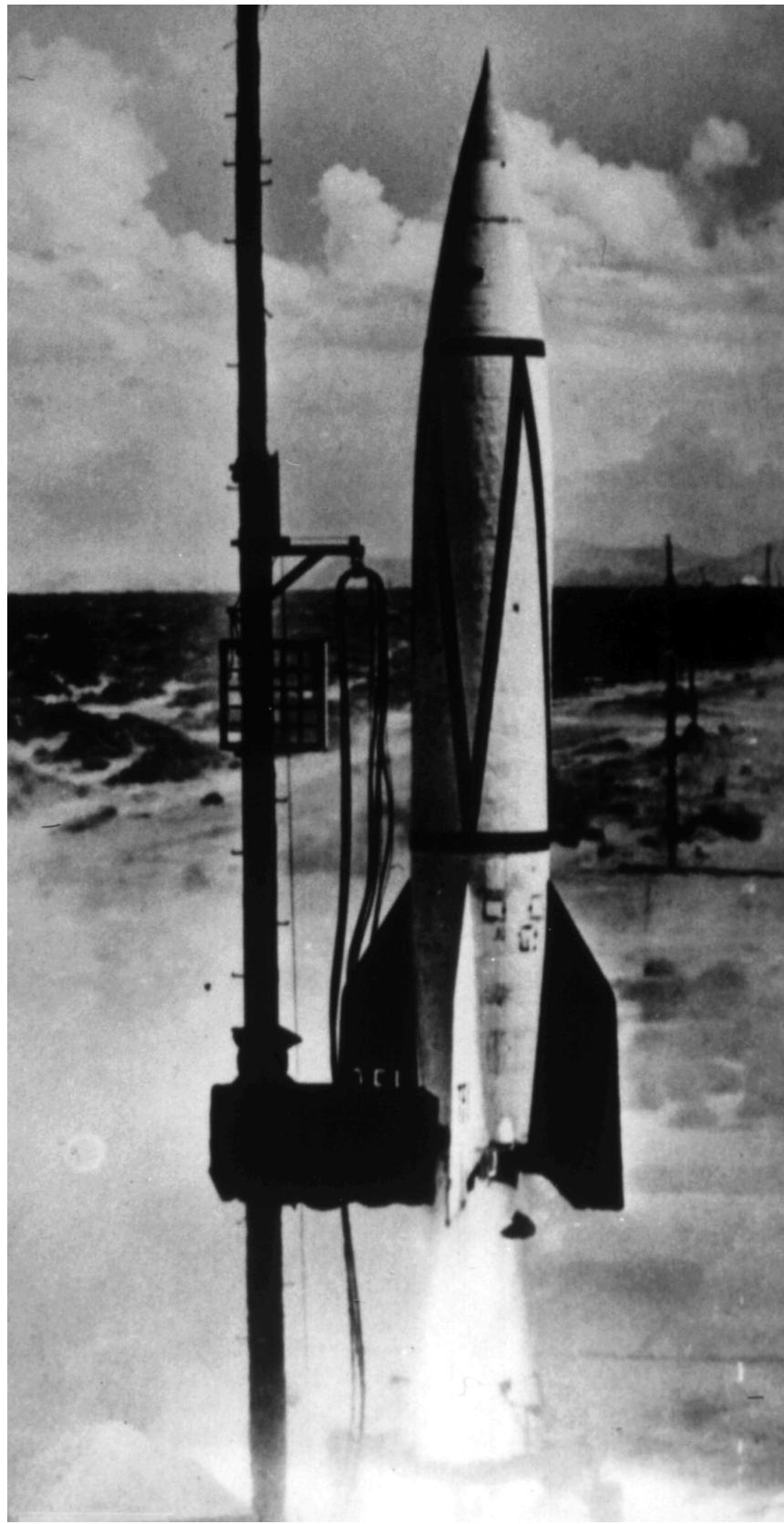


Figure 12a. V-2 configuration.

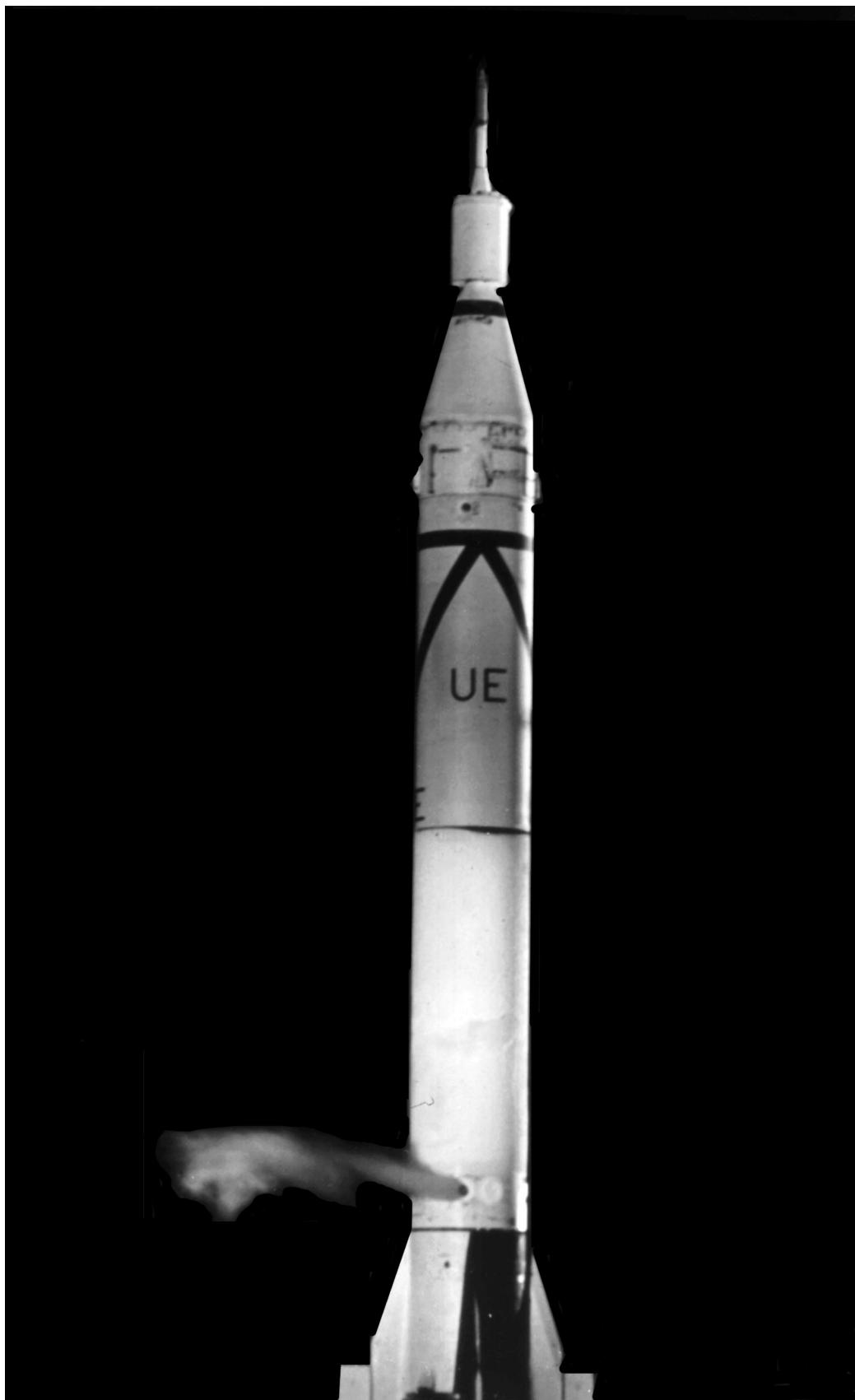


Figure 12b. Redstone configuration.

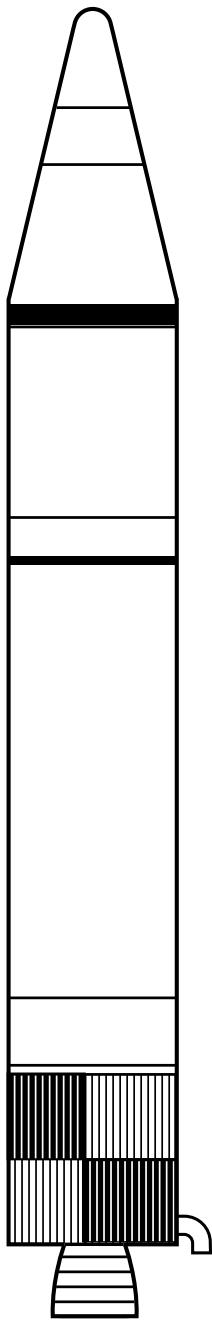


Figure 13. Jupiter configuration.

2. Evolution/Creativity/Innovation. During the Jupiter and Redstone programs, there was a struggle/trade of how best to deal with aerodynamically unstable missile systems, including performance loss due to winds, guidance, and loads. The aerodynamic stability question could be solved by putting large static aerodynamic fins on the aft end, and optimizing the external shapes and mass distributions (propellant tank locations). In many cases, the fins were too large, thus thrust vectoring was brought into play along with movable fins. In the case of the Redstone, movable vanes were put into the thrust stream. The Jupiter vehicle gimbaled the nozzle, driving the design of the gimbaling thrust bearing and actuation system. There were still the performance, guidance, control, and loads problems associate with the winds/aerodynamics-induced drift and loads.

Werner K. Dahm, whose experience spans the German V-2 through the present, provides his assessment of this area of development in the following sections.

Control Development of Rocket Vehicles:

a. During the time of the V-2 (World War II in Germany) and Redstone (figs. 12a and 12b) (after World War II in the United States), the control technology could neither cope with aerodynamically unstable vehicles, nor with aerodynamically unstable control surfaces. The vehicle had to be stabilized with fins of sufficient size to keep the center of pressure behind the center of gravity at all times, and the center of pressure of control surfaces had to be downstream of the hinge line. Primary control was exerted by jet vanes (aerodynamic controls require dynamic pressure to be effective, and are thus useless near lift-off and cut-off). Jet vane size had to be minimized to reduce the thrust loss from their drag. In addition, jet vanes are close to the single-engine-vehicle's longitudinal axis and provide only weak roll control. The effect of side winds on the fins creates "rolling-moments-due-to-combined-pitch-and-yaw" proportional to the dynamic pressure. The jet vanes could not overcome these moments. Therefore, aerodynamic control vanes were added at the fin tips of both missiles. These vanes were coupled with the jet vanes, and were driven by the same actuators. The Redstone had about the same range and payload weight as the V-2. However, the payload was a nuclear bomb, and the circular probable impact error was reduced to 450 m through terminal guidance of the separated entry body, i.e., the biconical nose of the missile. The circular probable error of the V-2 was about 10 times as large. The terminal guidance required:

- (1) Attitude control during the exoatmospheric flight of the separated warhead, to keep the control gyros from hitting their stops and losing their alignment. This control was accomplished with cold gas control jets located at the roots of the warhead air vanes. The jet controls were hard-coupled with the air vanes.
- (2) An aerodynamic design of the warhead and its air vanes for minimum center of pressure shift with Mach number, in order to minimize control system size and weight. This had to be accomplished on paper, and methods for this had to be developed along the way, since supersonic wind tunnels were not operational during the initial years of the Redstone design. The first warhead test data became available about 4 months before the first Redstone launch (August 20, 1953). The paper design was successful; no design changes were subsequently required.

The Redstone was really two missiles. The initial version, dubbed the "Experimental Redstone," was designed for a large, spherical payload; its design started in early 1951. In 1953, that payload was replaced by a more slender, cylindrical shape, which forced a redesign of the warhead and led to the "Tactical Redstone." That vehicle was eventually deployed in the field.

b. Hermes II: First Attempt to Fly an Unstable Launch Vehicle. The Hermes II project of the late 1940's was supposed to test fly a wing ramjet vehicle at Mach 2.7 and an altitude of about 30 km. The vehicle was a modified V-2. The tip up to the aft end of the instrument compartment was converted to a second stage, with a large wing ramjet, a pitch control canard, and two vertical control vanes at the rear end for lateral and roll control. In spite of the enlarged tail fins, the large ramjet wing rendered the launch configuration unstable. An accelerometer control was supposed to stabilize the vehicle. That control system worked well in bench tests, but failed in flight because the mechanical vibrations of the vehicle structure swarmed the accelerometer. The vehicle lost control somewhat into the flight. The launch took place at White Sands. Due to the severe limitations of the telemetry of that time, the flight vibration environment was largely unknown. The Hermes II project was canceled in early 1950.

c. Jupiter, IRBM, etc.: Swivel Engine Control of Unstable Vehicles. By about 1955, when the design of the Jupiter and Thor IRBM's and the Atlas and Titan Intercontinental Ballistic Missiles (ICBM's) began (all having the common Atlas developed engine), the propulsion community had managed to design swivel engines for thrust vector control. This powerful control authority, together with advances in control technology, permitted us to design and fly aerodynamically unstable missile configurations. Fins could by and large be abandoned. Jupiter became a simple blunt cone-cylinder body; the forward part of the cone was the reentry body, with the rear part being the instrument compartment. The control system included four cylindrical angle-of-attack meters sticking out from the surface of the instrument compartment, two each in the pitch and yaw planes. They compensated for the instability of the basic configuration. The system was successful and reliable. Jupiter roll control was accomplished by using the thrust of the engine's turbine exhaust, discharged through a swiveling pipe knee at the flank of the engine compartment.

The first breakthrough was the development of a drift minimum control logic that balanced the system between control, guidance, and performance in terms of a rigid-body system. This did not solve the induced loads problem. A rigid-body load relief scheme was developed that required schemes to sense the induced angle of attack. During these programs, two approaches were used. (1) Small vanes on the vehicle nose that turned to follow the aerodynamically induced flow were used, with transducers to measure the angular deflection. (2) Delta pressure sensors were placed on each axis of each side of the vehicle to provide a means of detecting angle of attack. In order to really work these issues, a good understanding of the atmospheric winds and density was required. The Air Force and the Army started programs to measure these atmospheric characteristics, requiring development of sensing systems, data evaluation, and modeling technologies.

Liquid propellant sloshing became a problem due to the loss of the second Jupiter vehicle. Analytical and experimental data and approaches were not available to characterize the problem. The first attempt was to place the full-size propellant tank on a railroad car then bump it against the rail end stop as an excitation mechanism. In terms of today's standards, this was not a well-controlled experiment. Large perforated cylinders with sealed spheres (commode floats) were floated in the tank and again tested, showing their good damping characteristics. The next Jupiter was flown successfully using this system. New technologies resulted as the program moved forward:

1. Analytical representation of fluid dynamic characteristics
2. Equivalent mechanical analog of the complex analytical equations (pendulum or mass spring slosh model)
3. Scale model testing techniques, instrumentation, and data evaluation
4. Development of slosh baffles that became an integral part of the structural stiffening ring, saving weight over the floating (beer) cans.

Because the warheads (payloads) had to reenter the Earth's atmosphere at very high speeds, aerodynamic heating and protection against it were major technology developments. During this time, the ablative protection attached to the structural metal was developed, verified, and used, and serves as a base technology for many future systems.

Plume heating of the vehicle base, both convection and radiation, was an additional technology issue. The first Jupiter was destroyed due to plume heating that destroyed the control wires, thus causing it to lose attitude control. Two technologies had to be developed and verified: (1) environment prediction, and (2) protection/control of environment.

Structural dynamic testing of the full Jupiter vehicle was attempted, but was not very successful due to improper boundary conditions (suspension systems); however, this served as a solid foundation for testing technology developed for Saturn.

During the Jupiter development, the interaction between the thrust vector control (TVC) servos and the second bending mode was a major effort. The potential problem was brought home in force on the first flight and subsequently fixed.

According to Harold Scofield, “Some people were never convinced that the second Jupiter lost was due to sloshing. We had only primitive models before the loss in flight. It is difficult to make definitive conclusions without some good analytical models. This problem (lack of definitive models) is still with us in some areas today, thus, exists the dependency on test data alone for those areas where modeling is inaccurate. The lessons keep repeating. We must pay attention to history or repeat it. Theories and analysis are always needed whereby to interpret the test and flight data.”

All of this was done with very little computational capability, which points out the innovation and practicality of the engineers in dealing with complex technologies. Those were the days of slide rules and Marchant and Frieden desk calculators.

As was mentioned previously, graphite jet vanes were used in the combustion gas stream (thrust stream) to control the thrust stream and thus the vehicle. However, severe erosion limited its effective control duration. This led to gimbaling the thrust chamber. Gimbaling the thrust chamber with the high-pressure ducts from the pumps input very large actuator loads and short duct life. This led to gimbaling the entire engine with the pumps attached to the engine. Balanced bellow ducts were incorporated for longer life, lower actuator power, and larger gimbal angles.

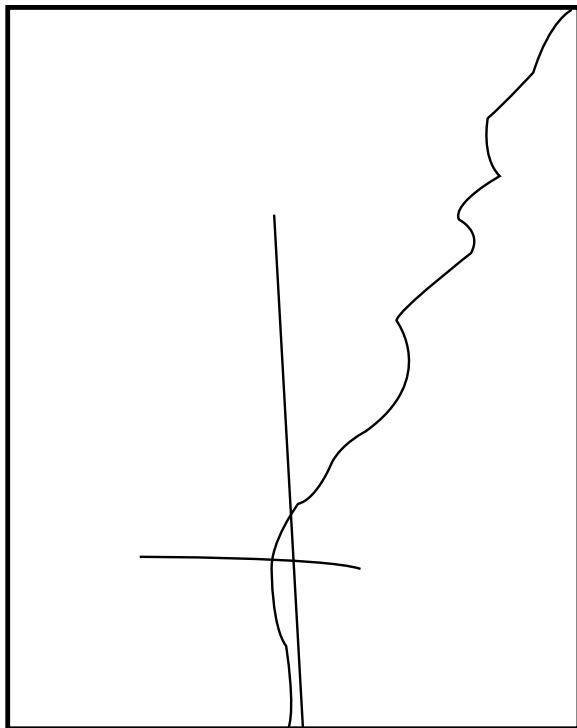
The Redstone thrust chamber was cooled through the double-shelled construction of $\frac{1}{8}$ - and $\frac{1}{4}$ -inch plates and contained several expansion joints. The shell plates were replaced by very long, thin tubes that significantly reduced the engine weight, eliminated thermal buckling problems, and allowed high-rate gimbaling at reduced actuator loads and power. Alcohol and hydrogen peroxide fuel systems gave way to a common kerosene lox propellant system for propulsion and pump turbine power.

It should be remembered that, initially, the Redstone was to be essentially a German V-2 built to American standards by the Army with the Germans brought to this country after World War II. The engine started out as an Air Force development program with a three-phase initiative: to understand liquid propulsion, derive the physics formulation in all associated disciplines, and apply the derived physics of a V-2 type engine using American standards and advanced physics. The final results were 50-percent less weight and 50-percent more thrust. This work was transferred to the Army team, which developed the Redstone. In the end, due to the innovations applied, it evolved to a true defense missile with a large warhead.

3. Problem Examples.

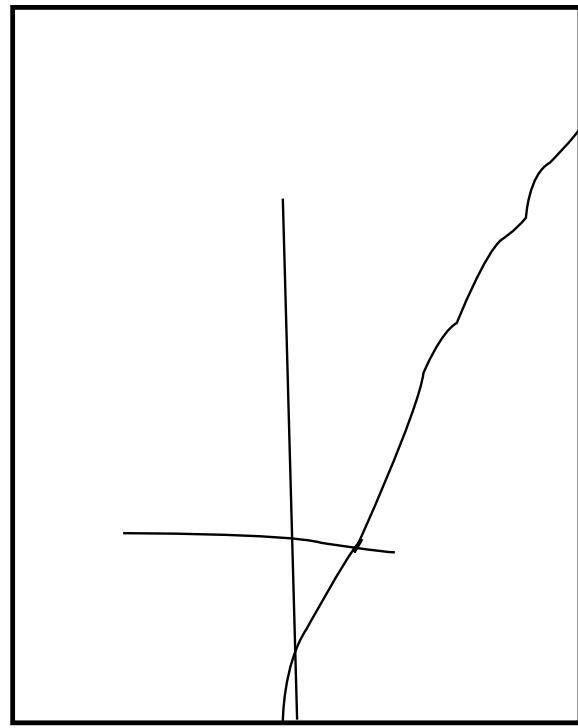
a. Redstone Potentiometer Feedback. The first dynamic problem experienced occurred early in the Redstone rocket program. A Redstone vehicle was in checkout and verification in a horizontal position in its transportation cradle. These early vehicles were manufactured at the Redstone Arsenal, checked out, then transported to Florida for launch. In this case, the control system was activated for checkout. The control sensors had a potentiometer pickup; due to some light shock, the wiper arm was moved from one wire to another, which resulted in a control signal. As a result, the jet vanes moved, exciting a structural mode. The structural mode in turn caused the wiper arm to move back, creating a new signal. The result was a closed-loop limit cycle instability between the sensor (pickup) jet vane (inertia), and structural mode, ringing out at the first mode frequency. The noise of this closed-loop resonance was very loud, vividly demonstrating closed-loop instabilities. The fix was simple in that a filter was incorporated in the loop that filtered out the frequencies associated with the modes and sensor pickup, breaking the loop and stabilizing the system. A later design also changed the pickup to a continuous magnetic type, adding margin to the problem solution.

b. Jimspheres (Atmosphere Sounding Balloons Erratic Response). Atmosphere environments are key to predicting space vehicle response during the ascent phase. Atmospheric winds are the key parameter to loads, flight mechanics, and control prediction. As a means of developing a statistical quantification of these winds, a balloon radar tracking system was developed by MSFC's atmospheric group under the leadership of Dr. William Vaughan. The goal was to not only measure large-scale or mean environments, but to get an accurate quantification of the wind gust down to 25-m wavelengths. The attempt to measure these small gust effects met with frustration. The smooth skin balloons were unstable (type of flutter or vortex shedding). In a controlled, no disturbance environment, a rising sphere would oscillate (fig. 14). A classical problem most would say. Dr. Jim Scoggins found the solution by observing the golf ball, then instead of small dimples, he added many conical spikes to the sphere's skin. The problem was solved. The sphere was stable (fig. 15). The resulting data base used throughout NASA is evidence. As a result, this modified sphere was named Jimsphere (after Jim Scoggin) as well as the data base (fig. 16). Many people have a small tie clasp with a miniature Jimsphere as a reminder of the agony one sometimes must go through to achieve the innovation required for solution to unexpected problems.



Time lapse trace of Jimsphere balloon released at 11:54 p.m., August 2, 1963 during stable atmospheric conditions and light winds.

Figure 14. Unstable Jimsphere time history.



Time lapse trace of rose balloon release at 11:25 p.m., August 2, 1963, during stable atmospheric conditions and light winds.

Figure 15. Stable Jimsphere time history.

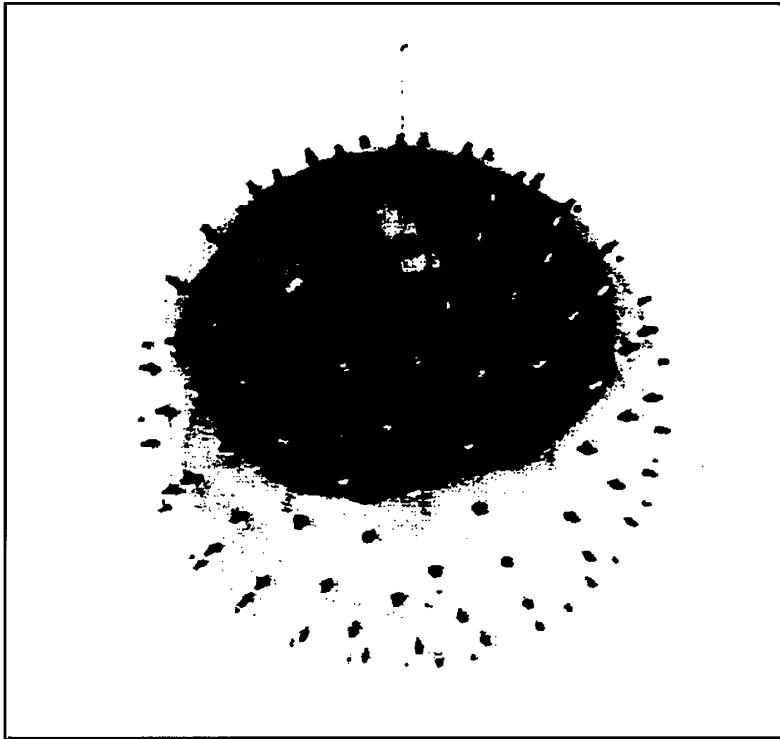


Figure 16. The Jimsphere balloon wind sensor.

c. Jupiter Sloshing Instabilities. A closed-loop control instability occurred on the early Jupiter firings. The Jupiter was a liquid propelled military vehicle. Sloshing propellant coupled through the control system and became unstable. This instability saturated the control system, and the vehicle went out of control during the maximum dynamic pressure regime of launch and broke up. The results were dynamic, with beautiful fireworks high in the sky, but very costly to the program. The instability was aggravated by the trajectory tilt program. The tilt program was a series of discrete steps instead of a continuous functional change that started the oscillation and reinforced the amplitudes of the wave through a forced oscillation. At this early phase in the rockets and space age, models did not exist for analyzing problems of this type. As a result, several things happened. Propellant sloshing data had to be obtained quickly. No analytical solutions were readily available. A test program was started that included both scale-model and full-scale testing. A slosh suppresser had to be found before the next launch. In order to meet this goal, a full-size propellant tank filled with water was placed on a railroad car. The railroad car was bumped against the track end stop as an excitation source. The first test was without suppression devices. Water was used to simulate propellant to establish frequencies, etc. Various devices were tried next as suppressers. The one chosen was called beer cans, which consisted of long perforated cylinders with flotation spheres at the top. The entire surface of the propellant was covered with these devices (fig. 17). The test showed more than adequate suppression was achieved, and the next launch was slosh free. In the meantime, other solutions were pursued, including development of analytical characterization of the sloshing propellant. This resulted in the development of slosh baffles (rings inside the tank that became part of the structural stiffness (fig. 18)), as the most effective analytical means of suppressing slosh, and parametric test data were acquired for oscillating propellants in both zero- and high-gravity (g) fields. Today, all space vehicles are analyzed and designed with this phenomenon in focus as a potential problem. The lack of analytical and experimental data prior to launching, as well as lack of experience in these type problems, led to the failure of the Jupiter missile due to propellant sloshing control system coupling. The fix was fairly easy and did not impact flight schedules drastically. This is not always the case, and repeats of this type instability should be avoided if possible. The innovative way special tests were conceived and conducted to meet launch schedules should be a lesson in this age of precise testing.

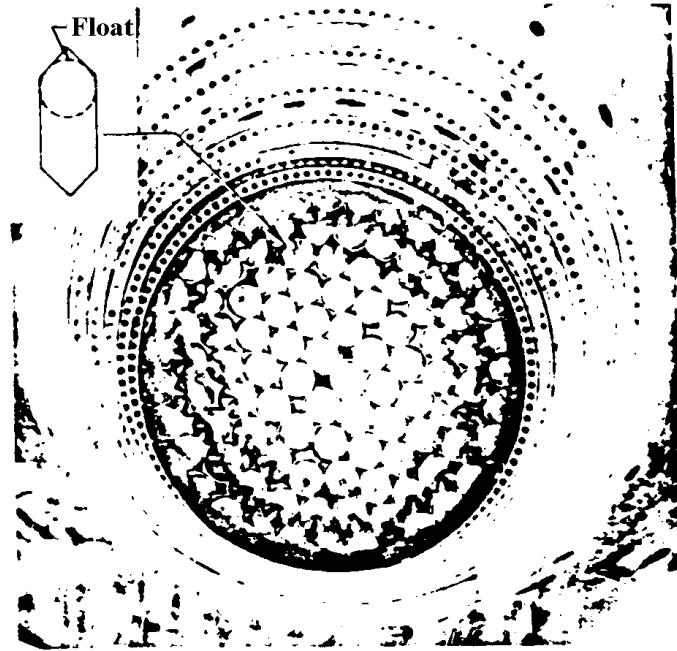
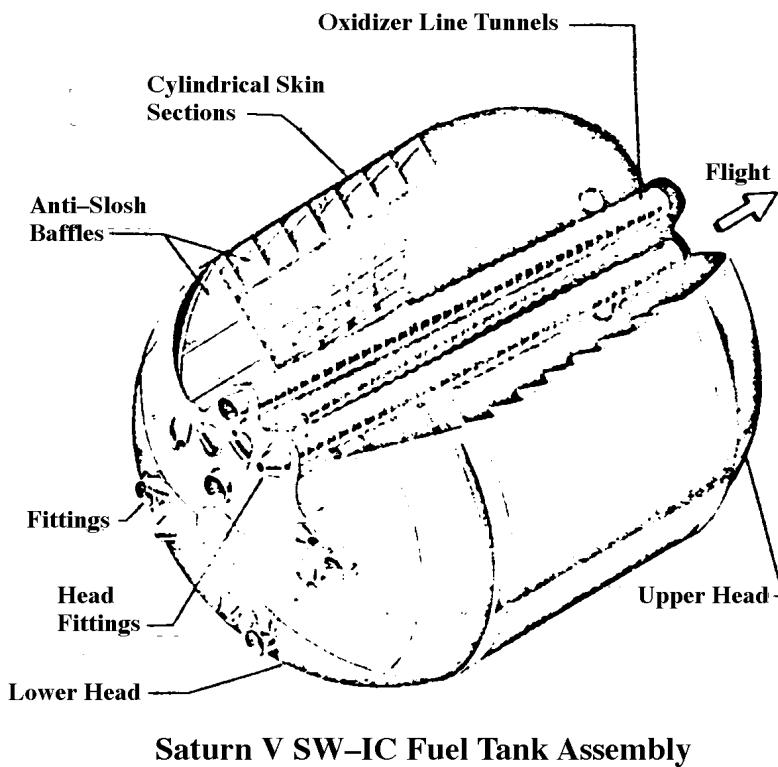


Figure 17. Beer cans, Jupiter anti-slosh devices.



Saturn V SW-IC Fuel Tank Assembly

Figure 18. Slosh baffles, Jupiter missile.

B. Saturn/Saturn Apollo/*Skylab* Launch Vehicles

The Saturn family was a unique experience in the evolution of space technology.²³ Being a family, it provided a building-block approach that led to the successful Moon landing and the *Skylab* space station program. It was all built on the foundation of the successful Redstone and Jupiter vehicles. The *Skylab* space program had two distinct parts: (1) the launch systems used to launch both the station the manned visits used modified Saturn V and Saturn IB vehicles, and (2) the operations of the *Skylab* space station and Apollo telescope mount (ATM). The *Skylab* launch vehicle is put in this section because of the commonality of the launch systems. The Saturn family is shown in figure 19a.

1. Characteristics. The Saturn I was envisioned as both a test-bed and a launch vehicle built using available technology and manufacturing tooling. The first stage was derived using a cluster of eight Redstone size and one center tank of Jupiter size propellant tanks and a cluster of eight of the Jupiter engine system. Figure 19b gives its characteristics. The lower ends of the tanks were attached rigidly to a large thrust frame. The engines were attached to the other side of this frame. The upper ends of the tanks were attached using a spider beam that also attached the second stage SIV. The tanks containing the cryo propellants had to have longitudinal slippage to account for thermal contraction, thus the load paths from the first stage engines were through the center tank and the four fixed fuel tanks. The sliding joint on the lox tanks were at the upper end as a connection to the spider allowed for the thermal expansion.

The second stage, SIV, was a liquid hydrogen/lox stage powered by a cluster of RL-10 engines. A very interesting story occurred relative to the first launch of the Saturn I vehicle. The SI was dynamically tested full scale to validate its complex dynamic characteristics. In order to simulate the flight conditions of free-free, the vehicle was suspended on bungey cords (elastic supports) where the vehicle suspended frequency was well separated from its first free-free frequency. In attempting to remove any residual effects of this suspension system on the free-free modes, an analytical technique (filtering of data) was attempted. Due to round-off errors in the data, instead of removing any unwanted effects, the process introduced three false modes in addition to the actual modes. Incorporating these modes into the control stability analysis indicated an unstable vehicle. The initial launch was held up for 2 weeks to sort out the problem. This problem got played in a *Fortune* magazine feature article. During the 2 weeks, it was possible to show that the modes were indeed false. In a big meeting with Von Braun, the decision was made to launch. The launch was totally successful. Testing approaches and data evaluation are a key element to the success of any program, as was so strongly illustrated here.

The Saturn IB was a derivative of the Saturn I with improvements to the structural and propulsion systems, including performance upgrade (fig. 20). The second stage SIVB had a new single liquid hydrogen/lox engine replacing the cluster of RL-10's in order to get more performance and higher reliability. The Saturn IB was used as the technology and operations demonstrator for the Saturn V Apollo. In fact, they both used the same SIVB stage and essentially the same guidance and control system. The Saturn IB became the launch vehicle for the manned launches to the *Skylab* space station.

There were two versions of the Saturn V: (1) the vehicle that launched all Apollo Moon missions and (2) the version that launched the *Skylab* space station. Saturn V Apollo was a three-stage propulsion vehicle plus the manned command and service module and the lunar excursion module (LEM). On later flights, it also contained the lunar rover stowed in the LEM. Figure 21 shows the overall configuration and its characteristics, while figures 22 through 24 are the configuration and characteristics of the three propulsion stages. In the case of the Saturn V *Skylab*, the SIVB stages were replaced with *Skylab* space station with only the first two stages being propulsive. This changed the external geometric configuration and thus its ascent loads. Figure 25 shows this configuration and its characteristics. The *Skylab* itself contained the living features with life support for the crew, a docking module, the ATM, and solar arrays on both the ATM and the workshop for power to operate all the systems. *Skylab* was a very successful program. After the four science missions were over, it

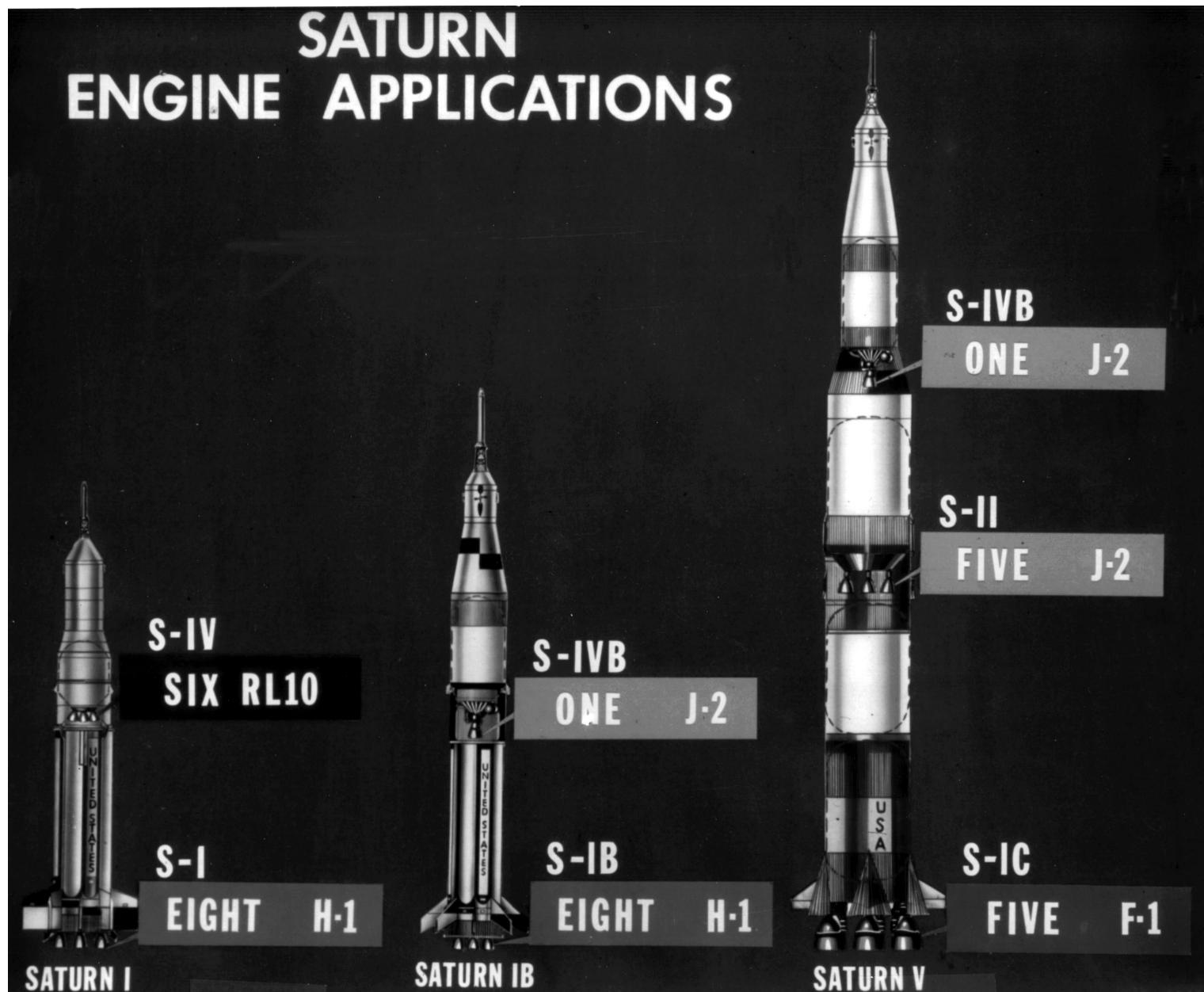


Figure 19a. Saturn family.

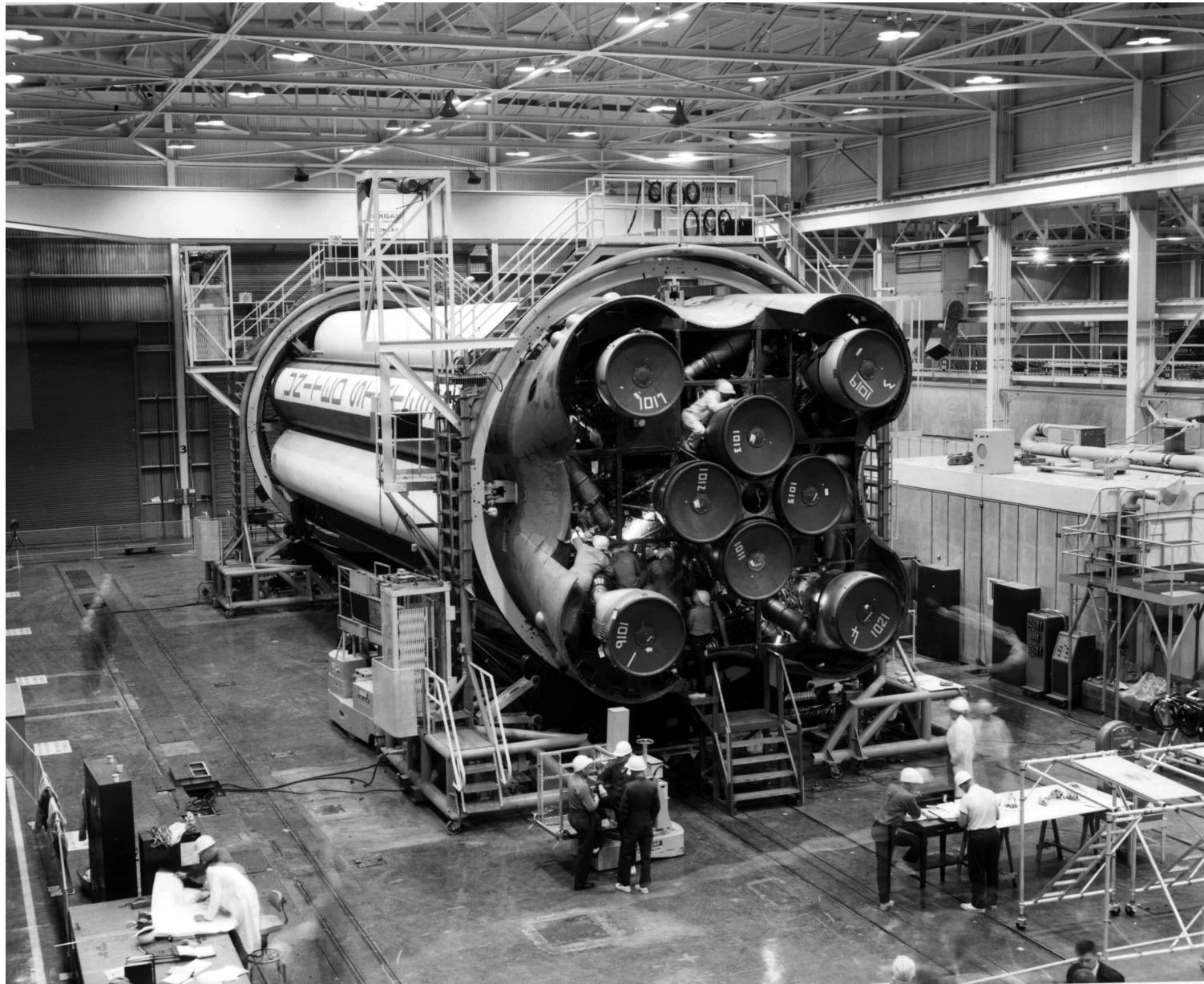


Figure 19b. Saturn I configuration.

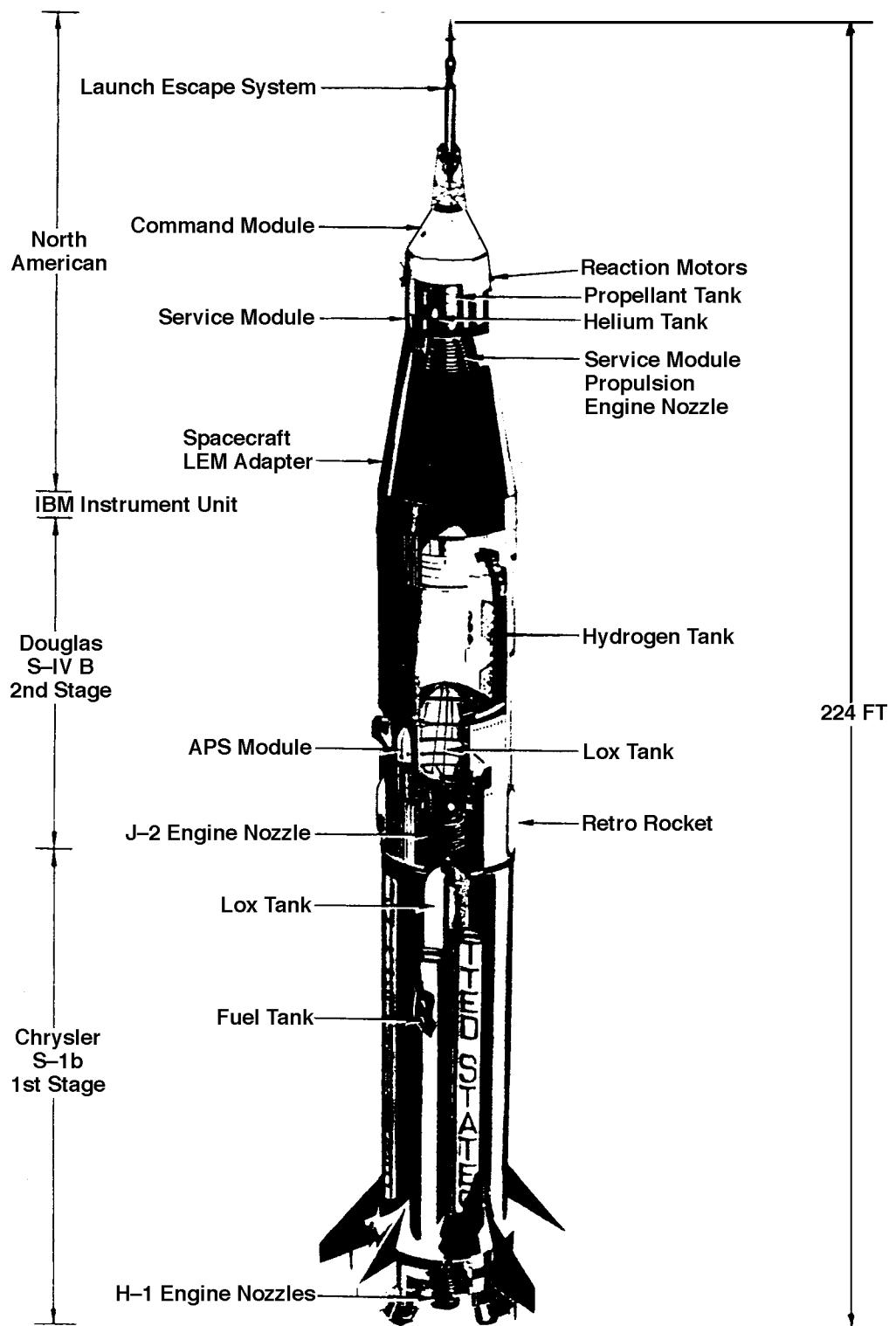


Figure 20. Saturn IB configuration.

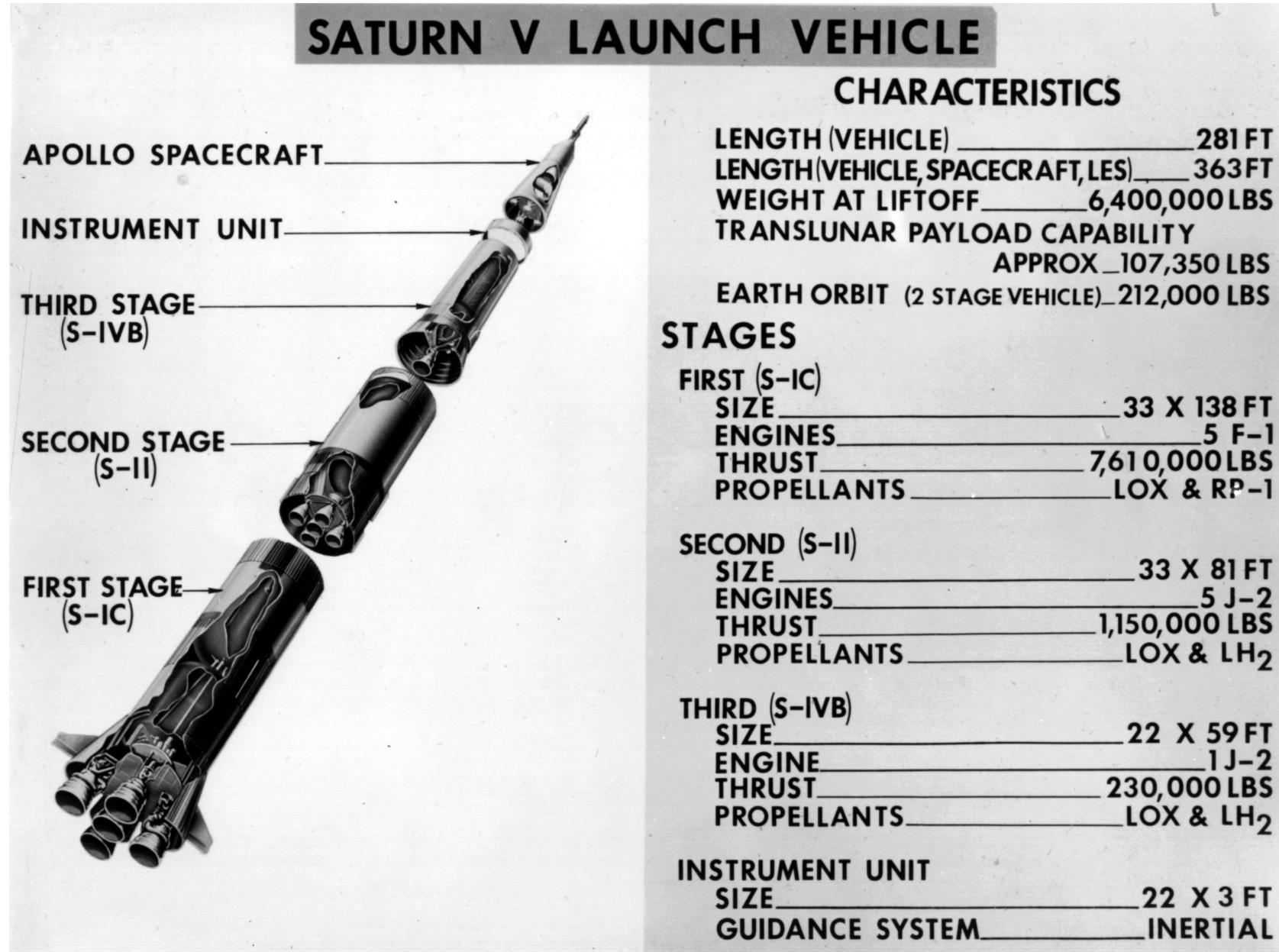


Figure 21. Saturn V configuration.

S-IC STAGE

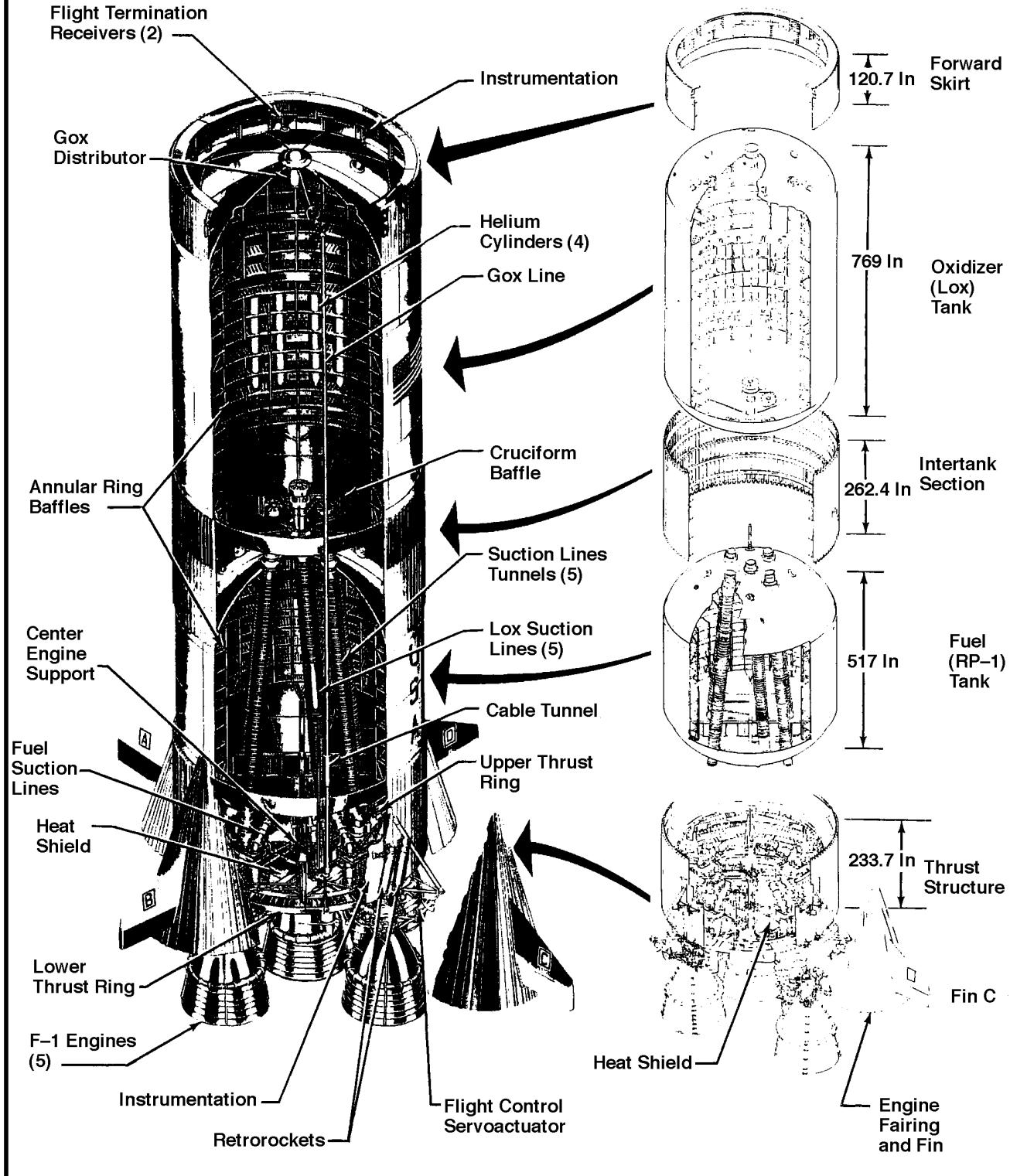


Figure 22. S1C Saturn V stage.

S-II STAGE STRUCTURE

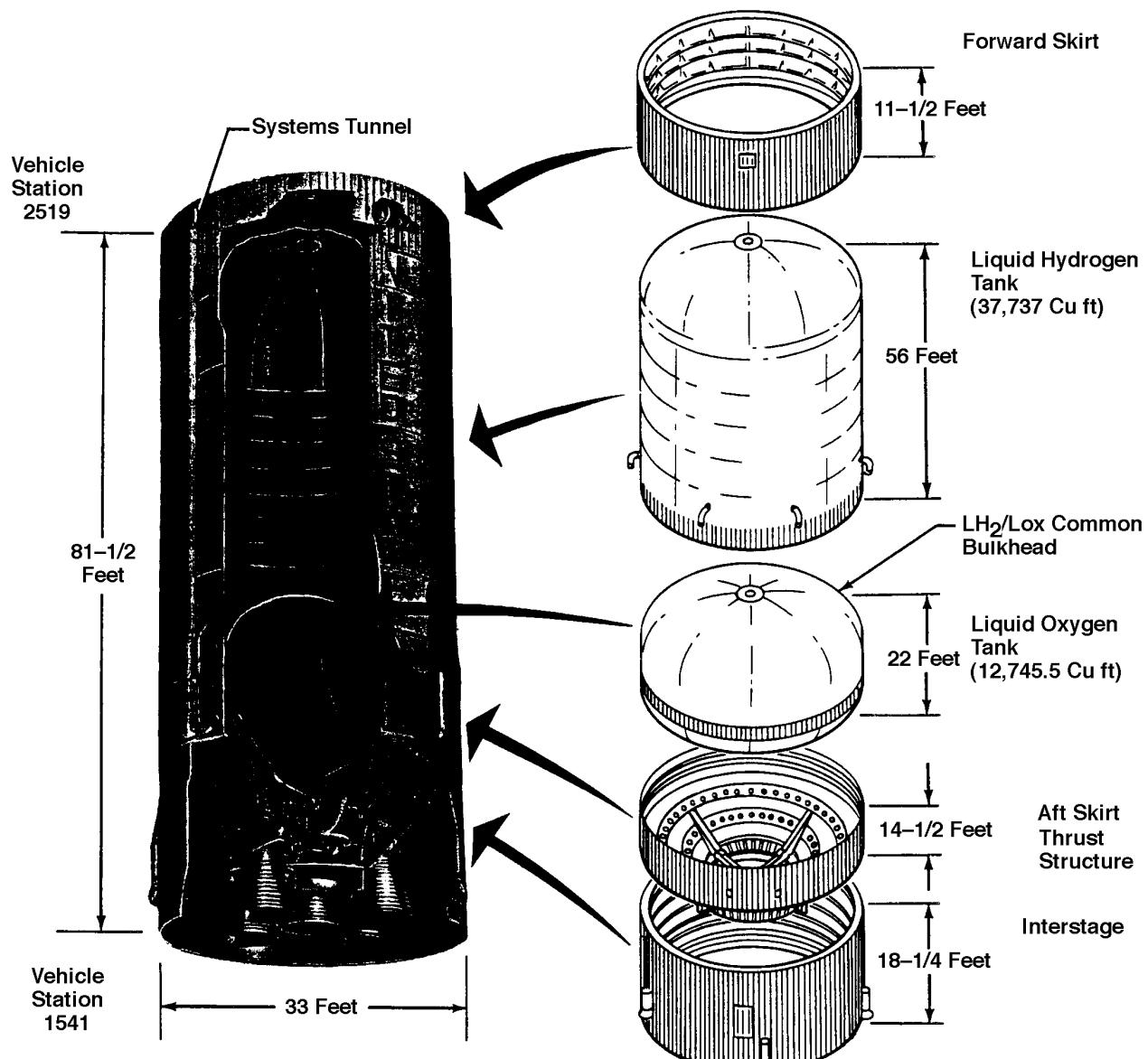


Figure 23. SII Saturn V stage.

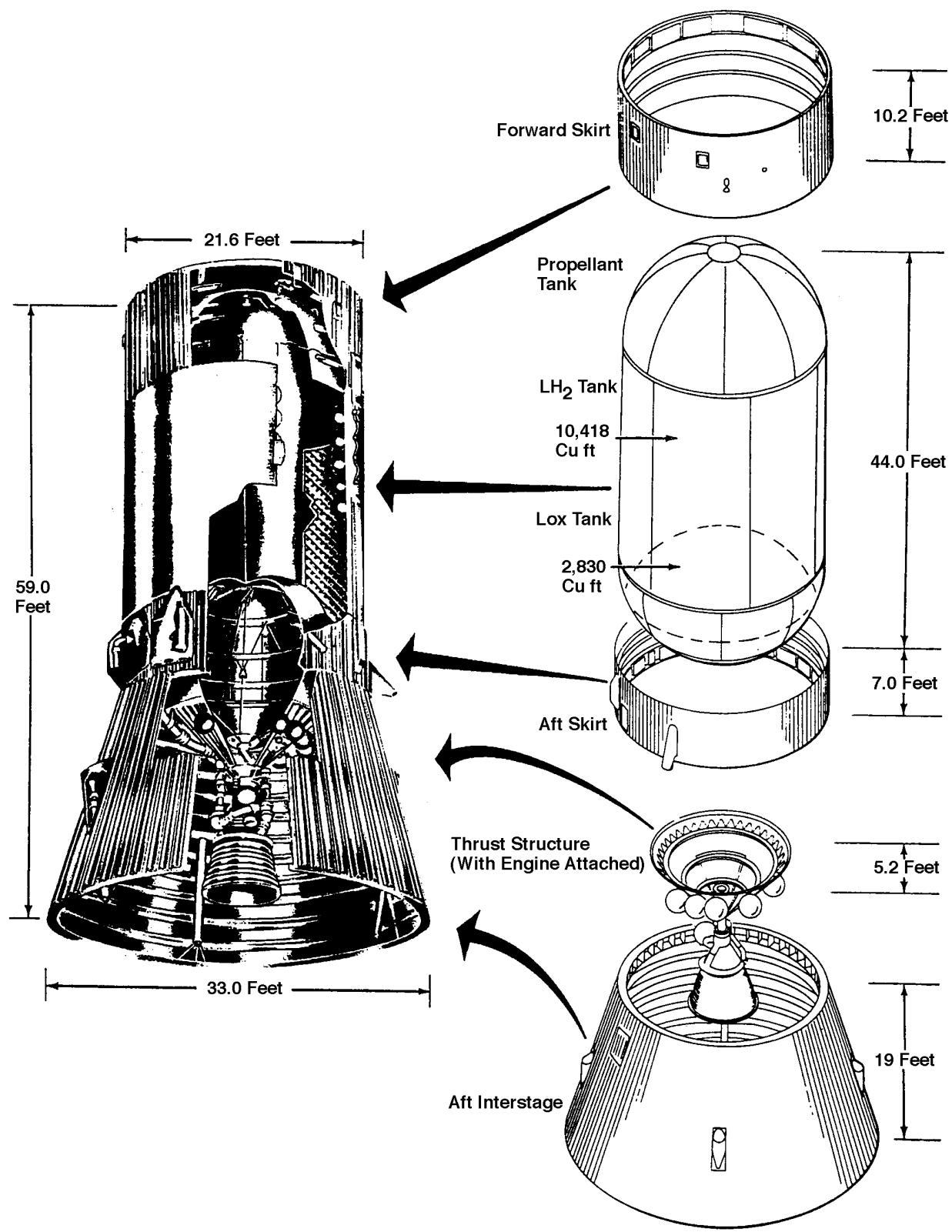


Figure 24. SIVB Saturn V stage.

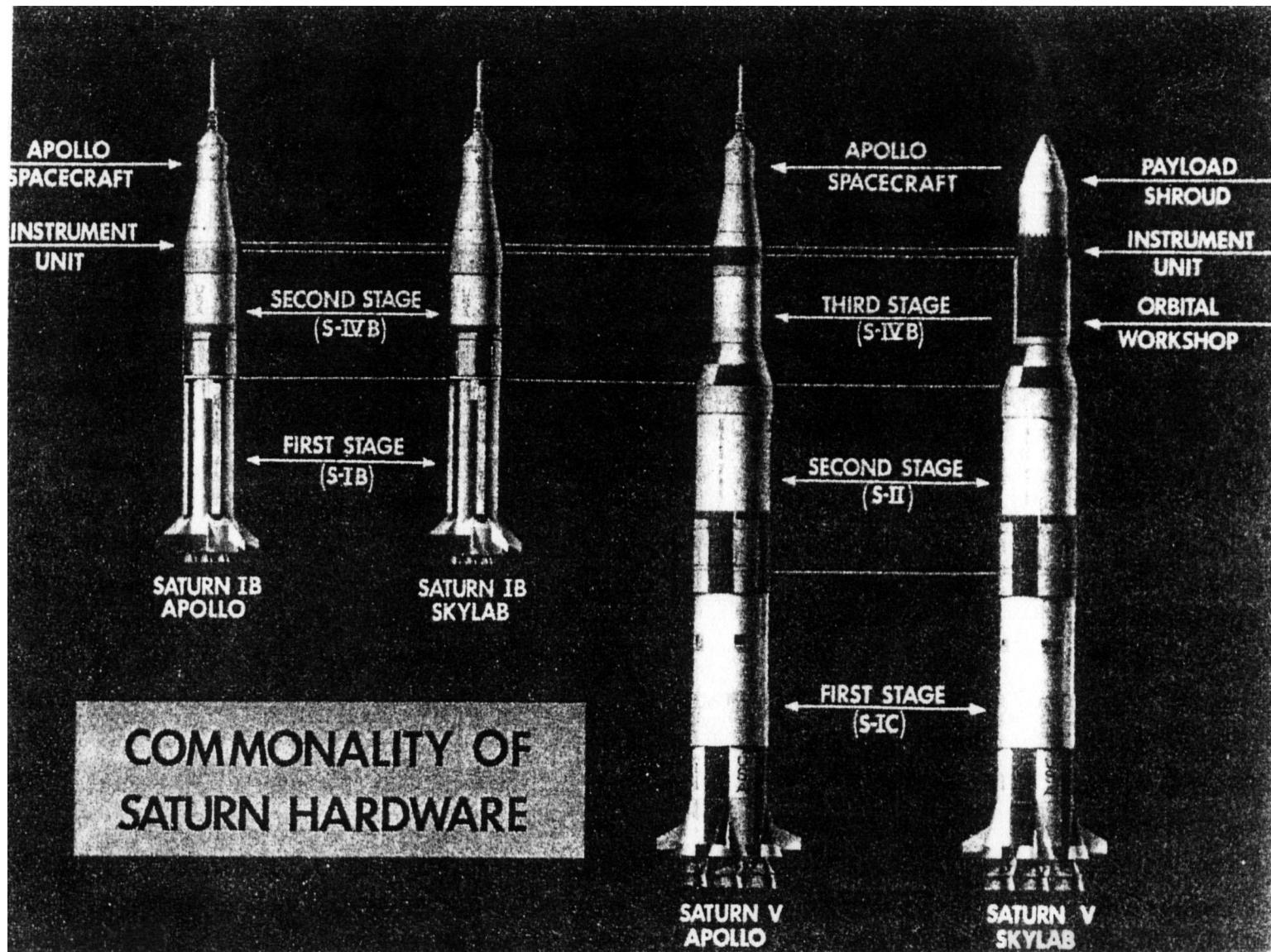


Figure 25. *Skylab* Saturn V configuration comparison.

stayed in orbit for several years. If the shuttle had made its original operations goal (launch), *Skylab* would have been reboosted and used as a forerunner to the planned larger space station. However, this did not happen and *Skylab* reentered the atmosphere, landing in Australia.

2. Evolution/Creativity/Innovation. During early development, many trades were done as the approach to use for going to the Moon, i.e., trans-lunar versus Earth-orbit rendezvous and docking. As history bears out, this heated debate ended with general acceptance of the trans-lunar injection. During this time, requirements indicated that four F-1's on the S-1C stage and four J-2's on the SII stage would meet performance requirements. However, it was decided to equip the S-1C with five F-1's and the SII with five J-2's. This was a very good stroke in that it compensated for weight growth, addition of margins, and the added missions using the Lunar Rover and the *Skylab* mission.

The Saturn and Saturn Apollo programs saw the maturing and harvesting of many technologies. This was due, in part, to the focus of landing a man on the Moon within the decade of the 1960's, the evolving of faster large computers with more mature codes, and the increasing education of engineers.

The control community developed and implemented the load relief technology using accelerometers instead of external flow-mounted angle-of-attack meters. Elastic body (modal) response and stability was a major issue and required the development of approaches to integrate structural dynamics and control. A natural extension was the development of modal suppression (in the aeronautics side, this was called ride control) techniques that, in conjunction with rigid-body load relief, not only reduced the basic aerodynamically induced loads, but also reduced the response due to elastic-body dynamics and wind gust and turbulence excitation. Years later, this technology evolved into active flutter suppression and aeroelastic tailoring.

The long, slender configurations raised many questions in terms of this control, structural, aero-dynamic systems due to potentially strong couplings (aeroelasticity). Technology evolved based on aircraft experience to deal with static aeroelastic effects, gust penetration, and the hammerhead effects. The technology involved not only analysis, but also scale-model wind tunnel testing.

While on the launch pad, vortex shedding was a real problem. This was solved through the use of a dynamic damper between the vehicle and the launch pad service tower, which had a disconnect for launch. This led to a wind constraint for launch in order to not have vortex shedding during the short time the damper was disconnected. One very innovative technique used to understand the vehicle structural dynamics characteristics while on the mobile launch platform (MLP) was the so-called tennis shoe test. The full-scale assembly of the mated test vehicle was assembled on the MLP in the vertical assembly building. Engineers wearing tennis shoes went to the platform near the Apollo capsule and excited the first bending mode by pushing with their tennis-shoe-clad feet. Modes were obtained with and without the modal suppression damper. This verification was accomplished with very little paper documentation and cost.

Wind biasing technology matured and became a viable operational tool for adding flexibility and increasing margins. All these technologies required developing and maturing technologies in several additional areas.

1. Measuring and quantifying the atmospheric characteristics in terms of wind speed, shears, gust, and turbulence
2. Statistical data evaluation techniques, tools, and models
3. Structural dynamic characterization of complex systems.

In the atmospheric characterization arena, the Jimosphere balloon and better radar tracking technology developed, allowing the description of the winds down to gust in the 25-m range. During this time, the synthetic wind profile matured to incorporate not only wind speed, but also shear and gust.

Three approaches were developed to handle the gust: (1) a 9-meter square wave that was tuned to match the vehicle lower mode frequencies, (2) tuned sine wave gust where amplitude varied with frequency, and (3) power spectral density formulation of turbulence.

A month-by-month statistically significant sample of what were called real wind profiles were developed through a comprehensive wind sounding program. These wind profiles were used to run Monte Carlo elastic vehicle control response analysis using a high-speed repetitive analog computer. This was a new analog computer that was developed for this analysis by a group of MIT professors. This system not only allowed a statistical characterization of the vehicle wind response, but also produced valid quantification of the synthetic profile approach for vehicle wind simulation. Additionally, it provided a technique to determine a probabilistic statement of the wind-induced vehicle loads. This probabilistic statement was key to the Saturn V *Skylab* launch statement.

One aside in terms of the synthetic wind profile development was the question of how to properly combine the wind speed, shear, and gust and how to do a conditional probability assessment. Putting together three-sigma (3σ) values of each was obviously too severe. Helmut Horn came up with the idea of conditionally dealing with a 3σ wind speed root-sum-squaring (RSS'ing) the 3σ shear and gust. William Vaughn was fundamental in this also, as well as in developing the wind sounding technology. Because the atmospheric (winds) had preferred monthly directional and speed characteristics, the technology evolved for biasing the launch trajectory to the monthly mean wind, reducing wind induced structural loads and increasing the launch probability/flexibility. Some performance loss occurred due to the path errors introduced in order to reduce angle of attack.

The synthetic wind profile approach did present one problem in that the vehicle response to winds had to consider variation of other vehicle parameters (control, aerodynamics, propulsion, structures) in addition to winds, and also the stress analysts and designers needed time-consistent data in order to perform adequate analysis. Judson Lovingood developed the A-factor approach, which took the RSS'ed responses from sensitivity analysis to produce scalings of the parameter variations that, when applied, produced a time response analysis with the same peak as the RSS'ed sensitivity analysis.

The development of structural dynamic characterization tools, their accuracy and validation, was fundamental to many of the design analyses and solution of problems discussed previously, as well as others such as pogo. Finite element structural analysis computational tools were developed late in the Saturn programs. SPAR was started, which was more of a process or procedure, as was NASTRAN, which started and matured during this time. Computer speed and capability precluded extensive use, thus the Saturn I (clustered tanks) was analyzed using 20 equations that were coupled through energy (generalized coordinates, Lagrangian approach) to get the overall system modes. This was a form of what is now called modal coupling. Full-scale dynamic, as well as scale-model dynamic test technology was initiated, and these analytical system modes were verified or correlated.

Suspension approaches/systems were key to dynamic testing of full-scale vehicles in the simulated free-free conditions. Criteria for these systems evolved along with configurations. Hydraulic or oil support bearings at the base of the total vehicle replaced bungy cords. Air bags were used on some configurations to isolate the suspension cables. Other innovative tests occurred such as the tennis shoe test discussed earlier was an example. Another was the use of 2-by-4's for damping the suspension system during dynamic test of the SI and SI-B by tying the suspension lines to the test stand with the timbers. It worked very effectively by changing the dynamic lateral mode of the line. It was demonstrated during this time that scale models had to have the manufacturing tolerances reduced by the scale factor or exaggerated deadband play would change the dynamic characteristics. Computer technology for both modal data acquisition, evaluation, and the excitation systems greatly simplified and started the process of automation. This computer technology used on Saturn model testing was a first. The hardest problem to model was the nonlinear fluid analysis of the J-2 which was needed for the SII and SIVB pogo analysis and their solutions. The data were finally obtained from flight results and some hot-fire ground tests.

It was learned and demonstrated during this time that in scale-model testing not only the materials had to be scaled, but also the manufacturing tolerances or else the nonlinearities of the gaps biased the dynamics. Modeling for pogo (S V) was even more complex due to the hydroelastic coupling. Tank bulkheads had to be modeled in detail. Propellant ducts had to have both their structures and acoustics determined as did the cavitation compliances of turbopumps. Verification required testing for many modes and instrumenting both the fluid and the structure. Single-point random structural dynamic testing technology was developed and applied to the lox tank/bulkhead hydroelastic testing. Defining and ensuring correct and known test boundary conditions were also major issues.

Technology in dynamic analysis evolved very fast during Saturn/Apollo development and required nonlinear simulations in order to characterize the cavitation effects, etc., associated with problems such as pogo that occurred on Saturn V. Analog and hybrid computers were the main tools used because it was easy to represent the nonlinearities if they were understood. Great physical understanding was required in order to implement the system on these computers as well as deciding what nonlinearities were essential. Capacity limited what could be described. This meant most of the describing equations could be linear and implemented on the digital portion of the hybrid. The nonlinear elements were implemented on the analog side.

Multidisciplinary analysis was becoming a major technological development, as the above discussions imply. The coupling between the propellant utilization system (optimizing propellant usage and lower residuals) and the guidance system of the SIVB stage was a major step in simulation. It was more than just propulsion and guidance. We called it (Harold Scofield's quote) "the world's longest closed-loop system: slosh/propulsion utilization/engine/ guidance/control." These various problems were the beginning of the major multidisciplinary technology development that would continue to develop throughout aerospace history.

The development, qualification, and certification of the various components were major technological developments. Analytical techniques were not too adaptable to developing the vibration criteria arising from mechanical and acoustical sources. This led to the development of data banks from both the environment and response side based on component masses and mounting configurations. Scaling technology was developed for taking new environments and new components and scaling new responses from this in order to develop the vibration criteria. The testing side developed force and control technologies and hardware for qualification of these components to these vibration criteria on shake tables.

The Saturn V vehicle was the first launch vehicle that took the developing linear elastic fracture mechanics technology into the initial design phase. The phenomenon of structural failure by catastrophic crack propagation below yield stresses had been known for many years. The Saturn V propulsive system tanks were the first space structure where critical flaw lengths were calculated and a proof test pressure was implemented on all production tanks to ensure that any undetected cracks or crack-like defects would not cause failure during the cycle life of the mission.

The author was not involved personally in other technological developments; however, many were just as innovative as these discussed. The development of the right materials, particularly the welding technology for the large propellant tanks that not only contained the liquid but also were the load carrying structure for thrust loads, is a good example.

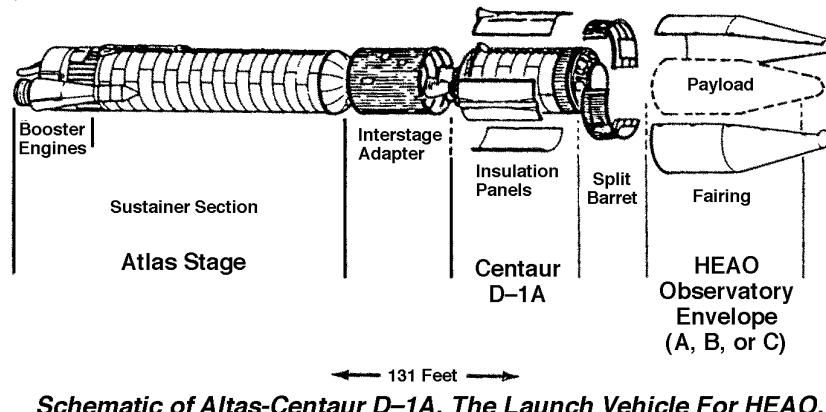
C. HEAO

1. Characteristics. The sophistication of the tools of high energy astronomy has increased dramatically over the past 2 decades, culminating in the HEAO program, which put three satellites into orbit, each carrying over 3,000 lb of experiments. This program has transformed high energy astronomy from an interesting tributary or side branch to one of the main channels of astronomical research. The

HEAO experiments have made it abundantly clear that if we want to understand an astronomical object, we must look at its radiation at all energies, not just in the narrow optical band.²⁴

HEAO was actually three different satellites designed to explore far space from orbiting systems using x-ray telescopes. The three different missions carried different instruments and program objectives. The first was a scanning mission to survey and map x-ray sources throughout the celestial sphere over a wide range of x-ray energies. The other two missions had more specific objectives. HEAO A was launched August 12, 1977, and ceased operations January 9, 1979. HEAO B was launched November 13, 1978, and ceased operations April 26, 1981. HEAO C was launched September 20, 1979, and ceased operations May 30, 1981.

The launch vehicle was the Atlas-Centaur rocket built by General Dynamics Convair Aerospace Division, with the engines provided by Rocketdyne and Pratt & Whitney. The total height of the HEAO Atlas-Centaur vehicle ready for launch was 39.9 m (131 ft), with a total launch weight of 165 tons, of which of the spacecraft plus experiments first of three and half tons, including one and a half tons for the experiments (fig. 26).



Schematic of Atlas-Centaur D-1A, The Launch Vehicle For HEAO.

Figure 26. HEAO Atlas-Centaur.

The spacecraft that carried the HEAO experiments was built by TRW Systems. Their role was to design and develop the HEAO spacecraft, to integrate the mission, to support launch operations and to perform flight operations of the in-orbit observations. The basic HEAO A is sketched on figure 27. A sketch of HEAO B is shown on figure 28, while HEAO C is shown in figure 29. The program was very successful, adding greatly to science and our understanding of the universe.

2. Evolution/Innovations/Creativity. There were many innovations in the HEAO program; however, most were in the instruments and science, which the author was not involved in at the time. Reference 24 covers these extremely well, as it also does for the evolution the program through its ups and downs, including one program cancellation that forced a total reassessment and redesign of the program from four large spacecrafts to the three smaller ones actually flown. This is a story within itself and deserves contemplation before embarking on new programs of this scope.

3. Problems. One problem associated with HEAO will be discussed briefly, the launch vehicle had a pogo problem. Pogo occurred to some degree on each Atlas flight near the end of the main stage burn. It was never viewed as a problem for most launches since the payloads could easily handle the loads. The concern for HEAO occurred because on the last two or three launches the amplitude increased. The concern was that this increase signaled a potential blossoming of pogo to the extent that it would be damaging to HEAO spacecraft. Much analysis of flight data and pogo studies were made in order to understand this potential problem. It was finally decided that it was an acceptable risk to launch HEAO without any pogo fixes. This was done with no problem occurrences and without further pogo amplitude increases.

HEAO Mission A Experiments Configuration

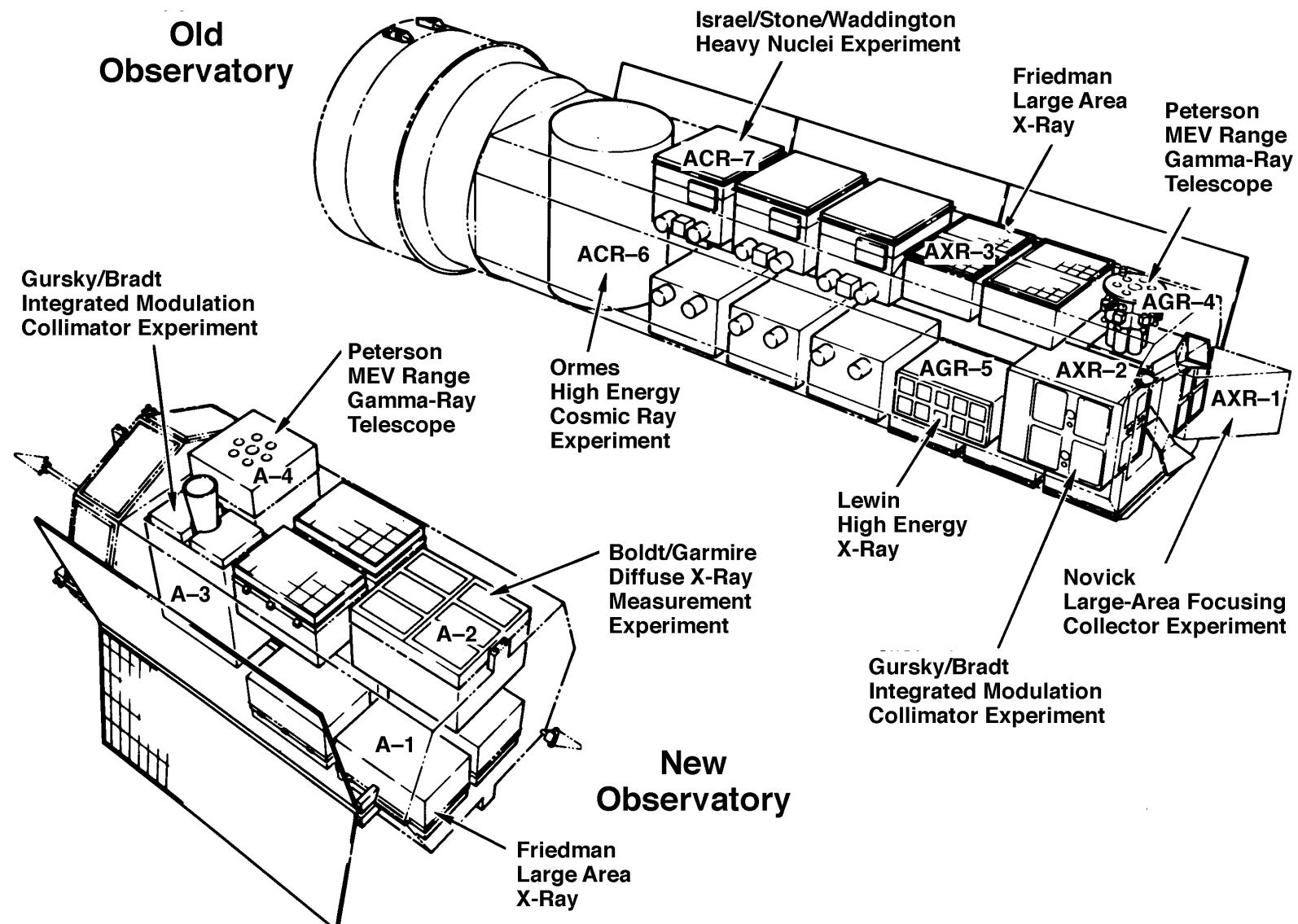


Figure 27. HEAO-A configuration.



Figure 28. HEAO-B configuration.

Figure 29. HEAO-C configuration.

D. IUS/PAM-S/TOS

1. Characteristics. The IUS was an upper stage vehicle under Air Force management that NASA bought for several of its missions, including the launching of the tracking data relay satellites (TDRS). It consisted of a solid propulsion stage and an inertial unit for guidance and control. For planetary missions, it was modified by adding an additional stage named payload assist module (PAM-S), built by McDonnell Douglas. The IUS was designed and built by Boeing. Figure 30 shows the basic IUS configuration as used by NASA for its different missions. It weighed 32,000 lb and was designed to fit into the shuttle cargo bay with the payload attached to the IUS nose. The configuration with the PAM-S attached with its payload is shown on figure 31. One additional part of the design had to do with the support cradle that had to raise the IUS perpendicular to the shuttle payload bay, then eject out of the bay. The first stage was not ignited until the orbiter had moved a safe distance from the IUS system. This cradle, with the IUS attached in the orbiter bay, had to have a special design to move the resulting scissors mode away from the shuttle lift-off dynamic modes, or pay a huge weight penalty. For safety reasons the expulsion system was energized by springs instead of pyros. Figure 32 shows the cradle design.

2. Creativity/Innovations. Several innovative design features were incorporated in the IUS design (fig. 33). It used an extruding nozzle extension in order to conserve space in the transportation mode. Pyro initiated time instead of electronics was used for timing of separation, etc., as was also done on the PAM. The design of the cradle and separation system was unique and of a safe weight, and it was safe for manned flight. The simplicity made for a robust system.

3. Problems. The main problem that occurred with IUS was early in the program when it was being launched from Air Force missiles for their payloads. The extruding nozzle extension failed, causing a loss of mission. Teams of Air Force, NASA, and industry were formed to evaluate the problem and come up with a fix. This was accomplished and no further problems have occurred other than the normal changing out of parts as they aged on the shelf between uses.

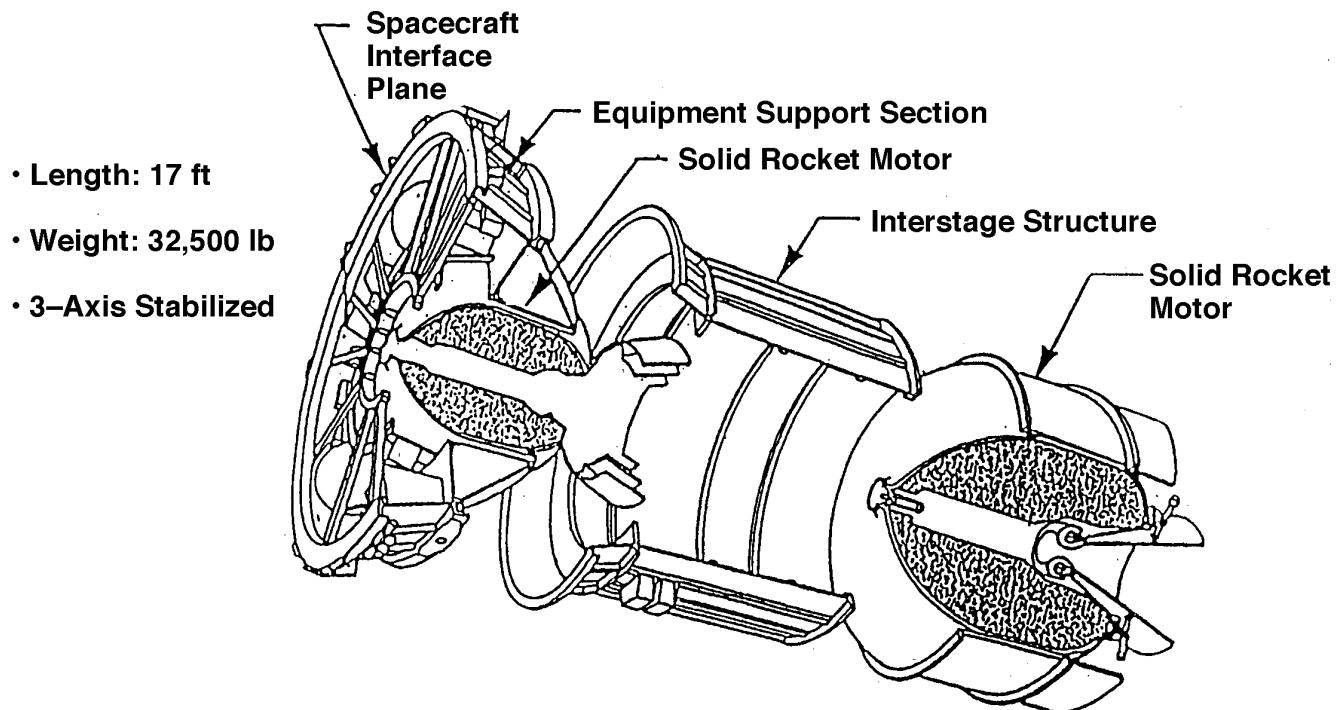


Figure 30. Basic IUS configuration.

ULS/PAM-S/IUS Configuration

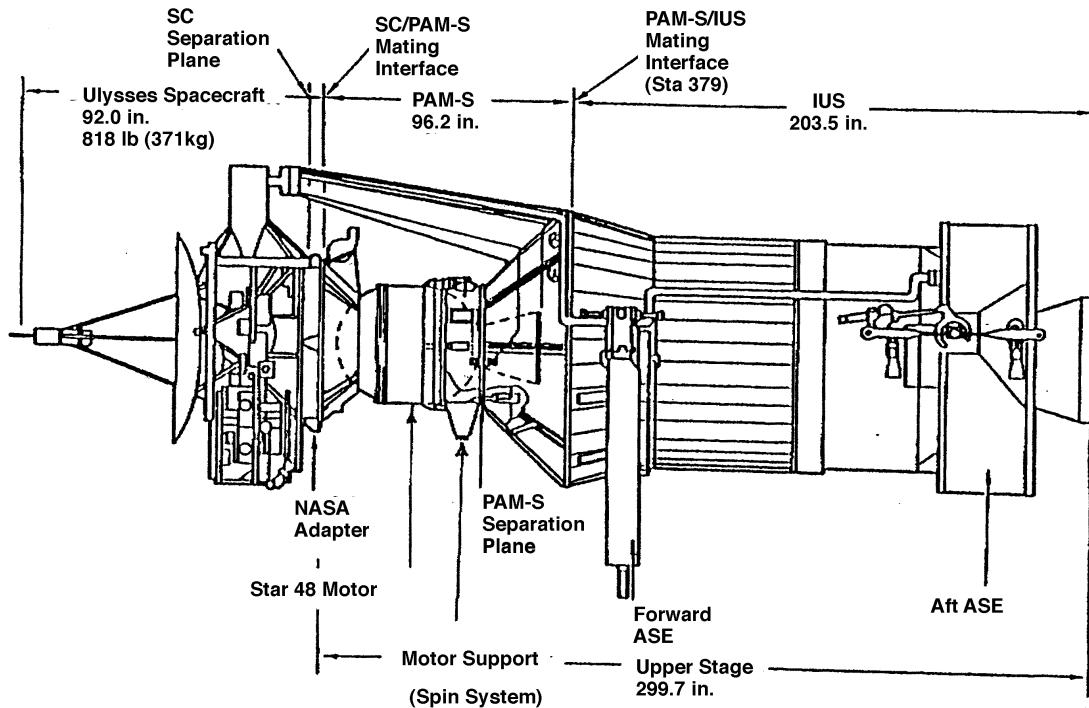


Figure 31. IUS PAM-S configuration and IUS-PAM-S Ulysses configuration.

IUS-17/PAM-S/Ulysses Sequence of Events

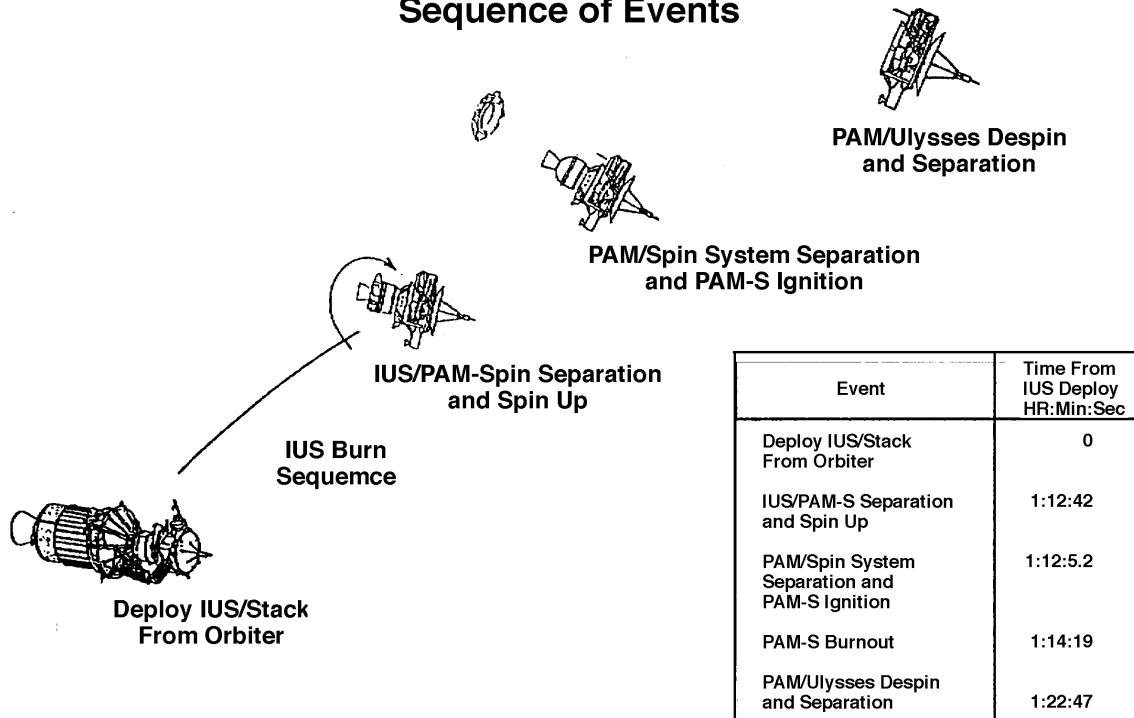


Figure 32. IUS PAM/Ulysses sequence of events.

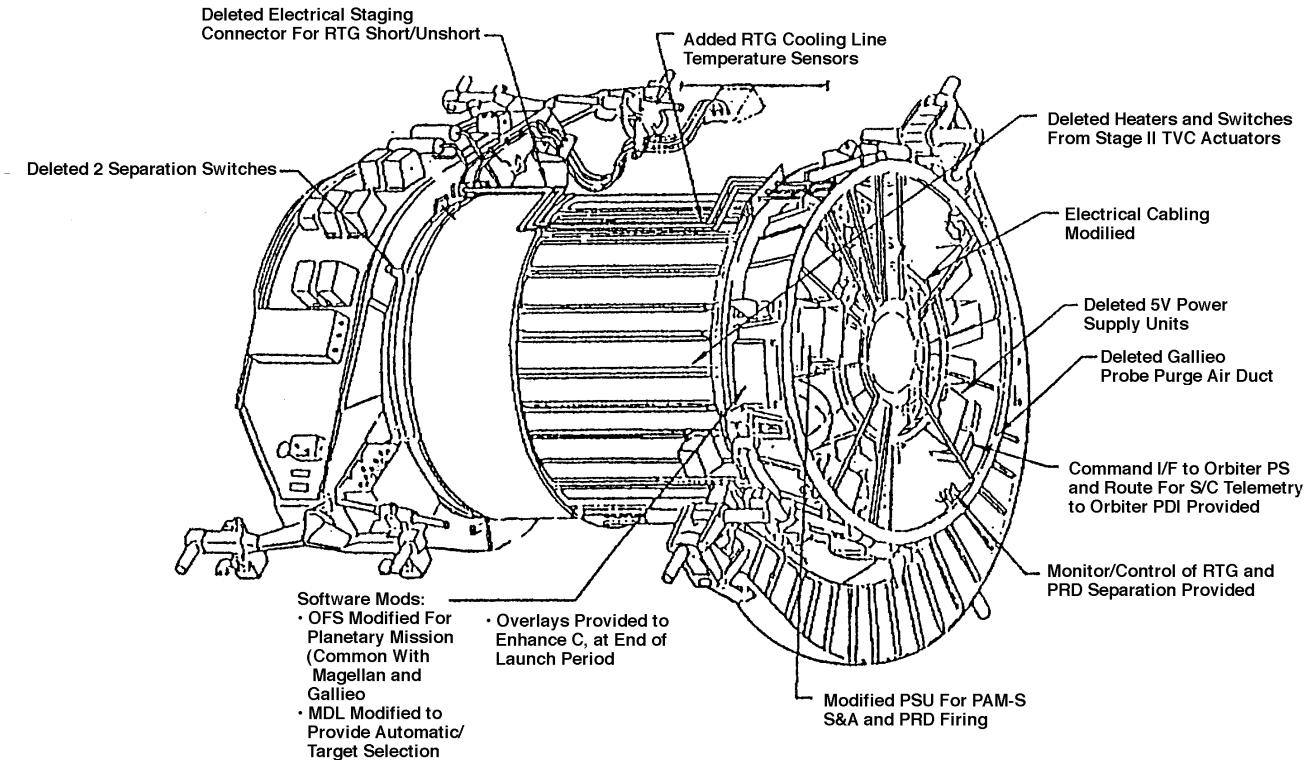


Figure 33. ASE configuration.

E. *Skylab*, America's First Space Station

1. Characteristics. “The vast accomplishment of *Skylab*—in solar and stellar astronomy, in detailed study of our planet from the incomparable vantage of orbit, in using the exciting new laboratory tool of weightlessness, and in proving man can work productively in space for extended periods—are almost too profound to grasp. The legacy of *Skylab* to be passed on to planners and operators of future manned space programs is best stated in two words: can do!” according to Rocco A. Petrone.²⁵

The *Skylab* space station evolved into a configuration composed of (a) docking adapter; (b) solar observatory; (c) airlock module; (d) workshop solar wings (solar energy for electric power) (was launched with two solar wings, but operated with one since the other was lost during launch); and (e) workshop and living quarters (fig. 34). Cutaways of the living and working quarters are shown on figure 35 and include all the life support system shown on figure 36. Table 1 lists the basic subsystems, the manufacturers, module name, function, length, diameter, and habitable working volume. The heart of the science mission was the solar observatory (fig. 37), which had its own fine pointing control system with ATM instruments, solar power system (four solar arrays), and EVA access. A typical mission profile is shown on figure 37.



Figure 34. *Skylab* configuration.

SKYLAB ORBITAL WORKSHOP

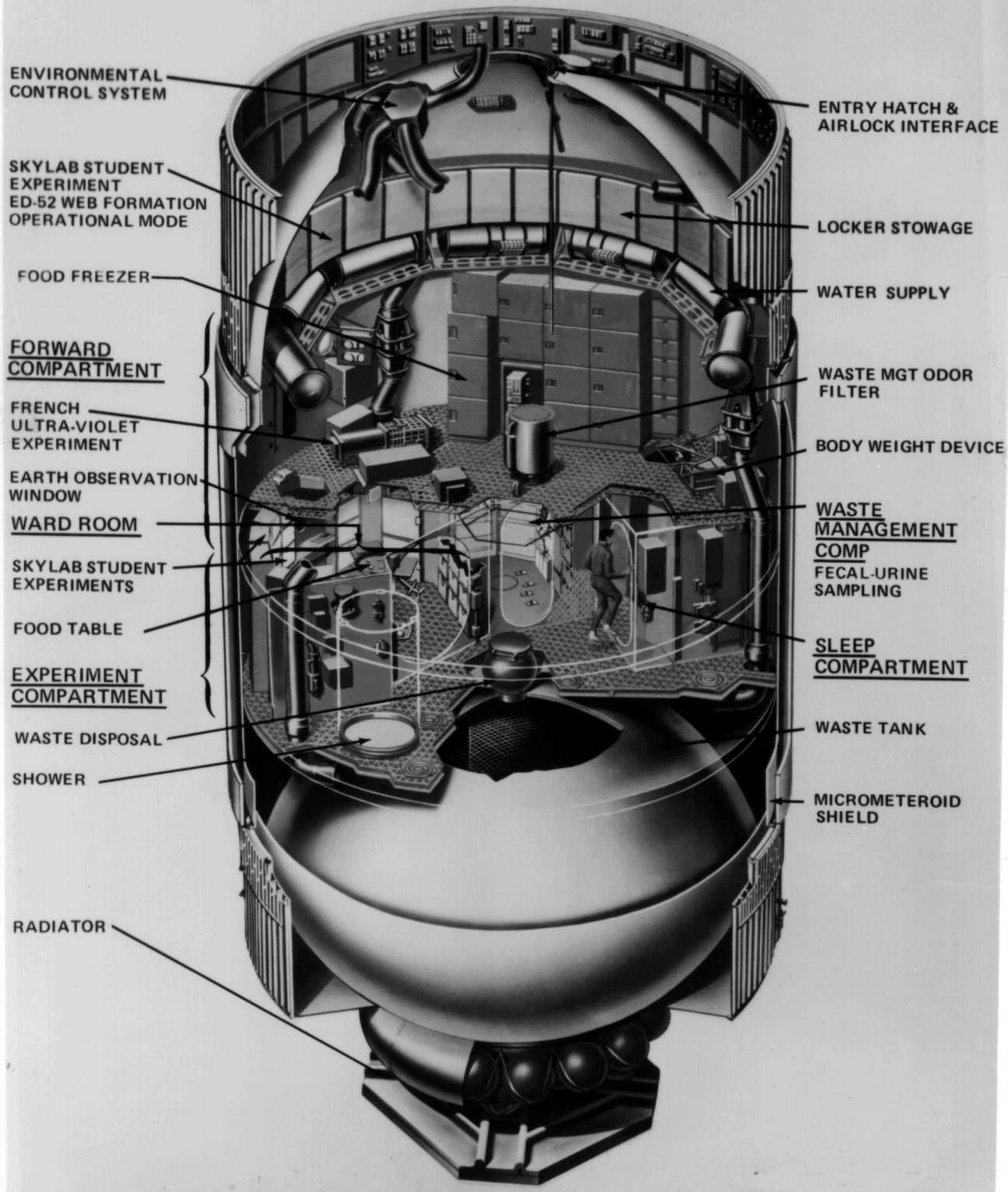


Figure 35. Cutaway of *Skylab* living quarters.

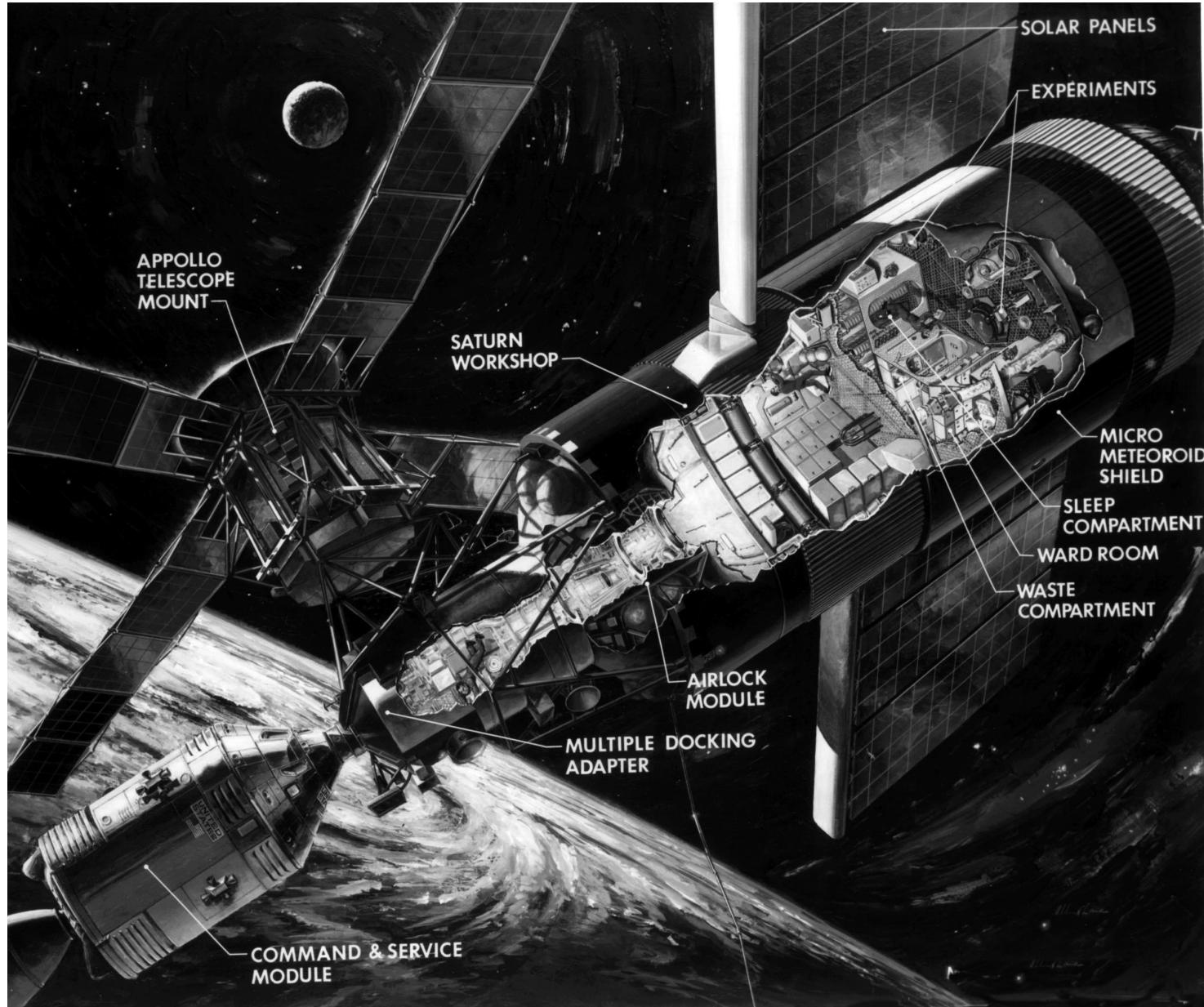
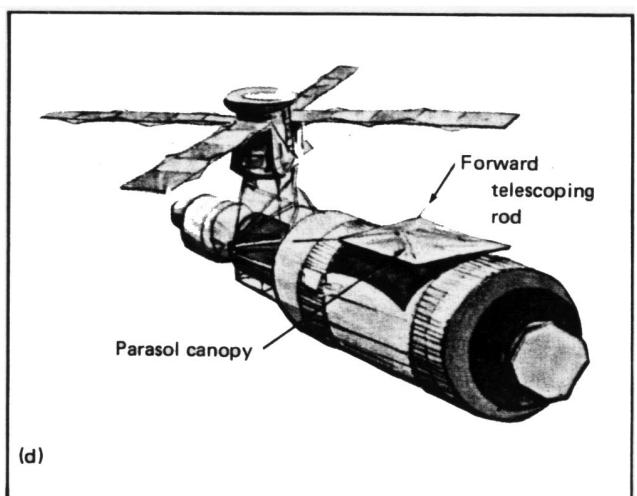
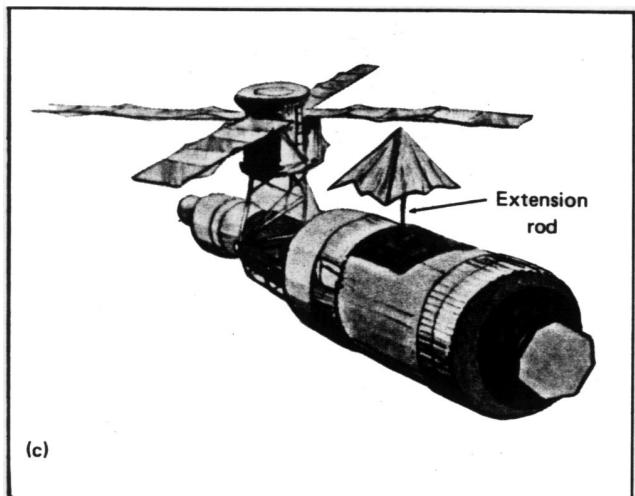
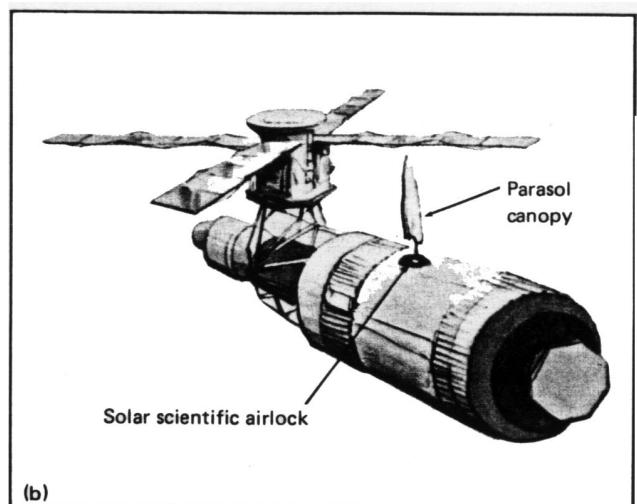
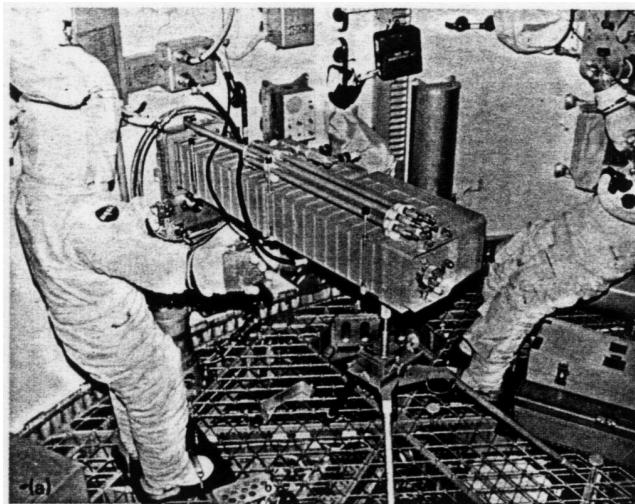


Figure 36. *Skylab* life support system.



Sequence of parasol deployment. (a) Parasol operation from interior of airlock. (b) Parasol at partial extension. (c)

Full extension and partial deployment. (d) Fully deployed and retracted for service.

Figure 37. *Skylab* parasol canopy.

The space station was launched using the Saturn V first two stages, with the workshop replacing the SIV B third stage and the command and service module (CSM). The CSM was launched separately using the Saturn I B launch system. Three visits (missions) were carried out with three different crews over the span of approximately 6 months, with a total manned orbit days in space of 1,171 days, dictating the need to provide the crews with comfortable living quarters and a healthy and safe living and working environment. This meant that not only did the system have to depend on the internal active thermal control system, but it also had to be shaded from the direct Sun rays.

A table of the missions of *Skylab* is shown in table 2, providing duration, EVA's, observations, etc., and showing the outstanding success achieved in the presence of major problems. The ability of the human innovation and creativity to solve major problems with work around demonstrated the "can do" attitude effects.

2. Evolution, Innovation, Creativity. It is not possible in a report of this type to even hint at the scope of the evolution, innovation, and creativity accomplished in order to ensure the successful

Table 1. *Skylab* configuration.

	A	B	C	D	E
Module	CSM	Docking adapter	Solar observatory	Airlock	Workshop
Manufacturer	Rockwell International	Martin Marietta	MSFC	MDAC-Eastern	MDAC-Western
Module name (development)	CSM	Multiple docking adapter	ATM	Airlock module/fixed airlock shroud	Orbital workshop
Function	Crew ascent and descent	Docking interface Controls and displays Earth observation Stowage	Solar observation Power source Attitude control	Power control and distribution Environmental control Data center Extravehicular activity hatch Caution and warning	Primary living and working area Laboratory Power source Stowage
Length	10.45 m (34.3 ft)	5.27 m (17.3 ft)	4.05 m (13.3 ft)	5.36 m (17.6 ft)	14.66 m (48.1 ft)
Diameter	3.96 m (13.0 ft)	3.04 m (10.0 ft)	3.35 m (11.0 ft)	3.04/1.67/6.70 m (10/5.5/22 ft)	6.7 m (22.0 ft)
Habitable working volume	5.95 m ³ (210 ft ³)	32.28 m ³ (1,140 ft ³)		17.66 m ³ (624 ft ³)	295.23 m ³ (10,426 ft ³)

Table 2. *Skylab* missions.

Manned Periods	First	Second	Third	Total
Launch	5/25/73 9 a.m. EDT	7/28/73 7:10 a.m. EDT	11/16/73 9:01 a.m. EDT	
Splashdown	6/22/73 9:49 a.m. EDT	9/25/73 6:19 p.m. EDT	2/8/74 11:17 a.m. EDT	
Duration (day:h:min)	28:0:49	50:01:9	84:01:16	171:13:14
Revolutions	404	858	1214	2,476
Distance (million miles)	11.5	24.5	34.5	70.5
SEVA ^a	0:37 (5/25/73)			
EVA 1 ^b duration (h, min)	3:30 (6/7/73)	6:29 (8/6/73)	6:33 (11/22/73)	
EVA 2 duration (h, min)	1:44 (6/19/73)	4:30 (8/24/73)	7:01 (12/25/73)	
EVA 3 duration (h, min)		2:45 (9/22/73)	3:28 (12/29/73)	
EVA 4 duration (h, min)			5:19 (2/3/74)	
Total EVA's	5:51	13:44	22:21	41:56
Solar observatory photos	30,242	76,600	75,000	182,842
Earth resources photos	8,886	14,400	17,000	40,286

- a. Standup (in spacecraft hatch) EVA.
- b. EVA (completely outside of spacecraft).

missions of *Skylab*. This is particularly true in light of the major damage done to the station during the Saturn V launch, which led to major on-orbit repairs, etc. *Skylab* was an observatory, a zero-g laboratory, and living quarters for three men, with all that implies in terms of power, life support, science, observation requirements, etc. If one has an interest in the *Skylab* space station, first read NASA SP-4000 edited, by Leland Belew. Other information is also present in references 26, 27, and 28, and other NASA historical publications.

Skylab started out as a “wet workshop,” which was an active SVB stage outfitted on the ground with laboratories, etc. After the SIV B burn, the stage would be purged and made ready as a workshop. The observatory, ATM, was launched separately and was first to be a free flyer or docked with the spent SIV B stage, the workshop. As the program evolved, these concepts were dropped, with the basic SIV B stage being remanufactured to being a dry workshop. In other words, it was built and launched as a combined ATM and workshop with the docking adapter. The first two stages of the Saturn V could effectively launch this combined system into orbit by replacing the CSM with a protective aerodynamic shroud that shielded the ATM and docking module during ascent and was then ejected in orbit, exposing the ATM and MDA. For this approach to be effective, the ATM and its solar arrays and the workshop solar wings had to be stowed during launch and then deployed on orbit. This allowed efficient use of launch vehicle space and maintained essentially the same aerodynamic configuration for the Saturn V launch vehicle. Using the dry workshop concept allowed the development of adequate living quarters, laboratories, supplies, etc., to be stowed and launched together, saving much on-orbit assembly.

Innovations and creativity were clearly evident in all aspects of the design. The meteoroid shield was also a thermal shield. Mixtures of gold and white paint further enhanced the ability to receive heat (gold) or reject heat (white). Combined with this system were refrigerator cold plates, radiant heaters, radiators, etc., which got rid of heat generated by heat-producing equipment as well as the Sun’s rays and, thus, maintained a balanced temperature. Exercise equipment for the crew was provided, as was carbon dioxide removal, waste management, etc.

In order to stay on orbit for 9 months, power had to be generated using the Sun’s energy. The station had to be controlled as to its position in space, as well as maneuvered, and the ATM had to point accurately at the solar system. Control moment gyros were used for pointing and stability, while the reaction jet control system maneuvered and maintained basic position. The ATM had a fine pointing vernier system relative to the basic *Skylab* orientation. As stated above, read NASA SP-4000 to get a full impact of this expression of creativity.

Probably the greatest innovation and creativity occurred due to the problems created during the *Skylab* launch on the Saturn V. The next section on problems will also discuss these great recovery efforts.

3. Problems

a. Meteoroid Debris/Solar Wing Failure. During the launch phase at around 60-s flight time, it was observed that problems had occurred with the micrometeoroid shield as evidenced by the rising temperature; however, the launch continued flawlessly and the *Skylab* was injected into orbit. In orbit, when the workshop solar wings were being deployed, it was clear that a mooring problem had occurred. Data indicated that one wing was gone and the other only partially deployed. Also, the temperature of the workshop was rising to dangerous levels, threatening medical supplies, etc. The workshop equipment was not designed for these high temperatures. The crew flight scheduled for the next day was delayed 10 days in order to understand the problem and design fixes. There was also a concern that the high temperature would cause the internal insulation to give off life-threatening gases. With much ingenuity, ground control was able to orient the workshop to balance the temperature around 130 °F. Since the workshop had very little solar power, most of the power to maintain the station was obtained from the ATM’s four healthy solar arrays. The system was designed to pass power from one system to the other. The cause of the failure of the meteoroid debris shield and the solar wing was an improper

venting design of the box that contained the wing, which caused it to trap the 14.7 lb/in² ground atmospheric pressure that in near vacuum ripped the wing loose and destroyed the debris shield, exposing the gold covering of the workshop.

NASA Centers worked around the clock to understand the problem and to design fixes. The missing solar thermal/debris shield had to be replaced. Many options were looked at, with two being designed, built, and verified. Their installation was practiced extensively in Marshall's Neutral Buoyancy Simulator tank, designed to simulate zero gravity working conditions for astronauts. One lesson is clear: "one must have in place the technologies needed when problems occur." The two solar thermal systems were: (1) a fold-up parasol type shield that could be deployed off the workshop airlock viewing port by the crew (this was accomplished on the first mission and went a long way in stabilizing the temperatures, which dropped to 90° (fig. 38)), and (2) a twin-pole sail, which was deployed by the second crew using EVA by the astronauts (fig. 39). The second shield was more effective and lasted the life of the station.



Figure 38. *Skylab twin-pole Sun shade.*

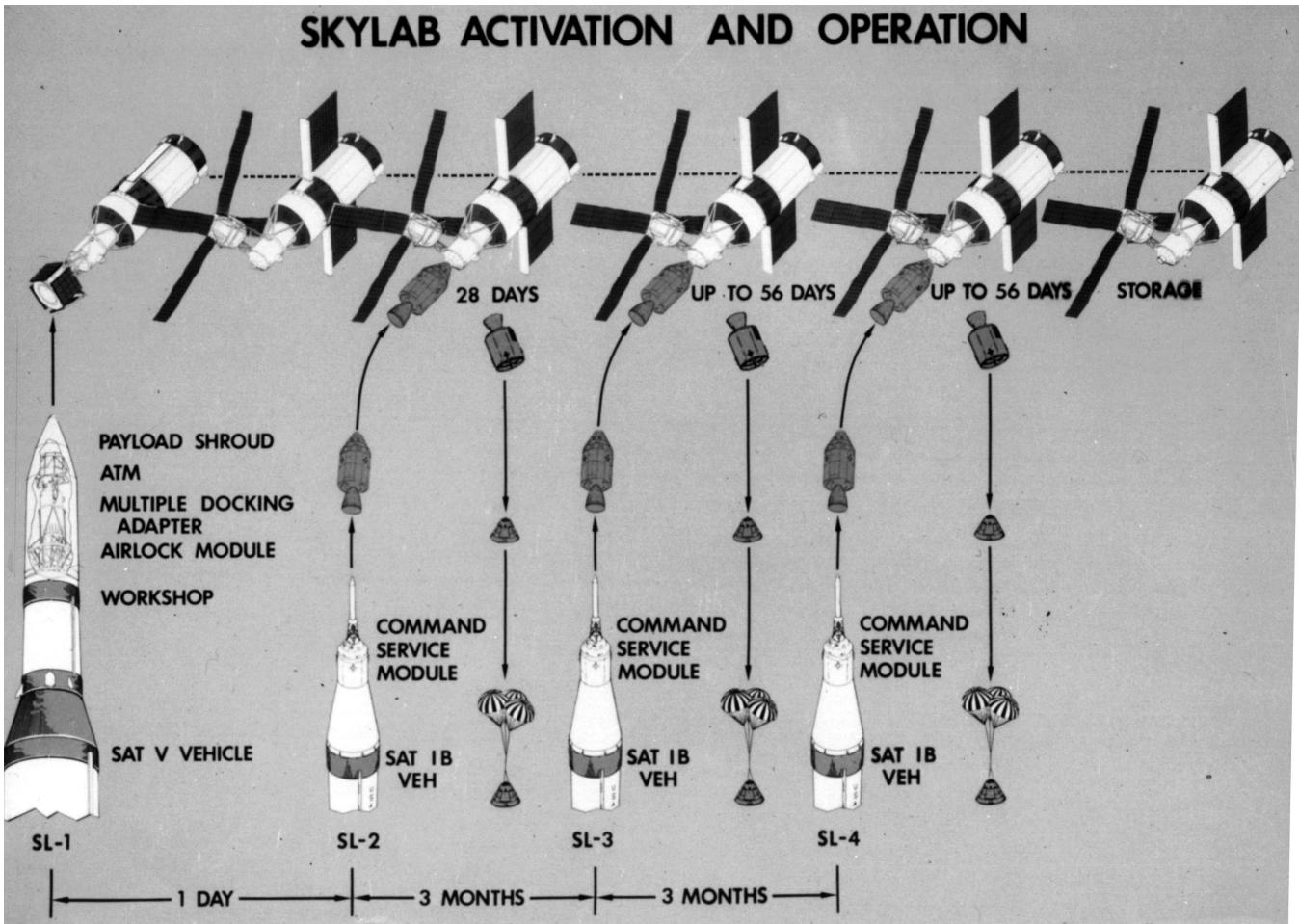


Figure 39. *Skylab* typical mission profile.

The second critical problem was to try to deploy the remaining solar wing in order to get enough power to run the workshop and free the ATM power system for solar observation. It was envisioned that some strap or debris from the failed debris shield was holding the wing. Various types of extendable tools, like those used by ground electrical crews, were designed and tested in MSFC's Neutral Buoyancy Simulator facility, as well as the crew was trained in the tank on how to use them. The simulator proved an extremely valuable means of approximating zero gravity, allowing training and understanding the system.

The crew, when on orbit, flew the command module around the workshop and observed the hung-up wing. They first flew the spacecraft in close and, using a pole out the window, tried to pry the wing loose with no success. They then tried to dock with the workshop but were unsuccessful. The next day they put on space suits and depressurized the spacecraft, then they disassembled and reassembled the docking adapter, after which they successfully docked.

The next day they entered the workshop and worked the damage, then deployed the parasol Sun shade as stated above. The ground personnel at Johnson Space Center (JSC) and MSFC continued to work out ways of releasing the solar wing. Fellow astronauts worked with the tools they had carried to orbit with the simulated problem in the Neutral Buoyancy tank.

On June 2, Conrad and Kerwin opened the hatch on the air lock module and moved out on the airlock shroud to assemble their tools and equipment. They tried without success to cut the strap loose that was holding the solar wing. Then Conrad made his way out to the strap hung wing with the cutter jaws. As he reached the scissors-type mechanism, the cutter severed the strap, moving suddenly, and throwing him away only to be saved by his tether. They next had to move the beam out a full 90°. This they did with a tether that Kerwin pulled and Conrad lifted in the middle (mechanical advantage), thus moving the beam out to the deployed position. The jolt knocked both of them away, but the tether contained them and they reentered the workshop. Once inside the workshop, it was possible to warm up the actuators and fully deploy and lock in the solar wing. The system could now be balanced and the solar observation and laboratory work begun.

At the start of the second mission, the second Sun shade was deployed during astronaut extra-vehicular activity (EVA). They attached the poles, then unrolled the shade, attaching it to the poles. Several concerns associated with this Sun shade were worked, and included dynamic excitation from the reaction jet control system firings, which potentially could cause large Sun shade deflections. All these problems were worked and put to bed. The new Sun shade worked in an excellent manner.

With these initial problems solved, the *Skylab* missions continued providing much science as well as information of long-duration space work by humans. *Skylab* set the foundation for the upcoming space station.

b. ATM Dynamic Response. A very interesting problem occurred due to the angles of orientation of the ATM during pointing relative to the MDA/workshop. The various observation angles changed the dynamic modes in a nonlinear manner, creating unusual demands on the control system logic. In order to work this logic problem, dynamic modal analysis was made for each generic telescope orientation. These mode shapes and frequencies were used by control analysts to change the control logic and verify it, allowing ATM pointing without bending mode instability. The result of this effort was a control system that operated for the three *Skylab* missions flawlessly as evidenced by the telemetered data, science gathered, etc.

F. Space Shuttle

The space shuttle, NASA's reusable launch vehicle/space platform, is one of the greatest design and operational achievements. Being both a manned spacecraft and a launch vehicle, with all but the external propulsion tanks being reusable, introduced many conflicting requirements and thus design trades. The problems that resulted and their solutions are the subject of this section. They will be discussed under the headings: (1) Space Shuttle Systems, (2) Space Shuttle Main Engines, (3) Solid Rocket Boosters and Motors, (4) Orbiter, and (5) Spacelab/Spacehab System.

1. Space Shuttle System

a. Characteristics. The space shuttle vehicle is composed of an expendable external propulsion tank (lox and hydrogen), a reusable orbiter with liquid main propulsion engines (SSME's), the orbiter maneuvering system (OMS), two SRB's (partially reusable), and the various payloads (fig. 40). The payload maximum size is 15 by 60 ft. The maximum weight depends on the desired orbit. The orbiter has a stay time on-orbit up to 2 weeks with a crew of up to five. The total weight is approximately 4.5×10^6 lb as shown on the schematic in figure 40. The liquid propellant capability is 1,391,144 lb 19,571 ft³ lox and 233,431 lb 52,908 ft³ hydrogen (fuel).

Payload usability is enhanced through the development of the Spacelab system, which has both habitable modules with experiment racks, thermal conditioning, etc., and platforms in nonhabitable regions for mounting experiments (section III-H). The remote manipulator arm allows working on or retrieving a satellite. The life support system, although important, is not addressed in detail. Extensive

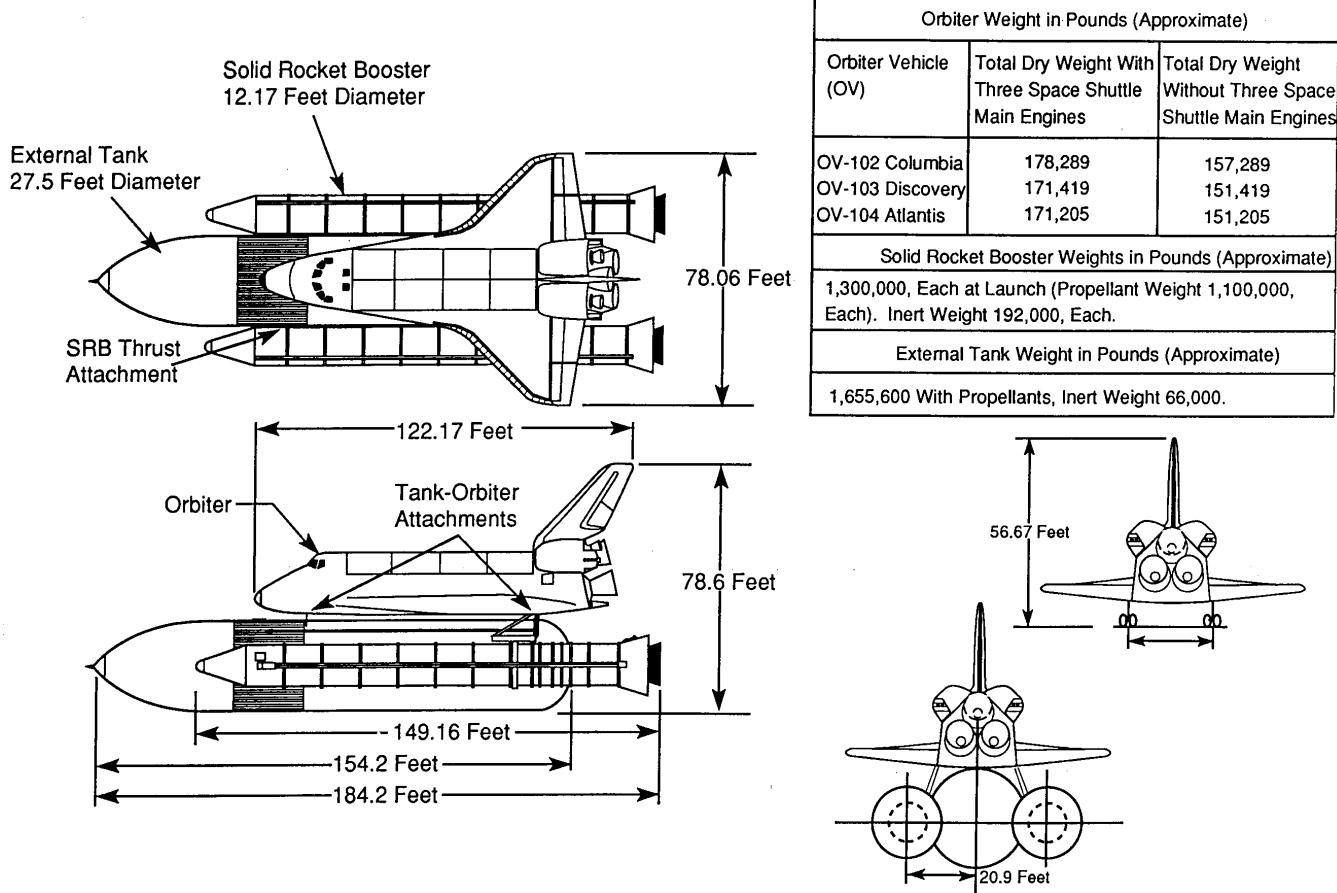


Figure 40. Space shuttle configuration.

assembly, launch processing, checkout facilities, equipment, processes, tools, and launch facilities exist, but are not a major part of this paper.

Work accomplished during phase B of shuttle demonstrated the strong interdisciplinary coupling of the shuttle system. Early trade studies (fig. 41) indicated areas that needed attention, such as trajectory, thermal, loads, control, performance, and launch operations interaction. Many developmental issues and problems surfaced (appendix) that required the use of multidiscipline teams (a pseudo type of concurrent engineering team) for resolution. Technology made many advances from the time of phase A until the present time. The two major technology drivers were the orbiter reentry thermal protection system and the high performance reusable liquid SSME. The evolution of the technology, processes, analysis, and tests accomplished on the shuttle are very informative.

Typical trajectory and mission profiles are shown in figures 42 and 43, which show the basic event sequences and the pertinent parameters.

The shuttle is, and has been, a marvelous machine. It represents the best in a multiple-purpose, finely-tuned, complex space system designed to meet very demanding requirements. Although its cost per launch did not meet expectations and the operations are more complex than desired, its 60-plus manned missions illustrate its capability. A recent one was the capture and repair of the Hubble Space Telescope. What a feat that turned out to be! Now the repaired HST meets or exceeds most of its original goals.

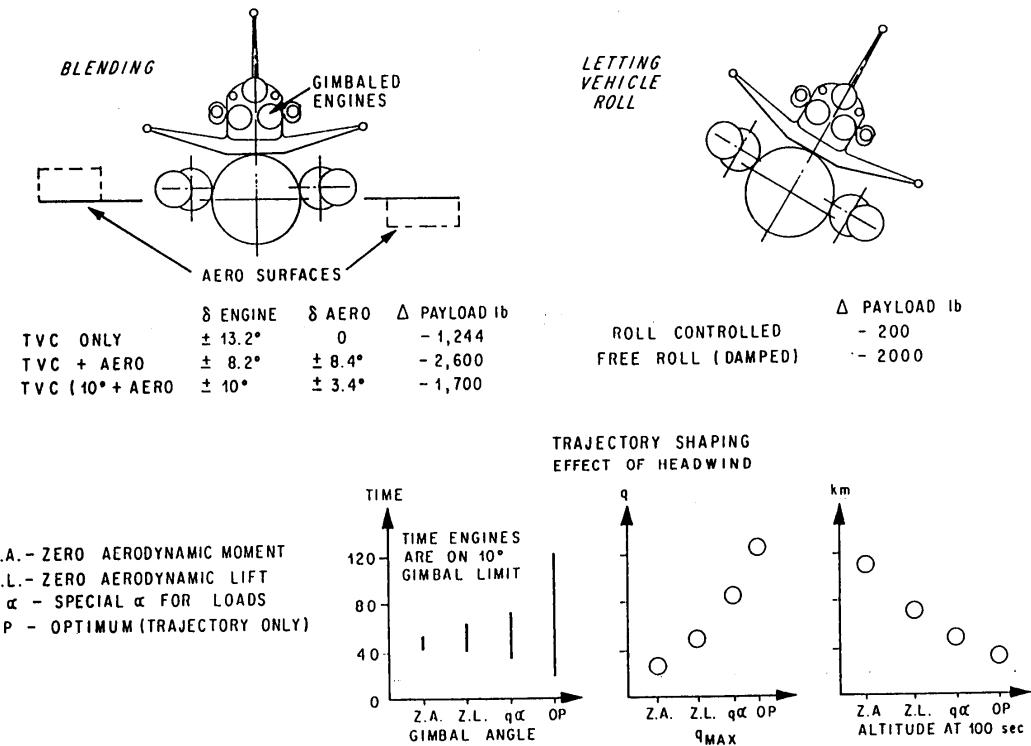


Figure 41. Space shuttle trades/sensitivities.

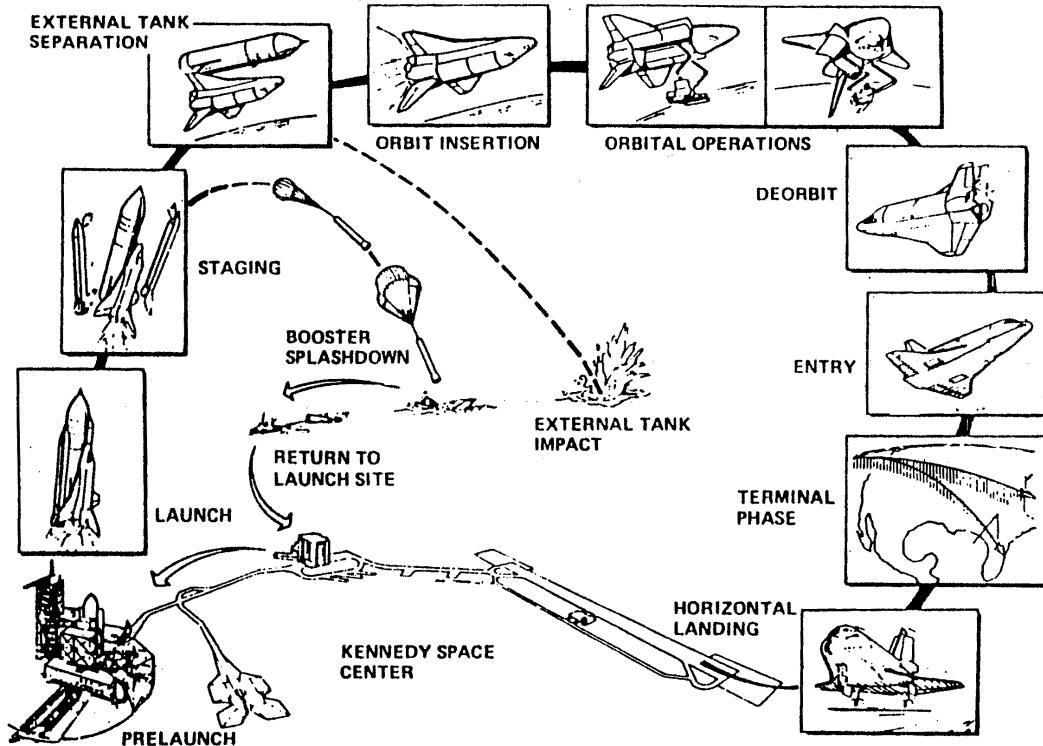


Figure 42. Space shuttle typical mission profile.

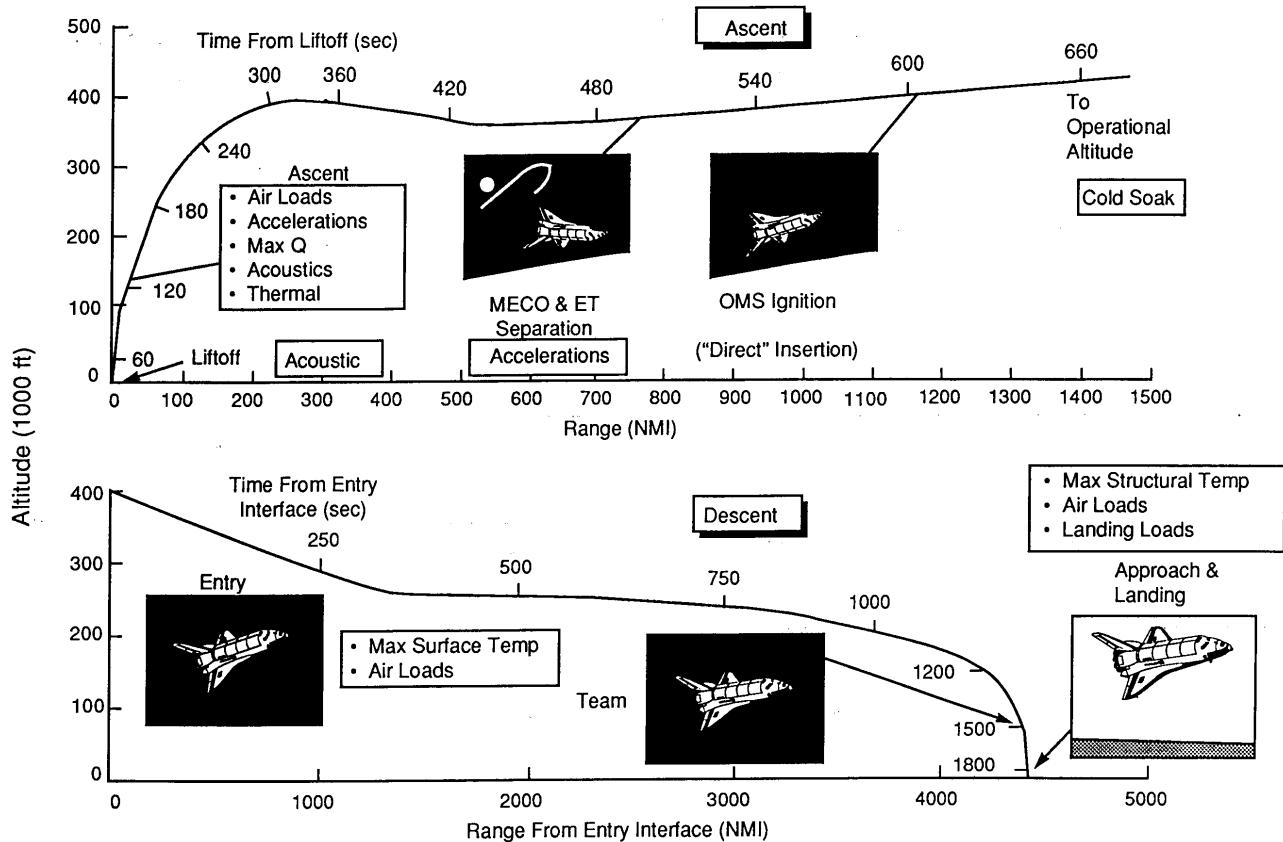


Figure 43. Space shuttle typical ascent and descent trajectory profiles.

b. Evolution/Creativity/Innovation. The development of such a complex machine required the best in engineering innovation, creativity, and common sense. Management of such a complex government-industrial teaming also required the same innovation, etc. At the same time, the shuttle marked a degenerative-trend use of suboptional off-the-shelf equipment, rather than special devices designed to specification. The Saturn computer was, for example, many times more reliable than the collection of processes used to guide and control the shuttle. However, in many ways the shuttle tasks were easier than many Saturn tasks. Stabilization and control and structural interaction differences were much more mobile for Saturn.

The two driving technologies required and developed for the space shuttle concept were the orbiter reentry heating protection system and the high-impulse reusable liquid engine. Other technologies also matured. Dynamic testing technology moved into the multirandom excitation, as well as the time domain, as an enhancement to the frequency domain. Scale-model technology was enhanced through a 1/4-scale testing of the total shuttle system. Integrated/multidisciplinary analysis was a key technology, i.e., control, performance (flight mechanics), loads, and thermal were highly coupled. Many innovations to accomplish this task were developed to support the program.

Understanding the physical phenomena of ignition overpressure and designing a suppression approach was not only a key technology area, but the innovation used in arriving at a fix. Testing of a 6.4-percent scale-model shuttle propulsion model (fig. 44) and its enhancement allowed both the physical understanding and the assessment of the water spray suppression and water troughs. This scale-model hot-fire propulsion model was started as a technological development for shuttle during phases A and B, and became the tool for fixing the STS-1 overpressure problem. Innovative water curtain testing was developed and showed that water blocks overpressure waves, thus the baselining of water troughs as

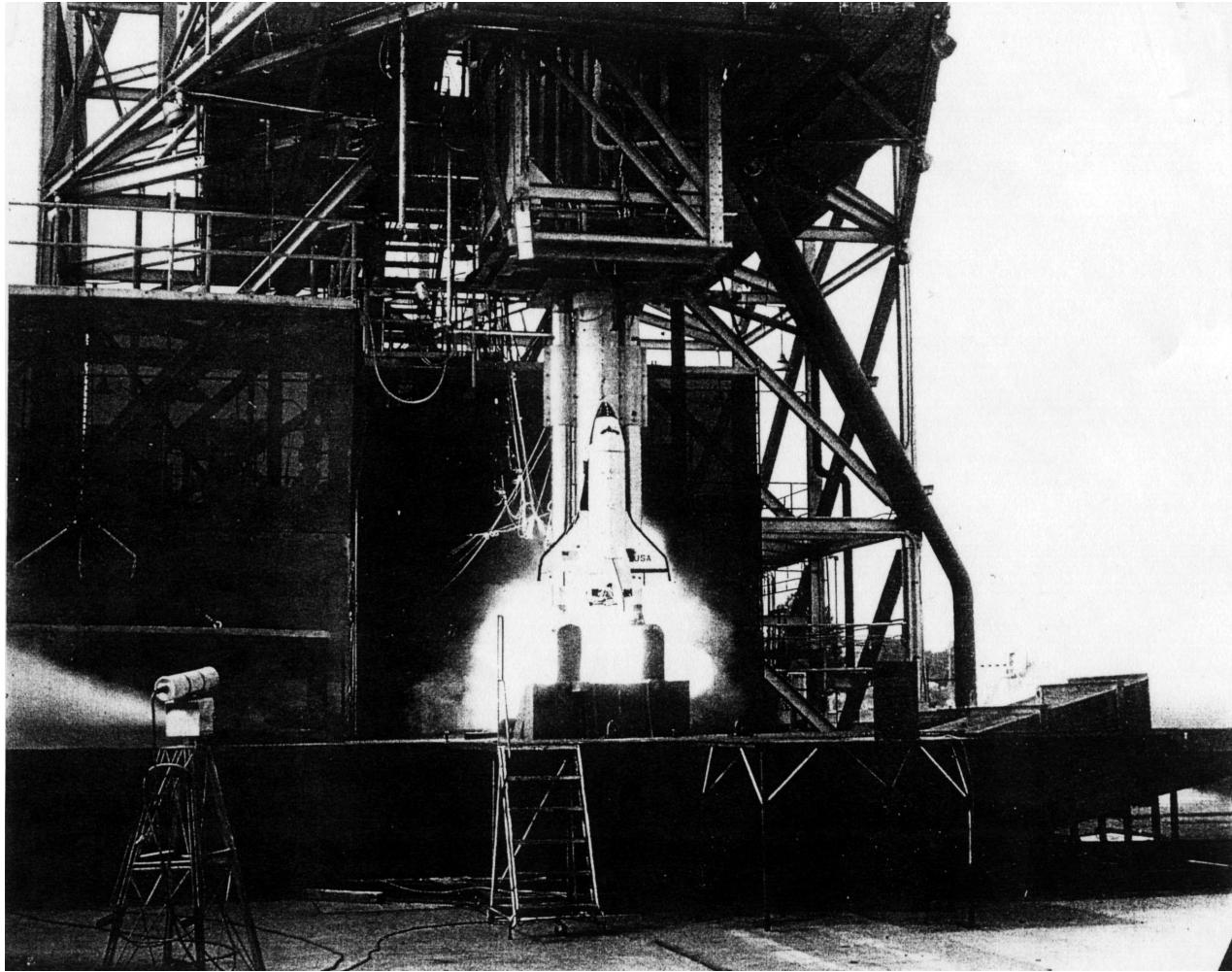


Figure 44. Space shuttle 6.4-percent propulsion model.

an insurance solution to the water spray injections (fig. 45). The solution of this problem also illustrated the complex technical management problem. During the development of the fix, it was thought that in the close vicinity of the SRB's permanent steel plates could be installed to block the overpressure wave. After moving along with the design, water spray, troughs in the drift holes, and steel plates around the SRB's, the team leaders kept questioning the feasibility of the steel plates. Eventually during this repeated questioning, data were uncovered that showed that the steel plates would reflect the SRM thrust and destroy the aft heat shield. Small water troughs replaced the steel plates, averting a loss of the vehicle. Innovative leadership/management is a requirement commiserate with the technical.

The SSME, with its high-performance requirements (454-s specific impulse (Isp)) and a 55-mission life coupled with geometric and weight constraints, developed fatigue and fracture issues that led to the development or enhancement of several technologies. Dealing with hardware life capability issues required a detailed characterization of both structural models and environments, which pushed development of computational fluid dynamics technology. This technology started, evolved, and matured in 10 years, greatly enhancing the maturity of the SSME. Finite element structural analysis matured during this time. Verification of environments required the development of special instrumentation for flows, acoustics, and vibration. Internal and external diagnostic instrumentation had to be developed. Data diagnostics (particularly dynamic) developed rapidly as well as data basing,

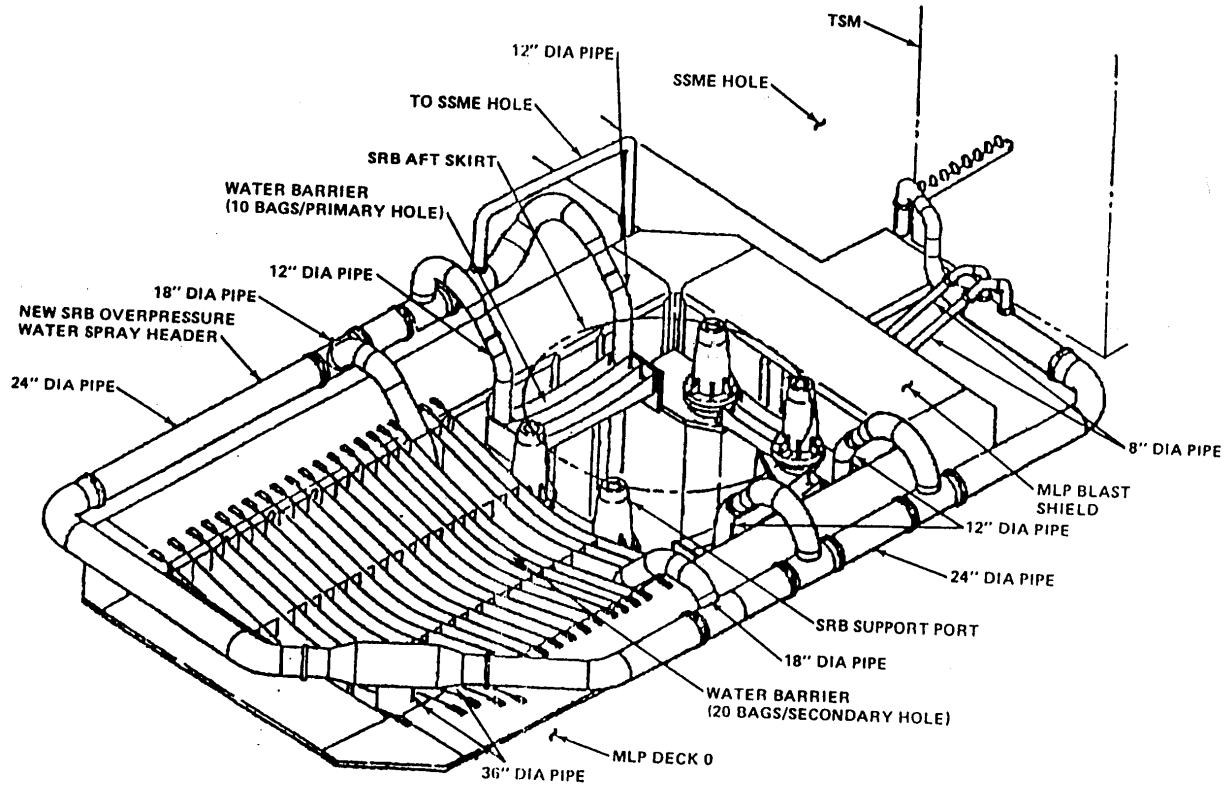


Figure 45. Space shuttle water troughs.

providing the ability to screen hardware and put a quantifiable maintenance and refurbishment program in place. This was true for both flow and mechanical vibration. Automation of the vibration data base allowed application of the established data base, with its excitation forces to new environments, processed components quickly through sealing and similitude laws automatically in the computer code. New criteria for shock and vibration testing can be generated in hours, where it used to take weeks. Health monitoring systems and engine shutdown systems were also based on these data bases. Cold-flow testing for benchmarking the CFD models and verifying environments reached a high level of sophistication. These facilities/technologies are both air and water. Normal fracture control technology evolved to include high-cycle fracture mechanics and an assessment approach for residual life. Combining low- and high-cycle fatigue was a maturing technology. In order to handle the high temperatures and pressures, new materials as well as processing techniques evolved. Castings that eliminated welds solved many fatigue and fracture problems before they occurred. The problems experienced on the engine required indepth understanding of very technically complex problems, which required formal approaches for studying the problem. Fault tree analysis complemented with logic diagrams and formal action items matured into a very useful tool or process.

Thermal protection systems in addition to the orbiter were major challenges. Hand-applied glue used on parts and the spray-on insulation for the ET were big adventures. The heat shields for the SRB and orbiter boat tail had to accommodate gimbaled nozzles and were an extension of Saturn technology. Plume heating was still an issue, but was in the realm of predictability using prior data, wind tunnel tests, and hot-fire solid and liquid engine ground tests.

Aerodynamic predictions were a major problem due to the unsymmetrical configuration between element-to-element tunnel flow and propulsion system plume interaction. The aerodynamic loads were excessive and had to be reduced. This led to elevon load relief and cross-coupled load relief using pitch, yaw, and roll control. The asymmetry of the vehicle also required the development of a vector wind synthetic profile to account for both shear and gust and the changing wind direction with altitude. Detailed measured monthly wind ensembles were enhanced, as well as the ability to accomplish measured wind Monte Carlo vehicle response and loads analysis.

A big technology that was developed was the day-of-launch I-load update (2 h prior to launch) that biased the trajectory to the last measured wind profile to increase launch probability and reduce loads. To accomplish this, the wind sounding system developed a new technology called LIDAR to achieve the wind data and analysis tools requirements, and to deal with required failure modes (for example, engine out) in conjunction with other parameter variations. An innovative technique was developed to use 1σ variations of the other parameters with an engine failure occurring 5 s prior to the wind gust of the synthetic profile. Therefore, the combined probabilities of the failure and parameter variations were near 3σ . For lift-off loads calculations, certain parameter variations changed the time sequence of the system, precluding the use of the A-factor approach developed for Saturn. To overcome this, all parameters that did not affect timing were RSS'ed in conjunction with a 1σ variation worse-on-worse of all parameters that affect timing.

Performance sensitivities of the one-and-a-half-stage concept to parameter variations led to the use of the wind data in a mixed fashion. Fifty percent of the wind shears and gusts were treated deterministically, while the effects of the other 50 percent were RSS'ed along with the system parameters. Because wind direction change was also important to the vehicle response, a vector wind synthetic profile was developed.

The space shuttle was the first vehicle that was coupled in the pitch plane due to center of gravity (cg) and thrust offsets, precluding the normal point mass trajectory/performance optimization programs. An innovative moment balance (programmed control forces) was developed that allowed a simplified approach for trajectory optimization.

Because the shuttle is a winged vehicle, the high dynamic pressure regime was a major analysis problem to adequately predict rigid-body vehicle response to winds and thrust-generated, vehicle-induced loads, which included all system parameter variations as well as wind statistics. This was accomplished in an efficient manner through the creation of squatcheloids (Tom Modlin's work) which was $q\alpha$ and $q\beta$ plotted continuously for nominal parameters as the wind direction varied. These squatcheloids are generated for each Mach number. A set is also generated for 3σ system parameter variations and becomes the basis for the design criteria (figs. 46 and 47). This work has been extended to the use of alpha, beta, and "q" combinations instead of the rigid boundary of the squatcheloid, providing more freedom in trajectory shaping and the tailoring of the day-of-launch I-loads updates to increase launch probability.

Many other innovations and technologies have evolved such as the use of animation graphics to enhance understanding of dynamic response; colored graphics to pictorially show stress, flow, and thermal fields; CAD/CAM systems to automate design and manufacturing; information systems/electronic data transfer; and finally, the evolution of probabilistic/risk assessment technologies to the point where it is impacting design, verification, and operation space systems.

The shuttle went beyond proof testing of propellant tanks, etc., to total fracture control. In the ET, one-half mile of welds required pre- and post-proof testing inspection. Observed flaws had to be dispositioned for flight use by analog testing. The SSME evolved along a different path. High-strength material was required to meet the high-performance requirements which were:

- Two-stage combustion and pump system producing high pressure and temperature
- High Isp thrust coupled with minimum weight and volume requirements.

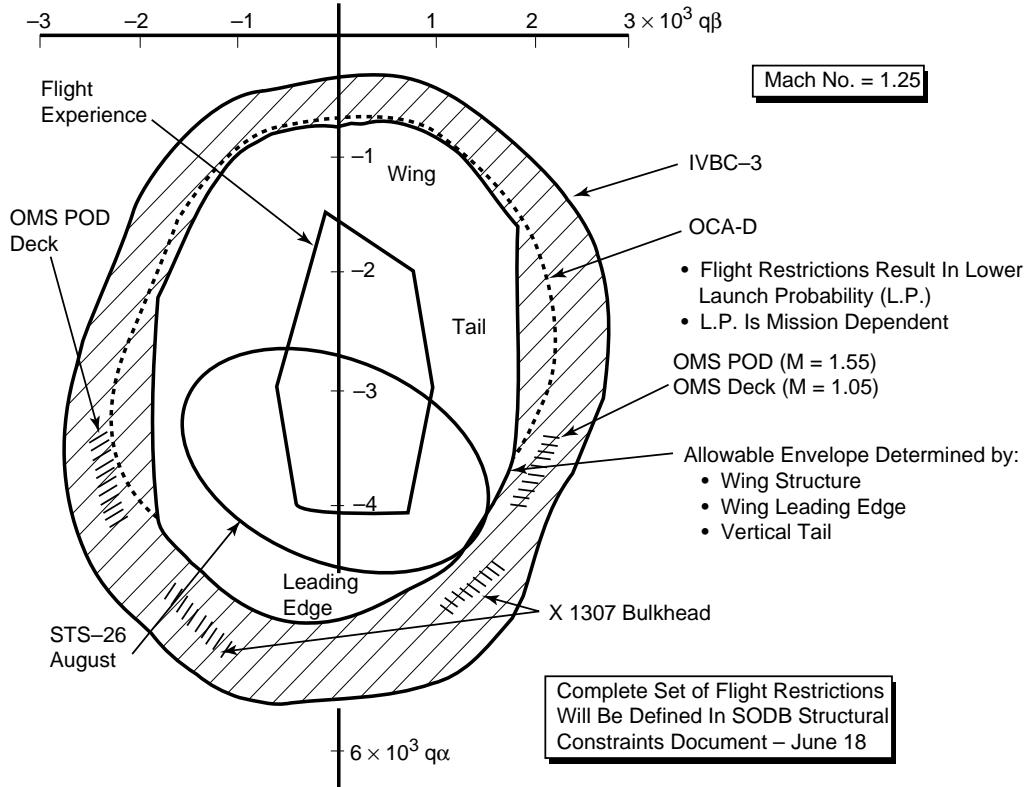


Figure 46. Space shuttle Mach 1.25 squatcheloid.

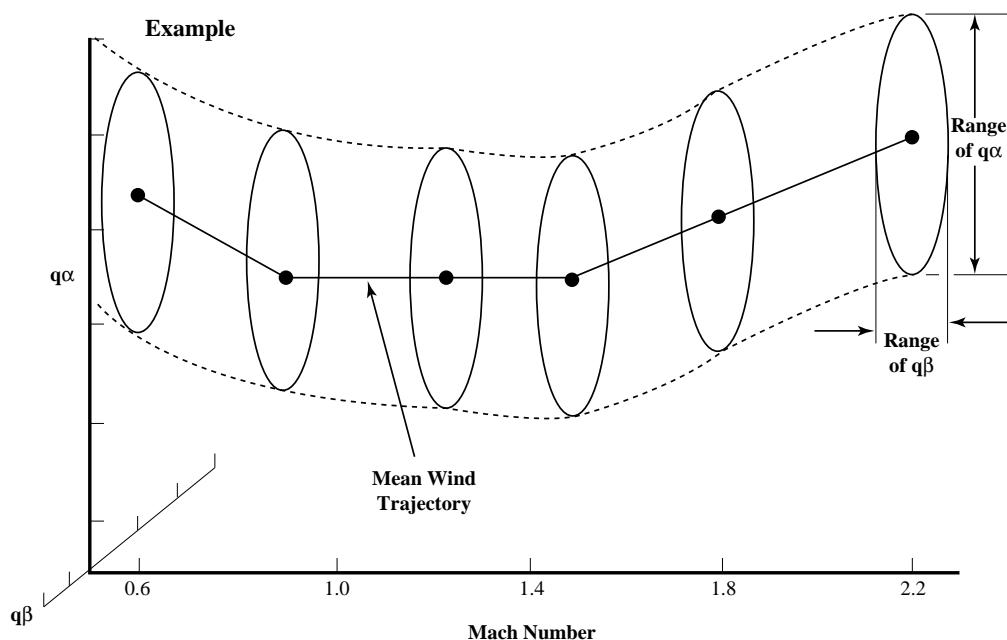


Figure 47. Space shuttle family of squatcheloids.

These resulted in weld problems. To save cost, formal fracture control was initially not a design requirement; following the pattern of the Saturn engines, F-1 and J-2. The early problems resulted in critical components having fracture assessment. Additional problems resulted in more components being added, later followed by a weld assessment. This effort has resulted in residual life assessment, probabilistic/risk assessment, and the extension of fracture control to include high-cycle fracture in addition to low-cycle fracture. Rocketdyne and its fracture control leader have made major technology contributions in fracture control state-of-the-art. The first major use of damping seals, invented by George Von Pragneau, NASA, had its first use in the turbopumps, which were used to damp whirl problems (fig. 48). This seal concept later evolved by Rocketdyne into a hydrodynamic bearing for the lox pump and was successfully tested six times.

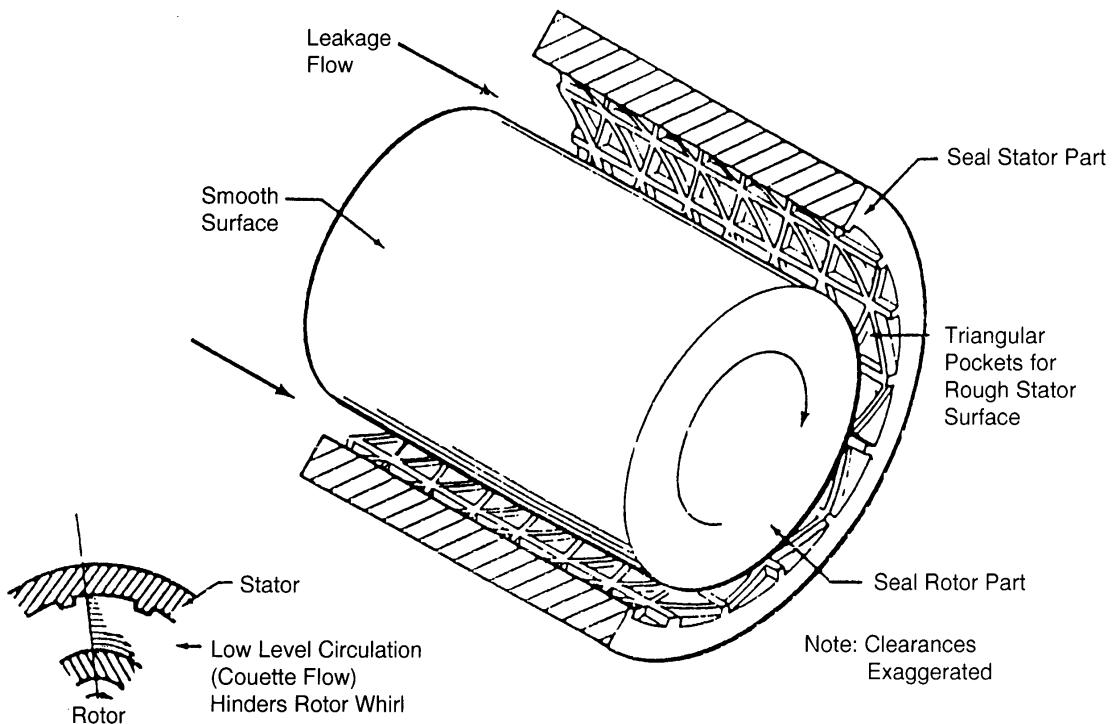


Figure 48. SSME lox turbopump damping seals.

Highly redundant TVC systems and APU power systems were developed. As fickle as hydrazine systems are, the four that fly on the SRB on every flight have worked perfectly. Parts of these systems are quad redundant, while others have only dual parts. It turned out to be an example of good system engineering.

Performance sensitivities led to several weight-saving programs. The ET made a major effort to reduce the weight approximately 10,000 lb, producing approximately the same level of payload performance improvement.

The innovative design of the total shuttle system used a unique load split or load path where the SRB loads were taken out at the forward attach point of the SRB to the ET. This was through all the I-beams in the tank inner stage (lox tank forward of load fuel tank to the rear), while the SSME thrust

was taken out at the rear of the ET, producing a net load of the difference. The I-beam also tied the two SRB's together. In addition, the hydrogen tank skin thickness was tailored circumferentially and longitudinally to the stress field. A portion of the ET that was designed by well-defined loads was based on a safety factor of 1.25 instead of the normal 1.4. The cryo shrinkage of the ET when filled created two problems. First, the radial shrinkage loaded the SRB (held down firm to the launch pad) against the tank (punch load). The SRB-to-ET aft struts were pretensioned to relieve this problem. The longitudinal shrinkage also increased loads. Second, if the tanks were perpendicular before filling, they became 7° off perpendicular when filled. Therefore, the struts were biased 7° off perpendicular before tank filling so that they were perpendicular during operations. The lox tank was located forward to reduce the c.g. tracking excursion by the thrust vector and reduced thrust loss due to the cosine effect of the thrust vector angle.

The innovative use of timing delay of SRB ignition allowed the minimization of the twang energy stored in the vehicle due to the SSME's being at full thrust. Further, the SRB design loads were chosen to be worse-on-worse, building robustness and precluding downstream redesigns as loads changed. This was a good system choice because the performance loss was only one-twelfth. All other elements used time-consistent loads due to the one-to-one performance impact of weight. Also, the SRB was not initially designed for water impact. A 5-percent attrition was accepted, thus no large weight impacts.

The programmatic management of the complex technical interaction was an extension of the use of working groups and panels started in the Saturn program. The discipline interaction was much more pronounced and sensitive than the prior vehicle, creating the need for strong integration of the various disciplines, subsystems, and elements. As this need became more and more apparent, various integration approaches were tried. The integration working group approach was chosen. In terms of the launch vehicle and the propulsion system, this was a group consisting of all the NASA centers' interacting disciplines, leads, elements, chief engineers, and contractor equivalents. The groups working the launch vehicle were named the ascent flight system integration group and the propulsion system integration group. The key philosophy was that the groups were to be exchange forums and recommending groups—not project decision groups. Problems were to flow down to the groups from the projects, up to the groups from their members supporting engineers, and from the groups themselves. Although structured through co-chair leads and formal action items that could be assigned to any group or individual, it was completely open. This created excellent communication and exchange, bringing many issues to resolution. This type of working group is obviously not the same as the Product Development Teams (PDT) in vogue today, although it carried out many of the concurrent engineering functions and also established excellent communications between NASA Centers and their contractors because all were a part of the working group and their supporting discipline panels. Whether the use of PDT's in their concurrent engineering focus eliminates the need for this type of working group or its equivalent is an open question. For large complex systems, however, some means of integrating and coordinating the system focus across the product developments is mandatory. For the shuttle, this approach was very effective in resolving the major multidisciplinary system issues inherent in this type vehicle.

c. Problems. As the space shuttle evolved, reality overcame dreams and visions, thus performance problems developed. These were of a wide variety, including payload to orbit capability, cost, operational complexity, refurbishment, and maintenance. When a project is faced with a performance issue (payload to orbit), there are three options available for dealing with the problem: (1) weight reduction (structures, consumables, propellant reserves etc.), (2) performance enhancements (thrust, Isp, launch site, burn time, etc.), and (3) operational procedures and constraints (reduced requirements, environment reductions, etc.). As has been discussed in this section, the shuttle employed all three techniques in order to meet performance requirements. This was accomplished in a very innovative and balanced manner from the reduction of safety factors on the ET for well known loads to the high-performance motor (HPM) SRM (3,000 performance enhancement, to the 104-percent power level SSME. Table 3 is a matrix of the problems that caused performance degradation and the average hit followed by a listing of the solutions used to gain performance.

Table 3. Shuttle performance degradation/performance enhancements.

1. Orbiter design against real requirements leads to not meeting weight requirements	30,000 lb
2. Environments (loads) increases due to employing quartering high q winds. Several impacts resulted: – Baseling monthly mean wind biasing for design – Three-axis and elevon load relief baselined – Increased analysis and testing – Aerothermal/thermal protection system impacts (SRB and ET) – Performance loss from path deviations	5,000 lb
3. SRM fixed nozzle was redesigned to a flex gimbaled nozzle for adequate vehicle control.	
4. Lift-off loads increases – Staggered SSME start and shutdown – Ground winds constraints – Weight increases/redesign shuttle elements → (individual payload impacts not documented) – Pre-tensioned SRB/ET struts	1,200 # eq payload
5. Missed aerodynamic predictions/STS-1 lofting – Trajectory constraints q , SRB separation, a trim – Performance loss → – TPS impacts ET and SRB – Protuberance (ET) redesign/rectification – Reduced launch probability winter months → – Orbiter wing mods (leading edge) – Day of launch I-load updates – Flight-derived dispersions	5,000 II 65 percent
6. Isp not meeting design goals – SSME → – SRM →	2 ^{1/2} s 1 s
7. STS-1 SRB ignition overpressure – Modified water injection system (into thrust buckets) – Water troughs over drift hole – Payload loads increases (limited redesign/revivification)	
8. Orbiter tile debond and debris damage	
9. Other – Landing gear and brakes – Engine upgrades – SSME fatigue and fracture – ET fracture control – Isp loss for plugged lox post – SRB recovers • Water impact damage • Acoustic (reentry) • Thermal (reentry)	1 s

Table 3. Shuttle performance degradation/performance enhancements (continued).

Performance Enhancement	
1. SSME	
– 104-percent thrust nominal	3,000 lb
– 109-percent abort	
– Redesign weight reduction	1,000 lb
2. ET	
– Safety factor reduction to 1.25	10,000 lb
– Redesign (super light weight tank)	8,000 lb
3. SRM	
– High-performance motor	3,000 lb
– Burn rate increase	1,500 lb
4. Orbiter	
– Weight reduction design changes	≈10,000 lb
5. Flight-derived dIspersions	

In addition to these performance type issues, several other major problems occurred during development not the least of which were the orbiter tile debond, debris damage to the tile, SRM ignition overpressure, and aerodynamic deviations.

d. Future Implications. Any future system will have to deal with realities of performance in the same three categories. It is prudent, however, that up front the systems sensitivities be understood and robustness designed into the system¹⁹ in order to eliminate, as much as possible, the impacts.

Although the space shuttle is a marvelous and successful machine, there are some downsides. For example, as a result many sensitivities of not only the shuttle system, but also its elements and components as well, have led to costly launch delays, maintenance and refurbishment issues, operational hands-on labor (touch labor), and cost. A list of the shuttle flight data and an in-flight anomaly list is published by JSC in JSC-19413 and is revised periodically. A similar report is published at MSFC in MSFC-1600 Rev C. They list the mission, Space Transportation System (STS) number, orbiter number, event, impact, and remarks. The problem list goes from winds aloft to auxiliary power unit (APU) failures to hydrogen leaks. A study was done by Analex Space Systems, Inc., Kennedy Space Center, FL, of the “Ability of Space Shuttle to Launch on Time.” This is a comprehensive report, again dealing with all the launch delays resulting from weather to sensor failures and everything in between.

The shuttle as a system is, therefore, very complex, requiring much touch-labor to operate, and is limited in performance by its constraints; yet it is indeed a remarkable machine, a marvel, a wonder. Future systems, however, need a different approach that is reliable, cost efficient, and, therefore, robust. The appendix summarizes the set of problems the author has dealt with in the space shuttle arena.

The development of space shuttle provided many technologies that have application elsewhere and in the future. The following is a partial listing:

1. Solid rocket motors gimbal nozzle (flex seal)
2. SRM ablative, refurbishable nozzle

3. Lightweight orbiter thermal protection tiles
4. Acoustics and ignition overpressure suppression techniques
5. Day-of-launch I-load update procedures
6. Active multiaxis load relief
7. Tailored/adaptive SSME throttling
8. ET spray-on insulation (propellant tanks) (robotics)
9. Robotic welding
10. High-strength materials
11. Hydro-static turbopump bearings
12. Silicon nitride ball bearings
13. SRB recovery
14. High-cycle fracture mechanics techniques (residual life)
15. High Isp liquid hydrogen engines
16. Simulated life fracture testing (analog testing)
17. Digital control
18. Computer assisted design/manufacturing (CAD/CAM)
19. CFD
20. Computational mechanics/finite element modeling
21. Microelectronics.

The next section will deal with the shuttle elements.

2. Space Shuttle Main Engine. The following section is a collection of materials, data, charts, etc., from several sources associated with the SSME. These sources are (1) MSFC and Rocketdyne failure investigation reports, (2) Rocketdyne presentations, (3) MSFC/Rocketdyne special problem resolution teams, (4) MSFC and Rocketdyne studies and reports, and (5) MSFC and Rocketdyne technical society papers and professional journal articles. The engine design work, testing, and operations were Rocketdyne's responsibility. MSFC was the responsible NASA Center.

a. Characteristics. The SSME is the highest efficient liquid propulsion engine ever built. Its complexity and performance are unequaled. It is an example of the degree of craftsmanship possible in the design and operation of a high-performance system. As discussed in previous publications, these characteristics, in conjunction with the external leveled constraints, produced problems, many of which were unexpected because of the high-performance requirements. Bob Biggs, Rocketdyne, wrote an excellent paper tracing the evolution of the engine, providing much insight into the engine and the program. Others have dealt with various aspects of its development and problems.²⁹⁻³⁴

The SSME is a high-performance, throttleable, liquid oxygen/hydrogen space vehicle engine. The high Isp and thrust is achieved through a staged combustion power cycle, with high combustion chamber pressure using a dual-stage pump and combustion process. The engine operates at a mixture ratio (lox/liquid hydrogen of 6:1 and a chamber pressure of approximately 3,000 lb/in² absolute) to produce a sea level thrust of 375,000 lb and a vacuum thrust of 470,000 lb. The engines are throttleable over a thrust range of 65 to 109 percent of design thrust level. This allows high thrust initially with throttle down capability to control maximum vehicle dynamic pressure (max q) and maintain vehicle acceleration at a 3-g limit for crew safety. In the staged combustion cycle, the propellants are partially burned at high pressure and relatively low temperature in the preburners, providing power to drive the high-pressure pumps. They are completely burned at high temperature and pressure in the main combustion chamber before expanding through the high-area-ratio nozzle. Liquid fuel or oxygen is used to cool all combustion devices. The hydrogen-rich hot gas from the preburners is used to drive the high-pressure pumps. To put this energy concentration in perspective, the high-pressure fuel pump is a three-stage liquid hydrogen pump driven by a two-stage gas turbine rotating up to 38,000 r/min, and producing 70,000 h.p., all contained in a space of approximately 18 inches in diameter and 2 feet long. The output from the high-pressure lox pump is a pressure of 4,600 lb/in² absolute at a flow rate of 795 lb/s, producing very high dynamic pressures. It is clear from this brief description that this system produces very harsh environments. Temperatures range from 1,700 to 30 °R; pressures up to 7,500 lb/in² absolute; and dynamic pressures 60 times those normally experienced on the shuttle wing during ascent. Figure 49 is a picture of the engine, while figure 50 shows the basic flow and typical values. This, in conjunction with hydrogen environments, leads to susceptibility to corrosion, hydrogen embrittlement, high-cycle fatigue, and fracture mechanics issues. Problems have been experienced in all these areas during the engine development program.

b. Creativity/Innovation. Some of the innovation and creativity brought to bear during the history of the SSME was discussed briefly in the shuttle paragraphs. The development and use of damping seals to damp turbomachinery instability and high-cycle fracture mechanics are examples.

The main oxidizer valve (MOV) 7,000-Hz vibration problem was a classic example of acoustical/structural dynamic tuning and also how innovation can be used to solve a problem. The problem was a resonating edge tone of the valve/sleeve housing gap as the high speed lox flowed across it, exciting the sleeve and causing the seal to rub and catch on fire. In order to find where the excitation was coming from, a buzzing valve was put in the lox cold flow facility at MSFC and tested to duplicate the vibration experienced on the engine. Once the buzz was established, wax was put in the various cavities, one at a time to determine which was the excitation source. When the sleeve/housing gap was filled, the vibration stopped. The solution was to tailor a washer to fill the cavity. No vibration has occurred in the MOV since 1980.

Turbomachinery and its ancillary field, rotordynamics, has had to make quantum jumps in analysis, testing, and data basing. Instabilities such as whirl, bearing life degradation, rogue force, and excessive vibration are evils waiting to happen in high-performance, high-density turbomachines. The creative use of damping seals, an invention of George von Pragenau, basically solved the whirl problems and reduced the forced vibration levels. Extrapolating from the damping seal concept, Von Pragenau, in conjunction with Rocketdyne, developed and demonstrated a hydrodynamic damping seal bearing that not only provided adequate damping but also was the rotor support bearing. This damping seal bearing concept was demonstrated for the SSME lox pump in the MSFC engine testbed. A total of five hot engine firings was used to demonstrate this approach. The bearing lifted off early in the start sequence, eliminating the concern of rub in gaseous oxygen (gox) during startup and shutdown. An innovative technique for balancing the spinning parts had to be devised because it was generally not possible to balance the parts assembled into a pump. Rotordynamic analysis with many rotor and case modes became routine. Nonlinear rotordynamic analysis tools were developed and used extensively.

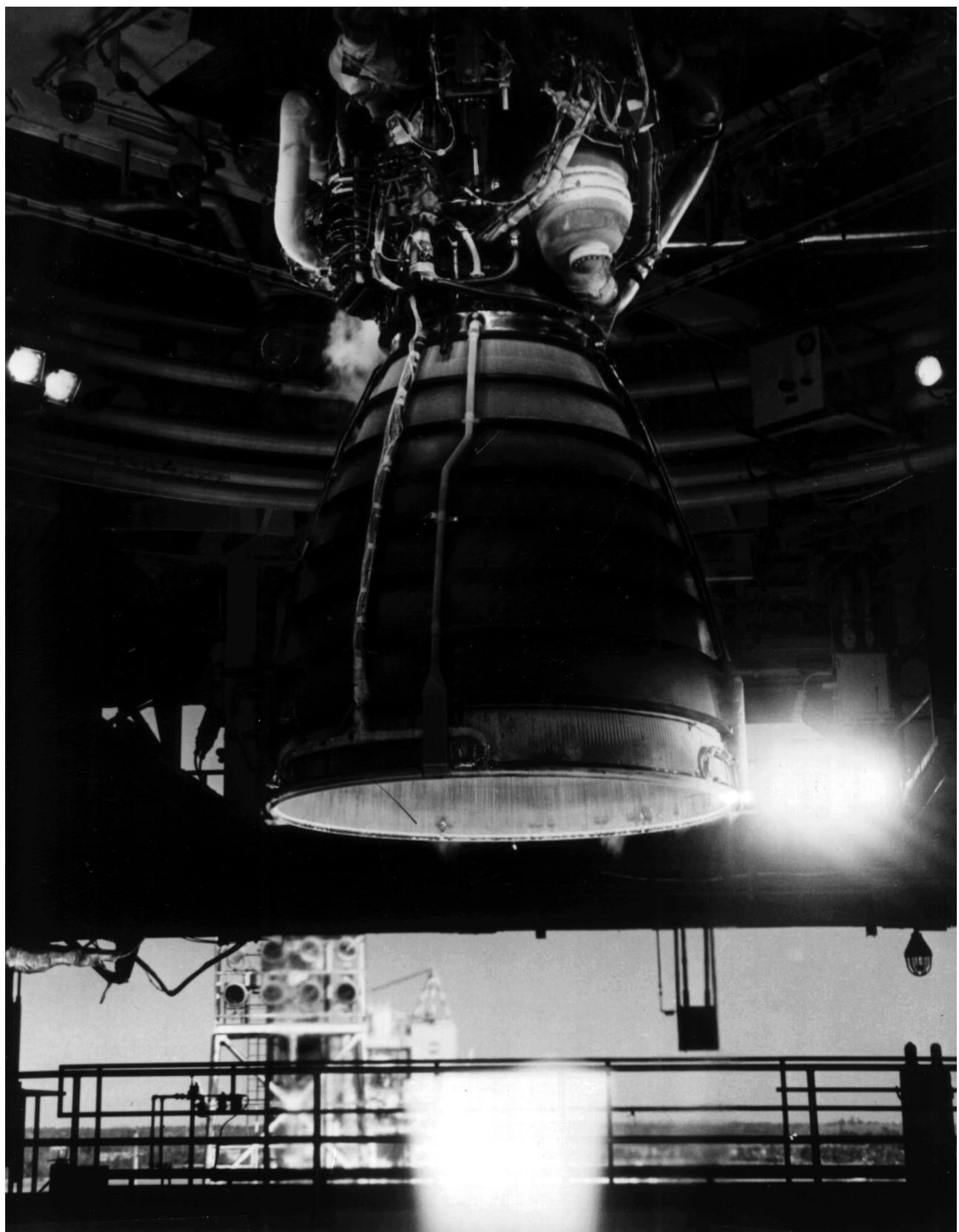


Figure 49. SSME configuration.

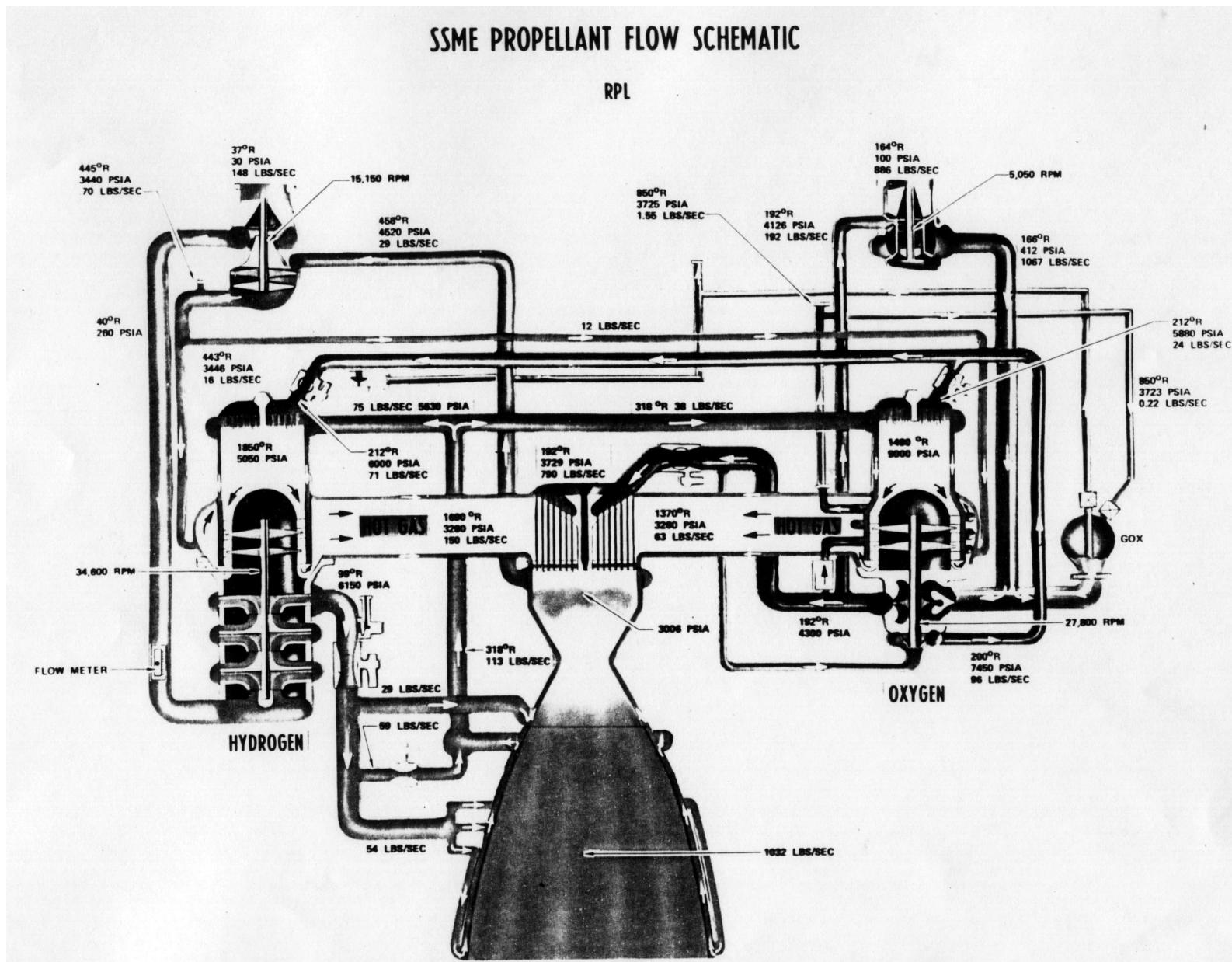


Figure 50. SSME propellant flow diagram.

Many other innovations were apparent as various problems occurred. The numerous major hot-fire development problems were studied and resolved using Government/contractor concurrent engineering teams, usually called failure investigation boards. Although the word “concurrent engineering” was not in vogue, the operating approach and team makeup were part of the characteristics of the “current” definition. (See reference 20 for a discussion of the approach.) The teams, in general, used fault trees or FMEA’s in a formal way to resolve very complex technically induced hardware failures. The use of these failure boards or teams was very effective and innovative. The use of concurrent engineering or ad hoc teams (tiger teams) has evolved to deal with complex technical issues such as fatigue problems, weld problems, and special problems such as 4,000 Hz, turbine blade cracking, bearing wear, and manufacturing issues. Major technology breakthroughs have occurred in the development of special materials and materials processing, such as the development and use of silicon nitride ball bearings to solve the turbomachinery bearing wear issues,⁶⁵ and single-crystal turbine blades to solve blade cracking issues. The development of CFD analysis techniques to predict the internal fluid and gas flow environment was also a major accomplishment. Parallel to this was the development of special air-and water-flow facilities to benchmark these CFD models. Detailed finite element dynamic and stress modeling reached its pinnacle for meeting accuracy requirements and performance in order to understand and solve the many fatigue issues. The extreme thermal environments and thermal gradients required the same level of sophistication. Materials characterization had to reach new levels due to the use of exotic materials, and the extreme environments created very small critical flaw sizes that drove many innovative, as well as more accurate, nondestructive evaluation (NDE) approaches. X ray, dye penetrant, ultrasonic, etc., were a part of these NDE advances. Internal instrumentation was developed to measure and quantify environments. Large advances were made in hot-fire data acquisition, data evaluation, and data banking; and electronic data transfer became standard practice. The engine system performance data and the high-frequency turbomachinery (vibration data) are good examples. The approach collects and evaluates vibration data from 0 to 4,000 Hz. Spectrums, ISO plots, RMS plots, mean and variances, etc., are produced and data banked. The historical data banks of physical data formulated in a probabilistic and reliability sense are used to manage hardware maintenance and to ensure flight safety. Data exist now for over 2,000 ground tests and 80 shuttle flights (three engines each flight). As a result of this large data base, decisions are made for flight based on expected demonstrated levels and hardware successes. In general, a piece of SSME hardware will not be flown with more accumulated time than one-half the ground demonstrated levels.

c. Developmental Problem Examples. SSME problems have taken many forms, some of which have been major failures.^{35–66} The flight history program is shown in figure 51. Figures 52 and 53 show the flight redlines required to safely shut down malfunctioning engines. In addition to the flight program, there has been an extensive ground development and verification test program involving 60+ engines and 2,000+ hot-fire SSME ground tests (individual engines), accumulating more than 748,000 s. This does not include the 10+ MPT engine cluster test and more than 80+ flights of three engines each. To date, there have been 36 major failures, all of which were investigated by an official investigation board with a self-contained report depicting the problem resolution and other recommendations. Figure 54 shows the 36 major failures as a function of calendar year. As the program has matured, major failures have decreased. Figure 55 is an engine schematic showing where the failures have occurred. Table 4 lists the 36 failures with other pertinent data such as cutoff parameters, failure cause, and resolution. Dennis Gosdin created these charts together with the assistance of Rocketdyne and MSFC engineers to serve as a historical data base. Many other problems have occurred that were not classified as major failures. One occurred in proof tests, some in vibration testing, and others were self-contained. Many were found analytically, others are the inherent problems of high-performance systems. In the sections that follow, appendices, and references, many of these problems are discussed—not to point fingers, because all have performed very well—but to teach what can be expected in developing high-performance systems and to point to areas for penetration and problem avoidance (total design) in new programs. All in all, these liquid propulsion systems provide great complexity and challenge. This can be understood by the range of technical areas these system encompasses such as: rotordynamics, thermal, flow, control, transients, stability, fracture, fatigue, combustion, exotic fuels, manufacturing, FMEA’s, testing, analysis, simulations, breadboards, tractability, etc. A few examples of these challenging problems follow; the complete documentation is reserved for the appendices.

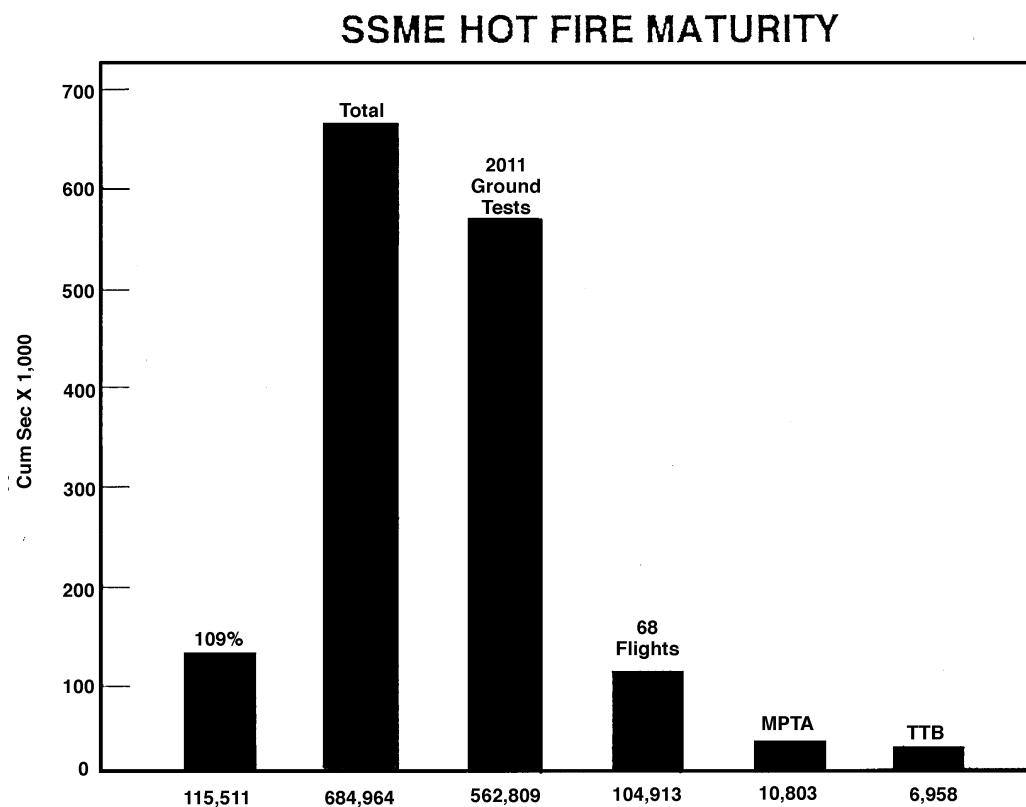


Figure 51. SSME flight history program.

Start Confirm Redlines

Parameters	Rationale For Criteria
• Main Chamber Pressure	• Detect Improper Ignition or Engine Control • Insure Proper Thrust Level Prior to Liftoff
• HPFTP Speed	• Insure Proper Engine Fuel System Conditions
• Heat Exchanger Supply Valve Position	• Insure Oxidizer Flow to Heat Exchanger
• Preburner Shutdown Purge Pressure	• Prevent Possible Hot Gas Backfill Into Oxidizer Supply With Purge Check Valve Failed Open
• Engine Status Word-Electronic/Hydraulic Lockup	• Prevent Liftoff With Engine in Lockup Mode
• HPFT Discharge Temperature	• Insure Adequate Margin For Flight Effects
• HPOT Discharge Temperature	• Insure Adequate Margin For Flight Effects

Figure 52. Start confirm redlines.

FIGURE 1. SSME FLIGHT REDLINES

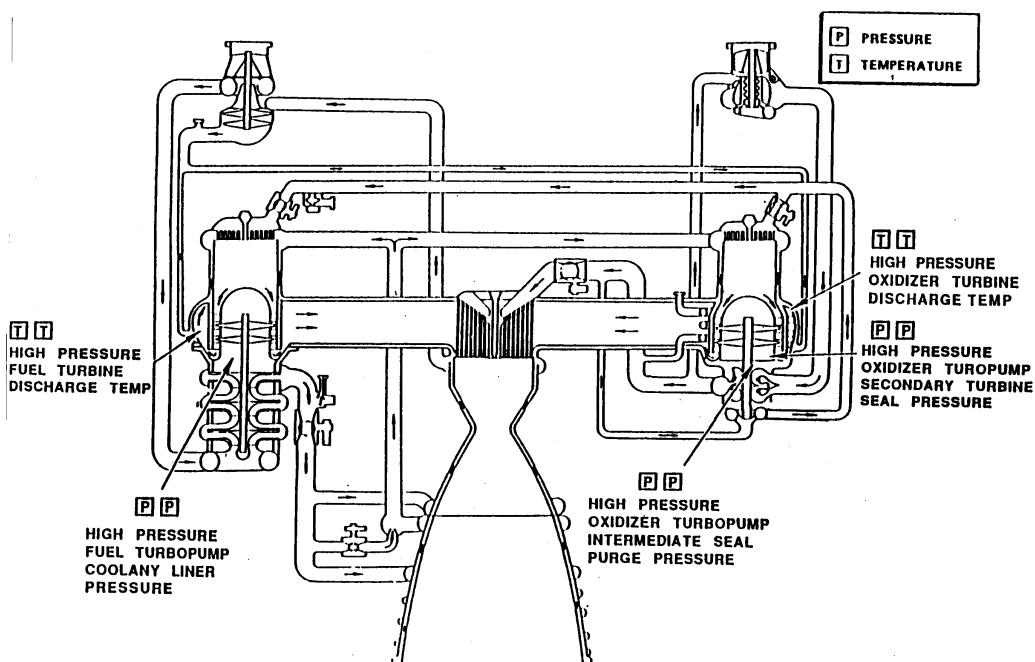


Figure 53. SSME redlines.

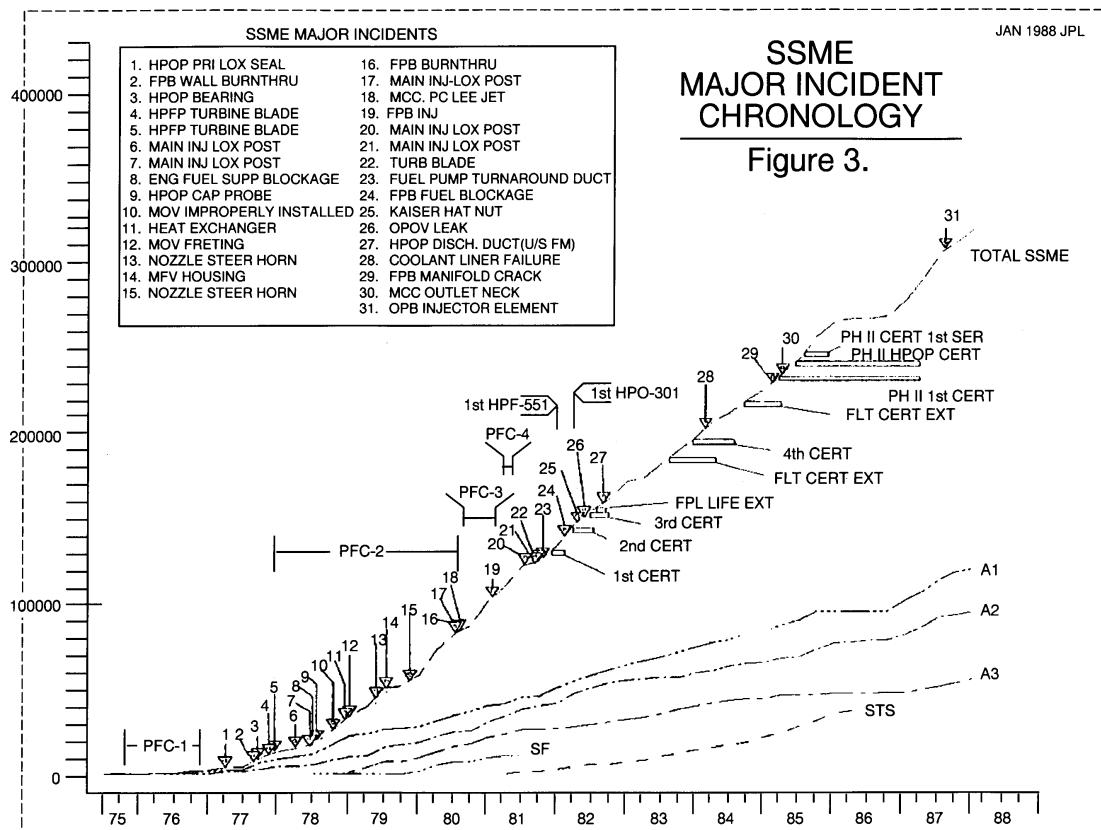


Figure 54. SSME major failures.

FIGURE 4. MAJOR INCIDENT ORIGINS

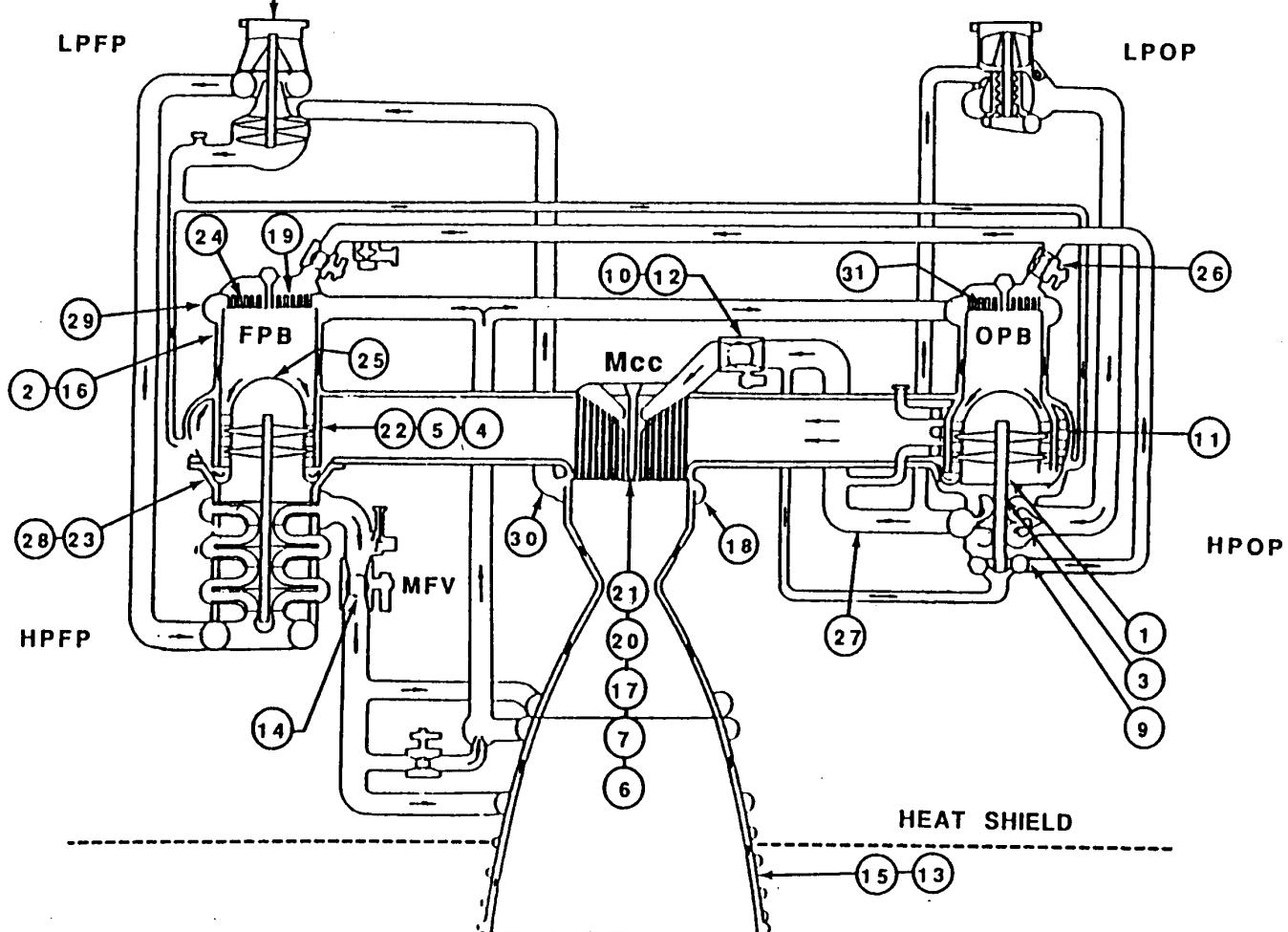


Figure 55. SSME schematic with failures.

(1) SSME 4,000 Hz. Flow instabilities have been the classical problem of structural systems. They have taken many forms, occurring unexpectedly, taking human life. Many textbooks and technical papers focus on the phenomenon; aircraft flutter garners the bigger emphasis. Internal flow in fluid systems experiences the same instabilities. An instability is generally defined as an increasing amplitude structural response where the fluid force is a function of the structural deflection causing the problem.

The SSME lox system fits the characteristics required to produce instabilities, high density-large flow velocities, thus, a very high dynamic pressure. Many problems have occurred in this system. The inlet tee to the lox dome of the main combustion chamber (fig. 56) is a classic example of what can happen. The tee has a two-blade splitter (flow between, and on each side of, the two vanes) designed to more evenly distribute the lox flow to the powerhead. On most engines (90 percent) nothing happened. However, on 10 percent of the engines, a high-frequency, high-amplitude (4,000 Hz, 100 g's) oscillation occurred. If run long enough, the vanes crack, detuning the structure from the flow, eliminating or greatly reducing the response. Two types of structural elastic flow interaction are possible: (1) vortex shedding, and (2) flutter-type instability. Extensive effort was expended to understand the problem through a team effort by MSFC and Rocketdyne for approximately 3 years. These efforts included (1) structural dynamic and static modeling, (2) scale-model and full-scale flow testing, (3) computational

Table 4. SSME major incident overview.

Incident No.	Date	Test	Engine	Failure	Cutoff Method	Damage
31	7/1/87	902-428	2108	OPB injector element braze crack	HPFT redline lower limit	Cont.
30	3/27/85	750-259	2308	MCC outlet manifold weld crack	HPFP accel	Uncont.
29	2/4/85	901-468	0207	FPB manifold crack	Power amb. temp.	Uncont.
16	7/12/80	SF10.001	0006	FPB liner and housing burn through	Observer	Uncont.
2	8/27/77	901-133	004	FPB housing burn through	Observer	Uncont.
28	2/14/84	901-436	0108	HPFP coolant liner failure	HPFP turbine disch. temp.	Uncont.
23	10/15/81	901-340	0107	HPFP turnaround duct failure		Cont.
27	8/27/82	750-175	2208	NPOP discharge duct failure (ultrasonic FM)	HPOP PBP radial accel.	Uncont.
26	5/15/82	750-168	0107	OPOV leakage during cutoff	Program duration	Cont.
25	4/7/82	901-364	2013	HPFP kaiser hat nut failure	HPOP PBP radial accel.	Uncont.
24	2/12/82	750-160	0110F	Fuel injector blockage (ice)	PFP turbine disc. temp.	Uncont.
8	6/10/78	902-112	0101	Eng. inlet fuel supply blockage (N_2)	HPFP SPED	Uncont.
22	9/21/81	902-249	0204	HPFP turbine blade failure	HPFP radial accel.	Uncont.
5	12/1/77	901-147	0103	HPFP turbine blade failure	HPOP radial accel.	Uncont.
4	11/17/77	9020095	0002	HPFP turbine blade failure	HPOP radial accel.	Uncont.
21	9/2/81	750-148	0110	MN injector lox post failure	HPOP turbine discharge temp.	Cont.
20	7/15/81	901-331	2108	MN injector lox post failure	HPOP turbine discharge temp.	Cont.
17	7/23/80	902-198	2004	MN injector lox post failure	HPOP turbine discharge temp.	Cont.
7	6/5/78	901-183	0005	MN injector lox post failure	HPOP turbine radial accel.	Cont.
6	3/31/78	901-173	0002	MN injector lox post failure	HPOP turbine	Cont.
19	1/28/81	901-307	0009	FPB injector lox post failure	Prog. C/O	Cont.
18	7/30/80	901-284	0010	MCC PC Lee jet failure	HPOP PBP rad. accel.	Uncont.
15	11/4/79	SF6-003	2002	Nozzle steerhorn failure during C/O	Cut by eng. 0008 HPOP turbine sec. seal pr.	Uncont.
13	5/14/79	750-041	0201	Nozzle steerhorn failure	HPFP turbine disch. temp.	Uncont.
14	7/2/79	SF6-001	2002	MFV housing crack	HPFP turbine temp.	Uncont.
12	12/27/78	901-225	2001	MOV fire due to fretting	HPFP turbine disch. temp.	Uncont.
11	12/6/78	901-222	0007	Hex coil failure	Hex disch. temp.	Uncont.
9	7/18/78	902-120	0101	HPOP capacitance probe rub	HPOP PBP axial accel.	Uncont.
1	3/24/77	901-110	0003	HPOP prim. seal failure	HPOP speed	Uncont.
3	9/8/77	901-136	0004	HPOP No. 3 bearing	Facility	Uncont.
10	10/3/78	902-132	0006	MOV improperly installed	MCC PC lower limit	Uncont.

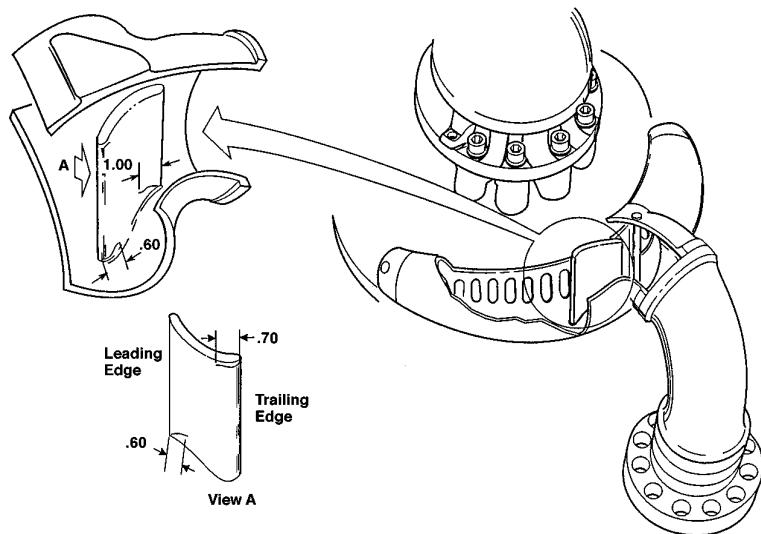


Figure 56. SSME lox dome inlet tee.

fluid flow analysis, (4) fatigue and fracture mechanics analysis, and (5) structural dynamic and static testing. These efforts led to the conclusion that the basic instability was essentially vortex shedding. Figure 57 shows a typical response of buzzing and nonbuzzing engines. The first is a spectrum showing the very sharp narrow band resonance peak indicating an instability. Conducting a correlation analysis shows that the response is single-frequency sinusoidal. The fix consisted of scalloping the leading edges to allow more flow between the blades and tapering the trailing edge to eliminate vortex sheets. In addition, the frequency of the vane was raised slightly, helping to detune the system. Figure 58 shows the fix. Figure 57 is a plot of the response before and after the fix.

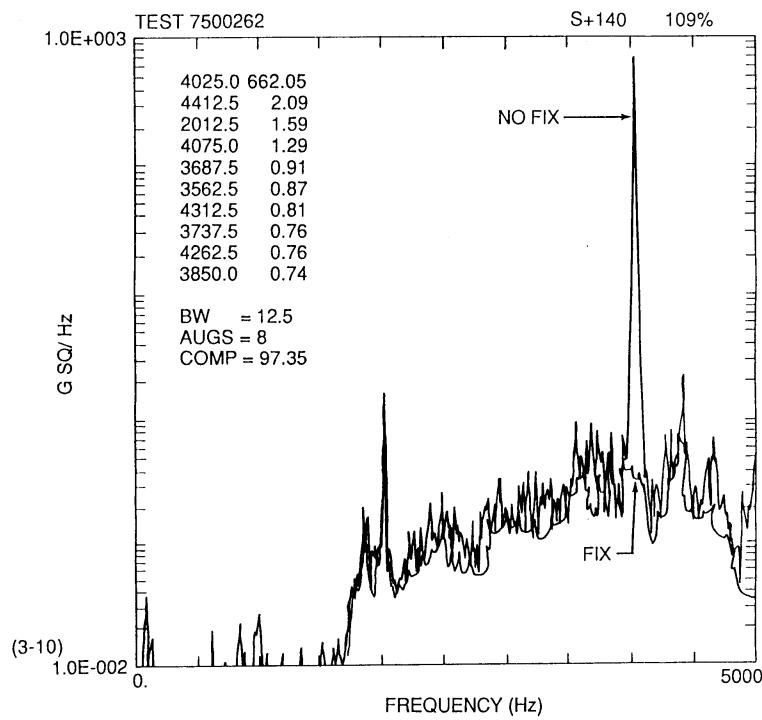
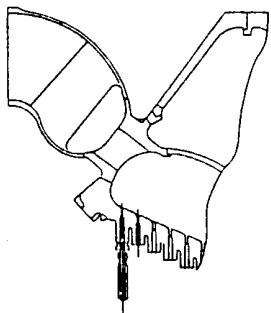
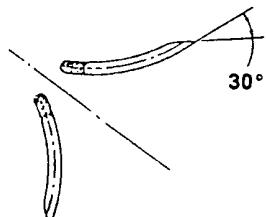


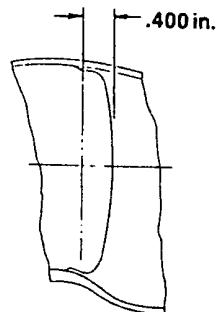
Figure 57. SSME response of 4,000 Hz.



Baseline Configuration



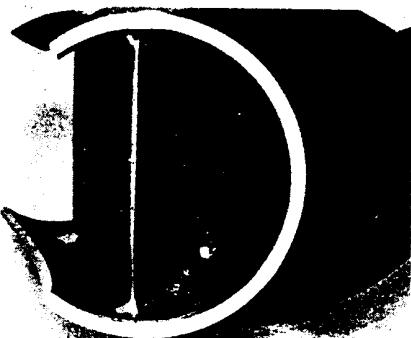
Trailing Edge



Leading Edge



Baseline Configuration



Trailing Edge



Leading Edge

Modified Configuration

Figure 58. SSME splitter fix.

One very interesting problem occurred during the investigation. It was thought that by putting a structural collar on the powerhead dome, the frequency shift would detune the system and, in addition, provide damping, further reducing the response. The collar was hot-fired on a buzzing engine resulting not in a reduced response, but a greatly increased oscillation. The dynamic detuning, in reality, became more finely tuned, creating the increased response.

These lessons stand out: (1) do not allow dynamic tuning in structural flow systems, (2) design out instabilities, and (3) realize that dynamically tuned systems are very sensitive to very small changes. In retrospect, it was the very small differences in the trailing edge of the vane, coupled with some thinning in the duct, that created the buzzer versus the nonbuzzer modal tuning. These differences were so slight that they were essentially undetectable through normal inspections.

(2) ATD Lox Pump Bearing Wear and Vibration Problem. The ATD lox pump is a replacement of the currently used Rocketdyne turbopump with a goal of greatly reduced maintenance and operational cost through redesign using advanced technologies and greatly reducing the number of structural welds. During early developmental testing on the E-8 hot-fire pump test stand, and later full-engine ground testing, vibration problems occurred leading to test cutoff or engine shutdown. These vibrations occurred as a sharp rise in synchronous vibration triggering the health monitoring shutdown systems. Figure 59 shows a typical high-vibration response in g's of acceleration. With the solution of the high-g

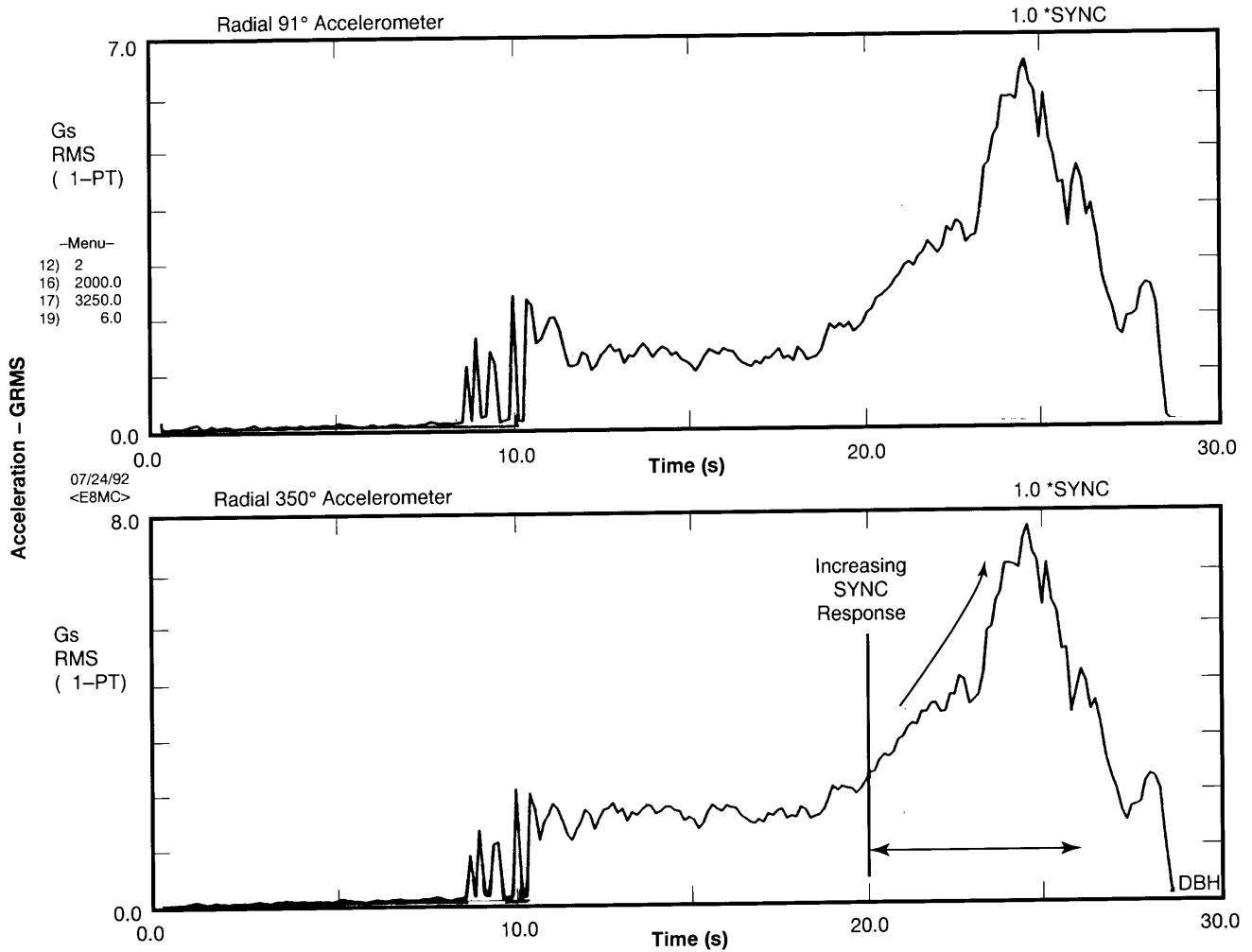
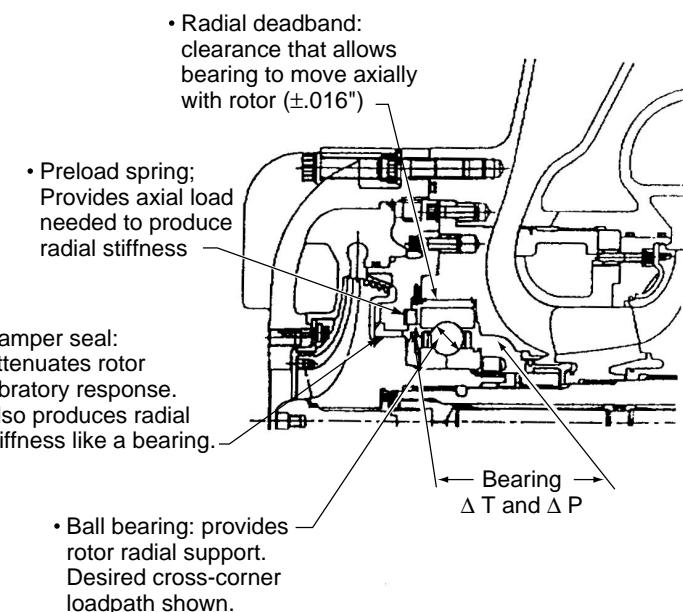
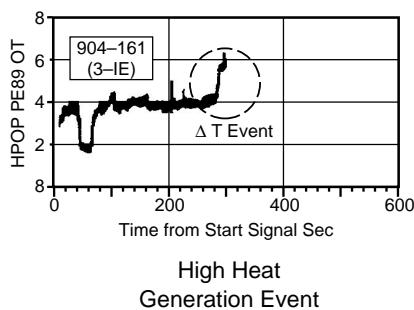
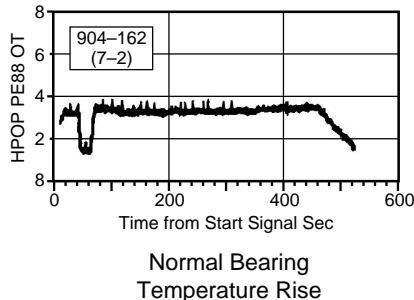


Figure 59. SSME high-vibration ATD lox pump.

vibration problem, long-duration hot-fire engine testing could commence. As this testing began, excessive ball bearing wear occurred, exemplified by a rise in the bearing cavity temperature (fig. 60). With the solution of the bearing wear issue, long-term developmental testing resumed. Initially, some changes made in attempts to solve the bearing wear resulted in a reintroduction of the high vibration for one pump. The problem was solved by going back to the original solution for high vibration (tolerances, etc.). Additional testing uncovered some undesirable vibration sensitivities (not of the high-q engine cutoff type) that required additional solutions. The next three sections will give a brief summary of the high vibration, bearing wear, and vibration sensitivities issues that occurred during early ATD lox pump development. They illustrate the complexity of high-performance machinery development and the system issues that usually occur. The approach used to solve all these problems utilized a Government/contractor concurrent engineering team. The team used the standard fault tree approach with special subteams designed to investigate special legs of the fault tree. Formal action items and logic diagrams were used to keep the activities focused. About one-third of the total team's time was spent at Pratt & Whitney. The rest of the effort was worked at the home offices, with a twice-a-week telecon to track actions and report programs. Concurrent engineering teams using fault trees are an excellent approach for handling these types of complex problems. See reference 65 for details.



Stennis History of PEBB Delta-T Spikes

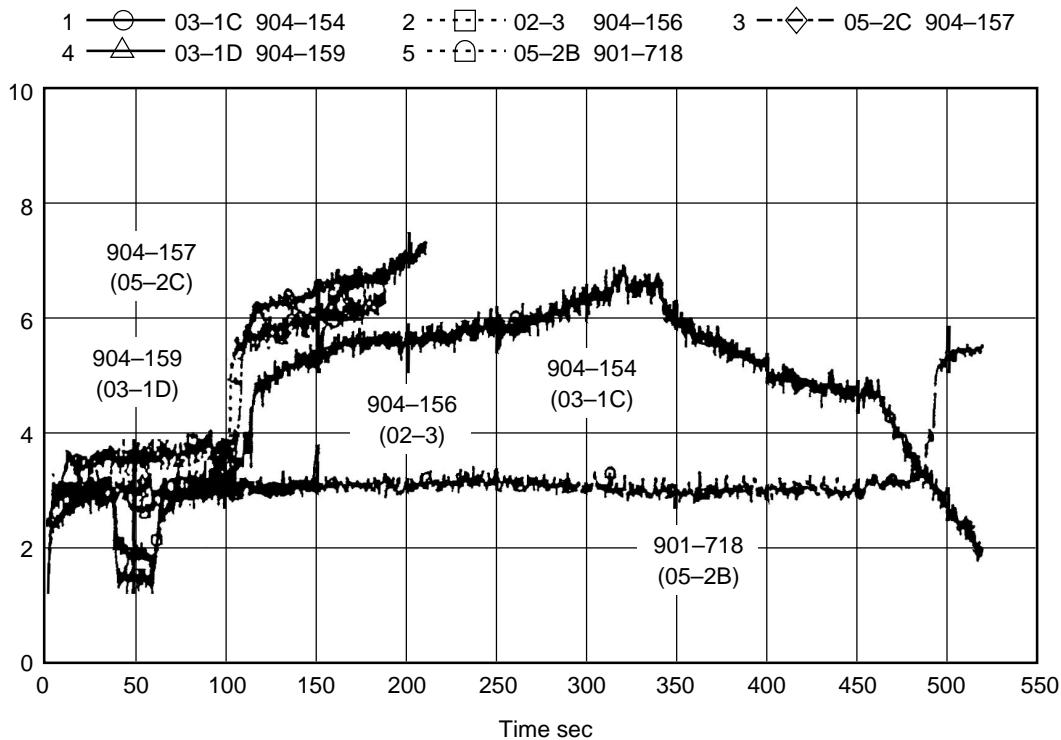


Figure 60. SSME ATD lox pump temperature excursions.

(3) ATD High-G Vibration. Initially, the source of the vibration problem was unclear. The pump end of the lox pump had a damper seal in parallel (adjacent) to the ball bearing support (fig. 61). The damper was installed to prevent whirl; however, in achieving the damping, a stiffness level of approximately 400k lb/in was produced. This kept the ball bearing from contacting the housing until forces developed that would overcome the damping seal stiffness. This split in rotor support stiffness created a split force between the damper seal and the ball bearing. Several items were proposed: (1) loss of damping in the damper seal, (2) loss of stiffness in the damper seal, (3) nonparallel housing producing bearing tilt, and (4) bearing clearance.

ATD-HIGH PRESSURE OXIDIZER TURBOPUMP

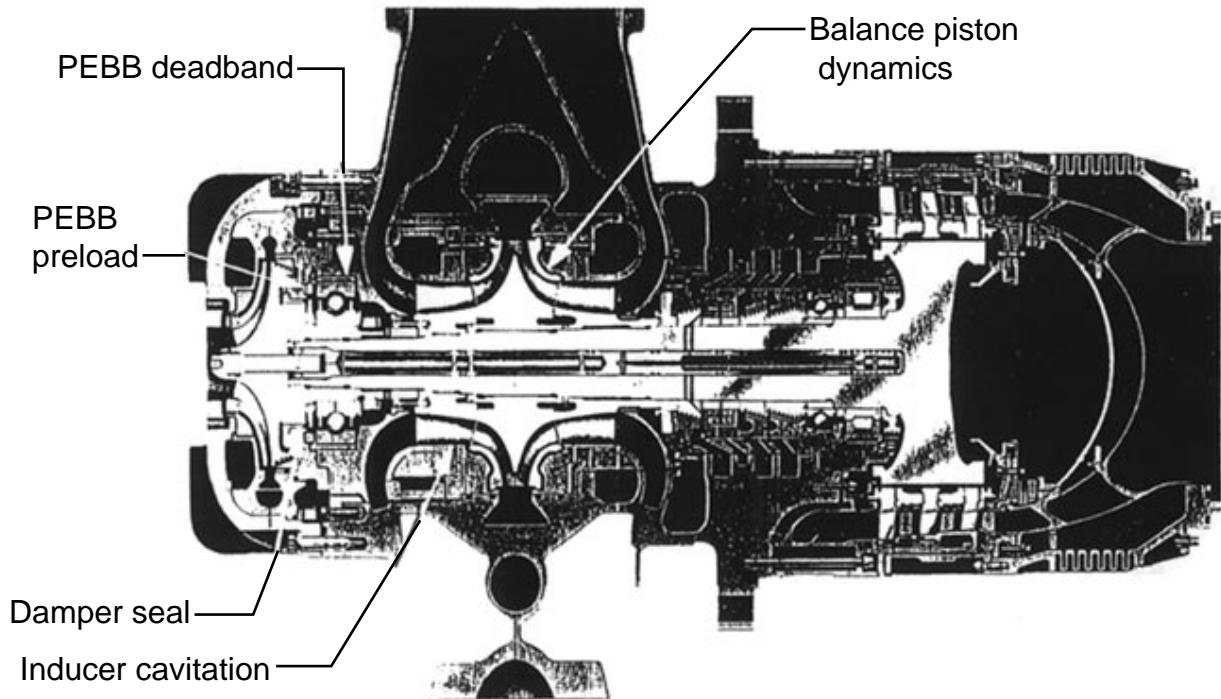


Figure 61. SSME ATD lox pump damper seal/bearing.

As work progressed, it was decided that the bearing sleeve had to be tapered such that the bearing did not tilt negatively. Figure 62 shows how the stiffness of the bearing is greatly reduced as a function of negative bearing tilt. Also, it became clear that the bearing clearance was too tight, possibly creating bearing axial hang-up. A pump was modified to ensure a flat bearing-to-housing contact and wider clearance to prevent hang-up. This pump ran vibration-free until it was pushed slightly outside its operational parameters by varying the net suction specific (NSS) pressure (increasing hydrodynamic forces). When pushed hard, this pump also experienced high synchronous vibration.

While this activity was progressing, a parallel activity was being pursued in the MSFC flow facility to see if a reduction could be made in the impeller hydro forces through inducer changes. The results showed that a change in clearance had a major effect on the forces, reducing them by up to an order of magnitude. In addition this change was incorporated in a pump that also had the tilt and clearance change. This pump configuration had margin for the high-g vibration. Figure 63 is a plot of

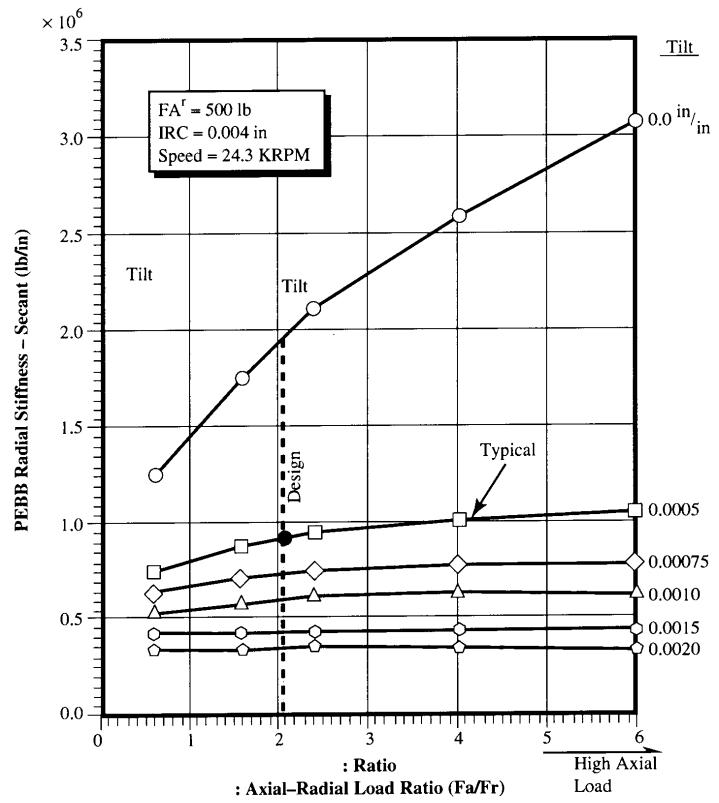
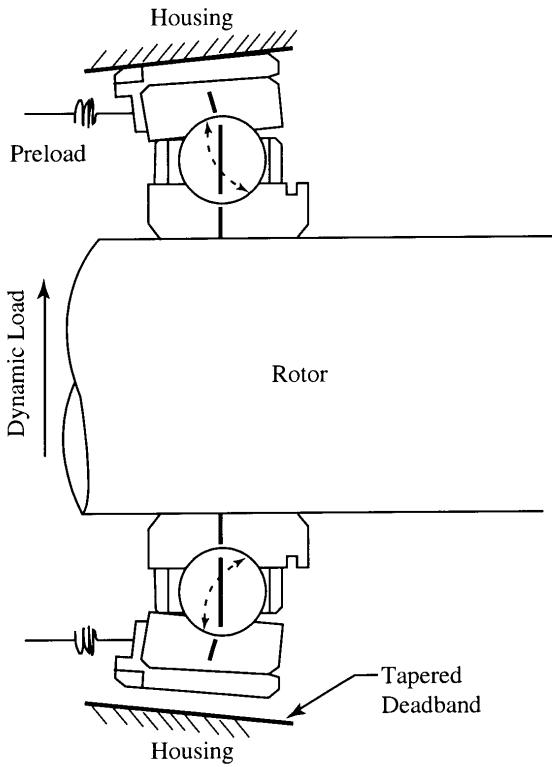


Figure 62. SSME ATD lox pump bearing stiffness versus tilt.

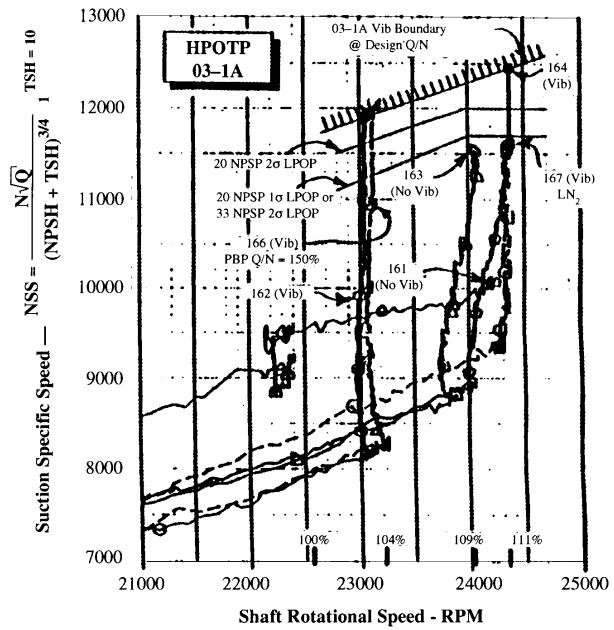
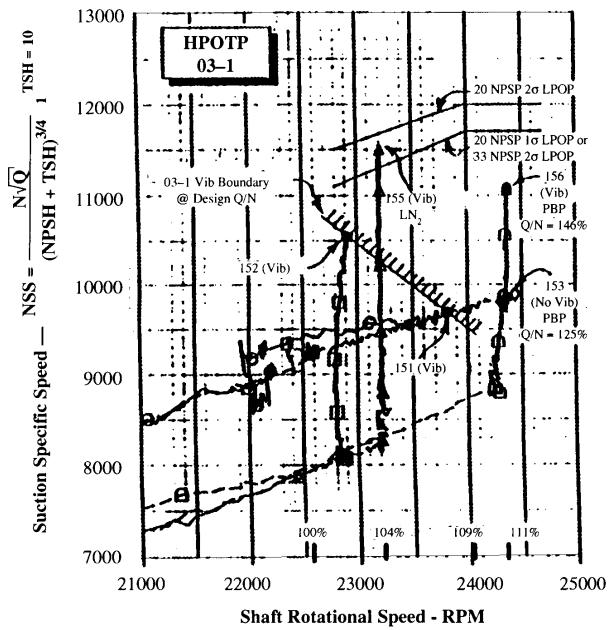


Figure 63. SSME ATD lox pump build accelerations versus NSS.

several different pump configurations showing acceleration versus NSS. Notice that the first two fixes helped the vibration characteristics of the pump; however, the major fix included was the inducer change, which reduced the hydro forcing function substantially. Later, a pump housing redesign was made that eliminated the requirement of tapering each bearing sleeve (called stiff housing).

As mentioned previously, after the bearing wear problem was solved one other high-g pump occurred. Figure 63 shows this response. This high-g response was caused by reducing the bearing clearance as a potential bearing wear fix. Increasing the clearance back up to the previous value fixed the problem.

(4) Bearing Wear. With the solution of the high-g vibration problem, bearing distress became an ATD program-threatening issue. Some bearings were lasting several test with small wear, while others were wearing within 50 or 100 s. This was exemplified by a rise in bearing cavity temperature. Figure 64 is a plot of the bearing cavity discharge temperature showing the temperature increase when the bearing starts to wear. Figures 65 and 66 are pictures of a bearing race showing the wear tracks. The balls become so hot that they turn dark. After many attempts to isolate the wear cause, a pump was built with silicon nitride balls while keeping the steel races. This pump showed no bearing wear. As a margin enhancement, special inner-race cooling and outer-race-guided cages were added. No bearing wear has occurred since making these changes. The story of interest is how the solution did evolve. This problem dramatically illustrates the many interacting factors/parameters in a complex nonlinear dynamic problem. Bearing sets have accumulated 20,000 s of hot firings without any apparent balls or race wear.

SSME / ATD HPOTP VIBRATION REVIEW

Test Experience

CORRELATION		
Param	Hi Vib	Vib Sens.
DS	NO	NO
DB	YES	NO

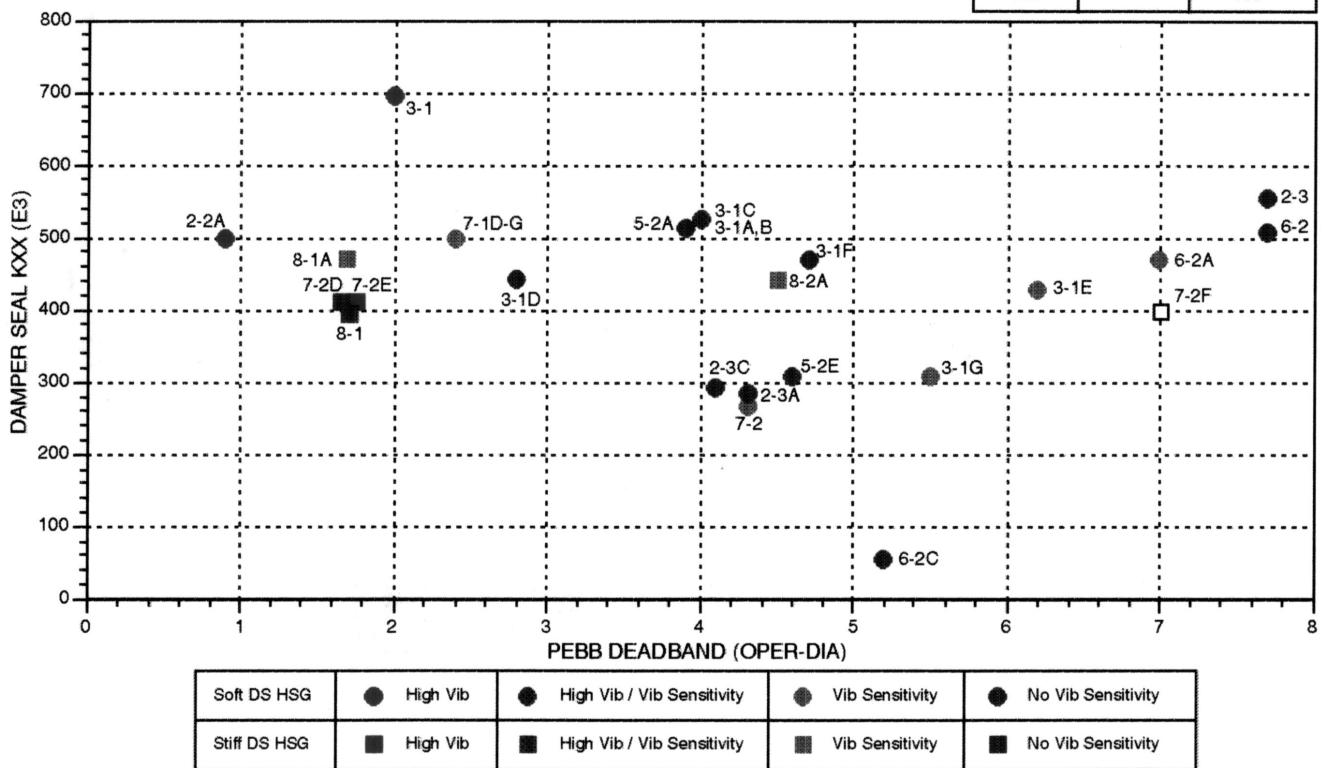


Figure 64. SSME ATD lox response versus bearing clearance.

Unit 3-1C Bearing Post Test Condition Summary

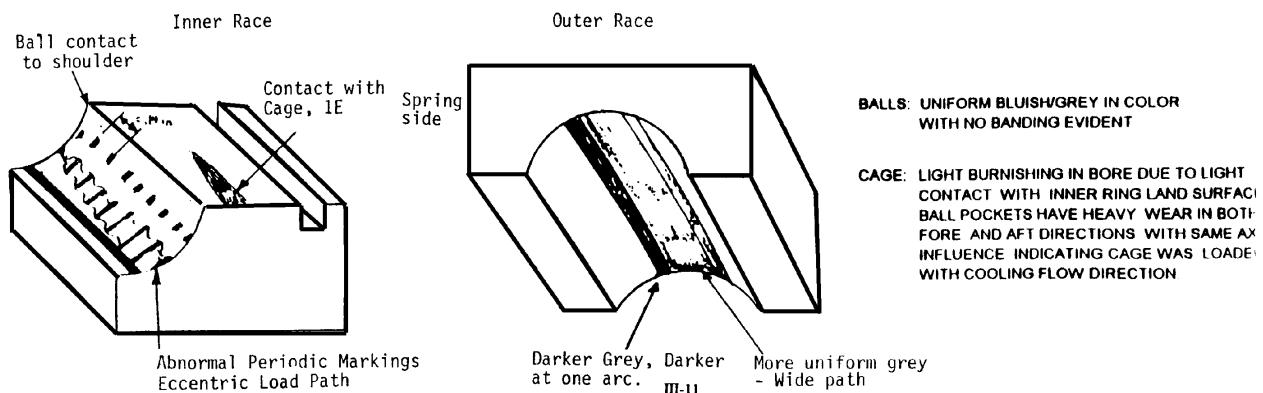
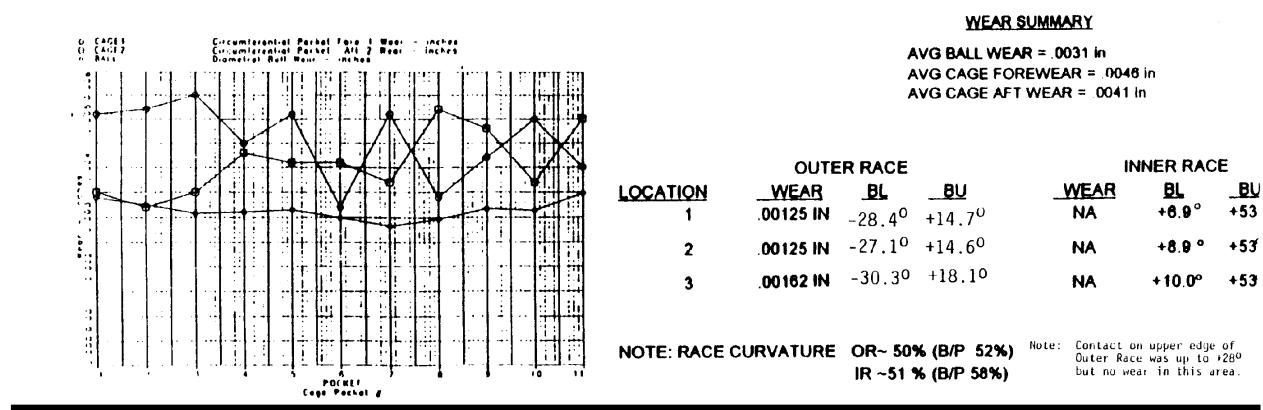


Figure 65. SSME ATD lox pump bearing wear.

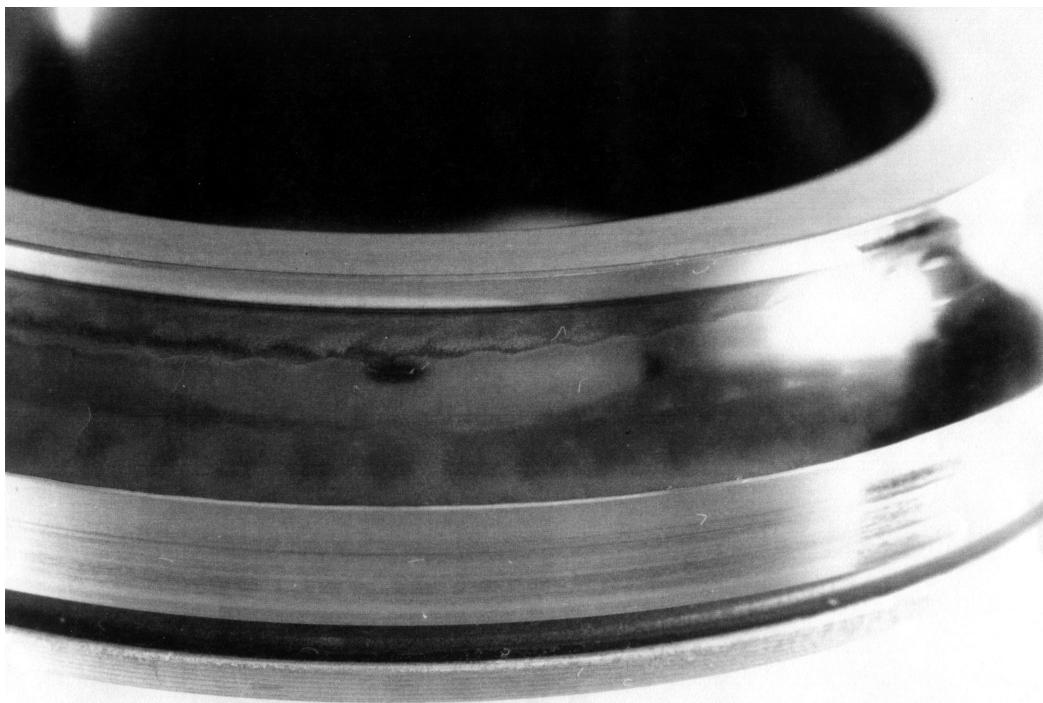


Figure 66. SSME ATD lox pump bearing wear.

There are two generic sources of bearing wear: (1) heat generation and (2) heat dissipation. There is always heat generation. The concern is excessive heat generation. Not enough heat dissipation will allow even a nominal heat generation bearing to overheat. Thus, the fault tree, or failure tree, had to list all the potential causes under each from lubrication, materials, to coolant. To illustrate how fault tree analysis can lead down blind paths until the data are collected and understood, the pump that ran the longest period of time had a different prelubrication treatment and still had lubrication present when the pump was disassembled. As a result, lubrication experts were called in and many tests were conducted on lubricant application before it was ruled out as a cause. A second blind alley occurred because the bearing rig test had demonstrated long life. These tests did not have the damper seal and had parallel bearing support with good bearing loads. It was thought that the damper seal caused the bearing to unload or that the bearing was tilting due to lack of load. The damper seal acted as a rotor support. Two pumps were built, one with no damper seal by the bearing and the other with a small clearance and an eccentric load. Both bearings wore quickly. By then, time was running out so the advanced-technology silicon nitride balls were installed in a pump. This pump ran many tests without ball wear. The team was simultaneously working on heat dissipation and, therefore, also added jet cooling (lox) to the bearings at the contact point and outer-race-guided cages. Both designs theoretically increased the bearing wear margins.

Silicon nitride balls alleviated the heat generation and heat dissipation aspects of the problem. From the heat generation standpoint, the balls are 70 percent lighter, hence lower centrifugal forces occur. These balls have a lower coefficient of friction, therefore, heat dissipation is better. Theoretically, to date, the total improvement of silicon nitride balls cannot be explained; however, approximately 10 pumps have been assembled and hot-fire tested without ball wear. Two pumps have over 20,000 s each without bearing balls or race wear.^{65 66}

(5) ATD Vibration Sensitivity. When the problem of bearing and high vibration of the lox turbopump was solved, extensive full-duration and margin testing of the turbopump on the shuttle main engine were accomplished. During this testing, vibration level shifts occurred that were a concern; however, these levels have not led to any early engine cuts or caused any known hardware damage. There is more than one category of these responses; however, the main concern has been the category II (vibration step increases by a factor of two or more) and category III (vibration step decreases of the same magnitude).

Figure 67 shows a typical category II type response. Notice the step in g level from less than 1 g to greater than 2 g's. These steps are not present on every test. Figure 68 is the end-to-end plots of several tests showing the variability of the phenomenon. There have been two major thrusts as potential causes. As discussed previously, the pump-end damper seal supports the shaft for a certain range of forces before the pump-end ball bearing stiffness becomes effective. Until then, the pump mode (one of the rotor criticals) is near the pump operating speed. This would make the pump sensitive to force perturbations until the ball bearing clearance is reduced to zero. The second thrust has to do with the roller bearing and its support structure on the turbine end. Hot-fire data have clearly shown that the hot gas is causing the structure to snap. Figure 68 shows the temperature rise in the cavity, indicating a structural shift. Also, impedance of the roller bearing hardware indicates that the roller bearing is slipping. If the stiffness of the support is greatly reduced, the turbine mode will reduce in frequency and tune with the pump speed, resulting in high g's of vibration. A simulation was developed that showed this potential. How the loss of stiffness occurs is still an unanswered question. With these changes, the vibration sensitivity has been eliminated. The lox pump now has over 100,000 s of ground testing and two shuttle flights without any bearing wear or vibration.

(6) 2308 Failure. A different kind of structural flow interaction occurred on engine 2308. In this case, the hydrogen inlet tee to the main combustion chamber, which provides liquid hydrogen for nozzle cooling, developed a crack resulting in loss of the engine. A movie camera trained on the engine during static firing clearly showed the whole sequence. This was not a flow instability, but a classical forced oscillation resulting in fatigue failure after 20,000 s of hot fire time in a total of approximately 100 tests. Figure 69 shows the tee splitter arrangement and the failure location. The failure contour plot shows the

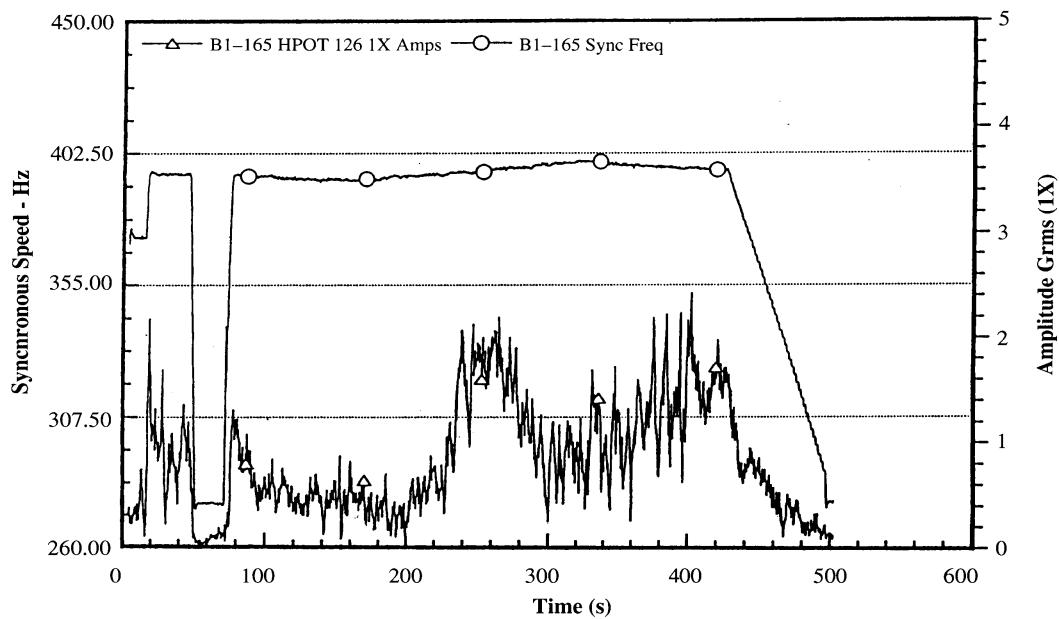


Figure 67. SSME ATD lox pump category II response.

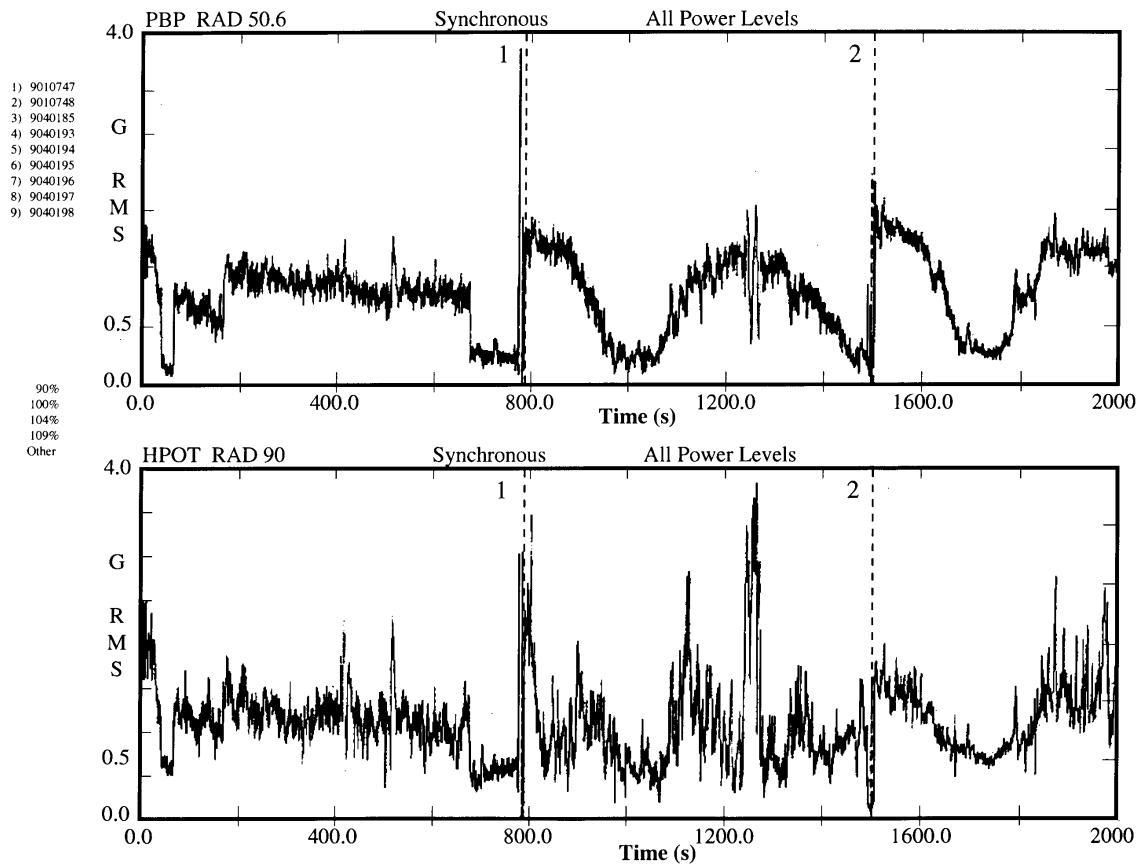


Figure 68. SSME ATD lox pump end-to-end accelerations.

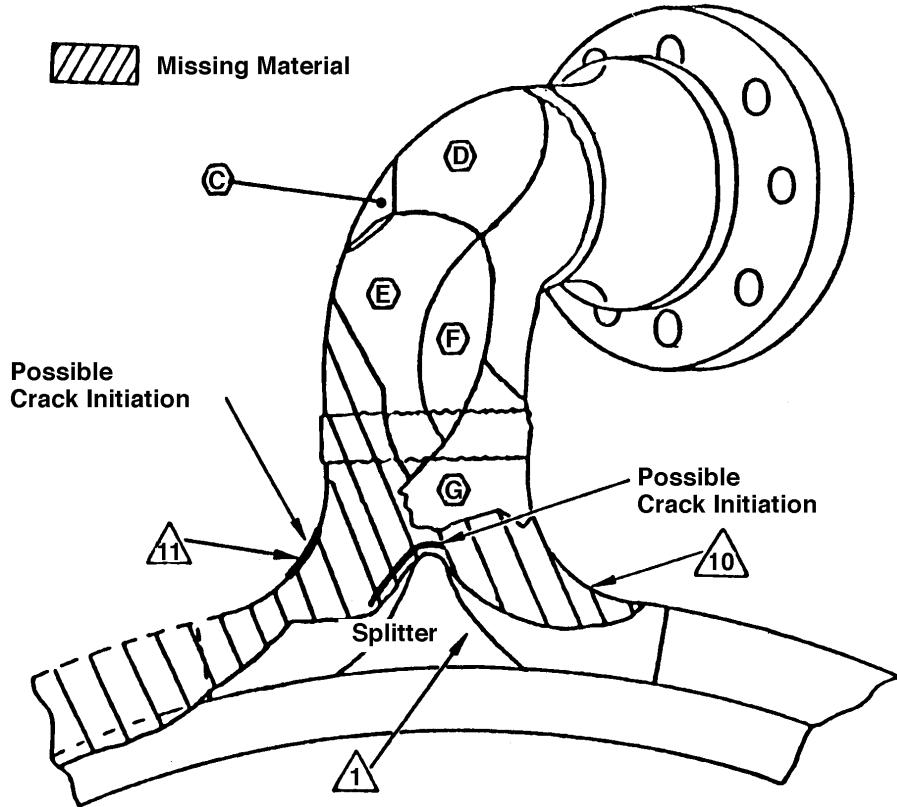


Figure 69. SSME hydrogen inlet tee 2308.

sharpness of the stress concentration. Stress risers occur from weld beads, sharp corners, etc., and can lead to problems. This failure focused attention on fatigue failures and stress risers, resulting in a structural audit of the total engine system, the next topic. In order to totally understand the 2308 failure, several activities were accomplished: (1) 3-D structural static and dynamic models (original analysis was 2-D), (2) static and dynamic testing, (3) strain gauging of the splitter in hot fire static test, (4) materials testing, (5) improved flow analysis, and (6) special flow testing, water table, and special full-scale hardware. As was evidenced in this problem, 2-D analysis is not sufficient to account for the stress distributions and concentrations. Although expensive to conduct, 3-D analysis of these kinds of geometry is required, as well as detailed static and dynamic testing and hot fire test stress instrumentation.

(7) SSME Structural Audit. The 2308 failure showed that many areas such as splitter duct tee interfaces were susceptible to fatigue. Most of these hardware areas had inadequate analysis/test predictions. The failure investigation team recommended that a structural audit be conducted with a focus on long-term durability through identifying generic structural weaknesses (analysis, test, etc.) and to institute the corrective actions necessary to enhance the confidence in flight reliability by accurate life predictions. This effort was started prior to the STS-51L accident. After the accident, greater emphasis was placed on its completion to satisfy the total recertification of the shuttle, including a complete reanalysis of FEMA/critical items lists (CIL) and hazards.

The audit process was carried out by a full-time independent team of Rocketdyne engineers with oversight review by an MSFC team. Initially, the two teams separated the engine parts into categories of risks according to parameters affecting hardware life. Essentially, three broad groupings were obtained: (1) parts that were criticality 1 on FEMA/CIL with potential short-term life issues, (2) parts that had long-term life issues, and (3) parts with no concerns. Obviously, category (3) parts were eliminated from the audit process. Category (1) had high priority, with category (2) to follow after completion of

category (1). The process was clear for category (1) and (2) parts. The total analysis/test reports, etc., would be reviewed indepth to determine the adequacy of hardware lifetime predictions, etc. This included a review of the materials properties, environments definition (flow, thermal, acoustics, vibration, etc.) dynamic model, dynamic analysis, stress models, and stress analysis. It was recognized early by the SSME project that a large number of welds on the engine was a prime concern in the area of flow-cycle fatigue, high-cycle fatigue, and inspectability and, therefore, lifetime description. This area, weld assessment, was assigned to a special team and was not included in the audit (it will be discussed in the next section).

As an aid to the audit engineers, check-off sheets were developed for each discipline review. A copy of these check-off sheets was available at both Rocketdyne and MSFC. Formal action items and action closures were installed. Deviation approval reports (DAR) were written for all parts that were found deficient. Several areas were found with inadequate (1) materials properties data, (2) environment definition, (3) dynamic analysis, and (4) stress analysis. This resulted in material testing to obtain properties, environmental testing, special hot-fire measurements, new 3-D dynamic and static analysis, redesign efforts of critical parts, and DAR's limiting use of certain parts were issued (seven ducts).

Are structural or other audits necessary to obtain good hardware? In most cases, they probably are; however, the process could be greatly reduced, if not eliminated, with proper in-line reviews. The check sheets could be used by all disciplines. System emphasis is important. Understanding key parameters and their sensitivities though analysis and test is mandatory.

A Rocketdyne document was published that describes the audit process in detail, the parts audited, action items assigned, and general recommendations.⁶⁴ The audit was very useful. It was used as a refresher for FEMA/CIL activities. Updated procedures and criteria manuals have resulted. The individual audits are excellent reference material for materials reviews (MR's), etc., occurring during manufacturing because drawings, analysis, etc., are all collated together. The materials characteristics update and redesign to eliminate problem areas are still in work. The results of the structural audit and the return to flight activities following STS-51L led to a reassessment of the FMEA/CIL and the need to assess the structural welds of the SSME.

(8) Weld Assessment. The weld assessment was accomplished by a Rocketdyne team with reviews by a MFSC team. The areas of concern were:

1. Weld offset
2. Weld notch factors
3. Materials properties degradation
4. Root-side inspectability
5. High-cycle fatigue critical flaw size inspectability
6. Manufacturing
7. Failures and lessons.

Because many of these welds were identified as criticality 1 failures, it was necessary that the risk associated with each weld be understood. This meant that each weld (criticality 1) had to be analyzed and understood the areas of:

1. Process control
2. Materials usage
3. Inspectability
4. Fracture mechanics
 - a. Low cycle
 - b. High cycle
5. Environments
 - a. Thermal
 - b. Corrosion
 - c. Loads.

The SSME design evolved from baseline design requirements to minimize weight and leak paths while producing the Isp and thrust required for performance. This baselined design philosophy was:

1. Lightweight plus minimum leak paths
2. Wrought, welded structure versus castings
3. Manufacturing technology plus cost
4. Subassembly welds versus numerical machined.

The baselined design had no “fracture control” requirements. A fracture mechanics requirement plan was established in 1973 after the basic design was complete and released. In this plan, 71 parts (150 welds) were identified as fracture critical. Instead of redesigning parts, the plan was implemented by NDE improvements. In addition, prior to STS-51L, welds were selectively reassessed, including (1) weld wire control and verification, (2) electron beam weld design, (3) vendor welding and inspection processing, (4) residual stress in as-welded welds, (5) geometric variables—mismatch and local stress risers, and (6) material properties upgrades. The CIL was expanded after STS-51L to identify all critical welds. A weld assessment program was established as a systematic, comprehensive evaluation of all critical welds to support the CIL and evaluation of future changes.

Teams were developed at Rocketdyne to carry out the activity with a review team designated at MSFC. The scope of this activity was major. The program objectives were: prepare rationale for retention for all welds prior to flight resumption based on reassessment of structural adequacy, fabrication process, and supporting NDE; identify and implement corrective actions as required; and establish a weld data base of selected design, analysis, fabrication, NDE parameters, and operation conditions as a rapid screening tool for future weld issues. It is not possible to convey the real magnitude of the effort. All the risk assessments were completed and results documented. The approach used is shown in figure 70.

The total number of welds by weld type, root side access, and high and low fatigue NDE for each subsystem are shown in figure 71. The results of the study, which was the most comprehensive and detailed assessment in the history of any program, showed that all welds were acceptable for flight resumption. The assessment provided relative ranking of most critical welds so that remedial efforts could be focused at the highest payoff.

Weld Assessment Program Objective

- Prepare rationale for retention for all welds prior to flight resumption based on reassessment of structural adequacy, fabrication process, and supporting NDI.
- Identify and implement corrective actions as required.
- Establish a weld database of selected design, analysis, fabrication, and NDI parameters and operating conditions for all welds as a rapid screening tool for future weld issues.
- Define FMEA/CIL welds—part name, part number, weld number, weld type, class, material, filler, and heat treat.
- Review and summarize structural analysis—operating conditions, factors of safety, proof factors, and life.
- Review and summarize fabrication process—automatic, manual, single or multipass, and other special conditions.
- Review and summarize NDI processes, related detectable flaw size and requirements flowdown—visual, penetrant, radiographic, and ultrasonic. Specifically identify welds without root side penetrant inspection.
- Calculate critical initial flaw size for all FMEA/CIL critical welds—compare to NDI detectable flaw size.
- Summarize operational test history—units, starts, and seconds.
- Photographs of sectioned hardware (if available).
- Summarize other supporting data—lab tests, hot-fire data, and UCR or MR history.
- Identify and implement corrective action—NDI, and planning or design changes.
- Prepare rationale for retention.
- Review and approve in-house and with NASA team.
- Documented results in the SSME weld assessment report—RSS-8756 CIL summary includes weld type, class, and root side accessibility for penetrant inspection and detectability of critical flaw size

Figure 70. SSME engine welds and NDI.

The high- and low-cycle fatigue welds with unverifiable roots were cleared for flight by one or more of the rationales shown in figure 72.

Figure 73 is an example of one weld to show the complexity of the problem. Notice that critical flaw sizes for low- and high-cycle fatigue are shown, along with the inspection technique and detectable flaw size. A drawing of the part is included. Documented in conjunction with these summary sheets is a several page write-up of process control, inspections, fracture mechanics, etc., and the rationale/risk assessment that allows the hardware to be used if the critical flaw size is not inspectable. The total documentation is approximately 20,000 pages.

The audit has resulted in some redesigned parts having better environmental definition, improved inspections, and improved processing. Clearly, fracture control should be implemented in future high performance systems and indeed was implemented on the ATD program. Welds are a potential problem requiring design (1) inspectability, (2) reduced stress fields (thicker weld interfaces), (3) elimination where possible, and (4) robotic processing, etc.

FMEA/CIL Weld Statistic

LRU	Total (1)	GTAW	EBW	Other	Root Side Access		HCF CIFS ≤ NDE	LCF CIFS ≤ NDE
					Yes	No		
Comb devices	1,288	588	93	607 ⁽²⁾	232	1,056	840	732
Turbomachinery	801	662	117	22	313	488	187	43
Pneu controls	1	0	1	0	0	1	0	0
Prop valves	65	24	38	3	8	57	26	4
Actuators	5	0	5	0	0	5	0	0
Igniter, sensors	186	25	149	12	0	186	68	51
Lines, ducts	701	611	89	1	210	491	310	64
POGO system	115	115	0	0	1	114	3	1
Joints	3	0	0	3	0	0	0	0
Total	3,165	2,025	492	648	767	2,398	1,434	895
Percentage		64%	15%	21%	24%	76%	45%	28%
Percentage without LOX posts		79%	19%	2%	30%	70%	33%	12%

Notes:

1. Includes castings, optional welds and allowable alternate configurations
2. Main injector has 600 inertia welded LOX posts

Figure 71. SSME fatigue rationale.

Rationale For Retention	
• Welds where the largest undetected flaw size > CIFS were cleared for flight by the integrated weld assessment which included	<ul style="list-style-type: none"> – The probability of detection, if less than 90% reliability and 95% confidence – Size, Type, and location of critical flaw <ul style="list-style-type: none"> – Surface, LCF CIFS assessed as more critical than subsurface, HCH CIFS – CIFS relative to typical weld pass thickness – Proof test – Process controls – Design validation <ul style="list-style-type: none"> – Factor of safety and life
• Operational history	<ul style="list-style-type: none"> – Number of units/total starts/total time – Fleet leader – Number of units greater than 50% fleet leader – Hardware sectioning/inspection results – UCR history
• Engineering judgment (review process)	<ul style="list-style-type: none"> – What is our experience with this material, process, and configuration from a fabrication and operational viewpoint? – How well do we know the operating environment? – What is the credibility of the worst case consequence of failure?

Figure 72. SSME example summary sheet.

Nozzle Aft Manifold Weld Joint Reduction

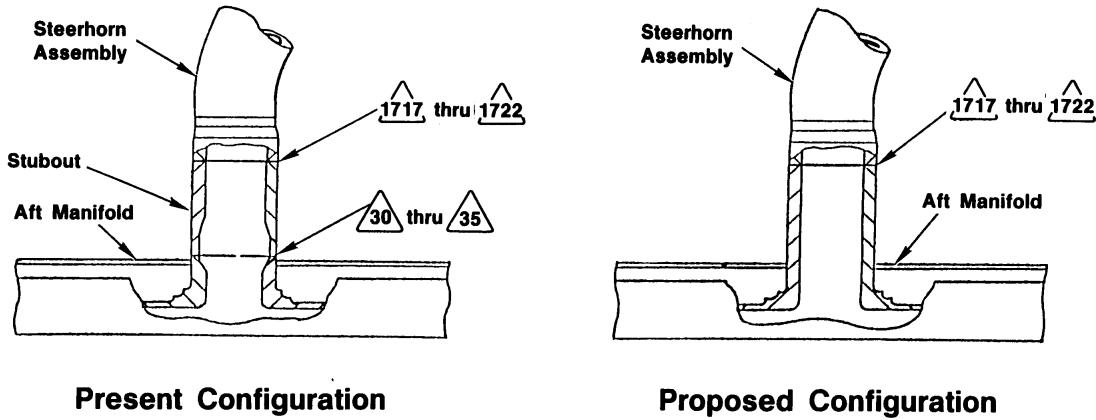


Figure 73. Example weld.

(9) Engine 2206 Low Pressure Fuel Duct Bellows Failure. Engine 2206 was an engine with 90 starts, 31,853 s, and the equivalent of 1,054 gimbal cycles. During hot-fire testing, the low-pressure fuel duct ruptured and started leaking. Figure 74 shows the duct, the engine duct configuration, and the location of the 4-in by 1-in fracture. The low-pressure fuel duct contains bellows installed to allow the engine to gimbal for control. The input to this duct is the low-pressure fuel pump rigidly affixed to the orbiter, therefore, the requirement for the bellows. The bellows have internal tripods mounted to the two duct attachment ends of the bellows and connected together with a ball and socket type arrangement (fig. 75). This allows a conjunction with the bellows and the duct to gimbal without extension due to internal pressure.

Engine 2206 LPF Duct Bellows Failure

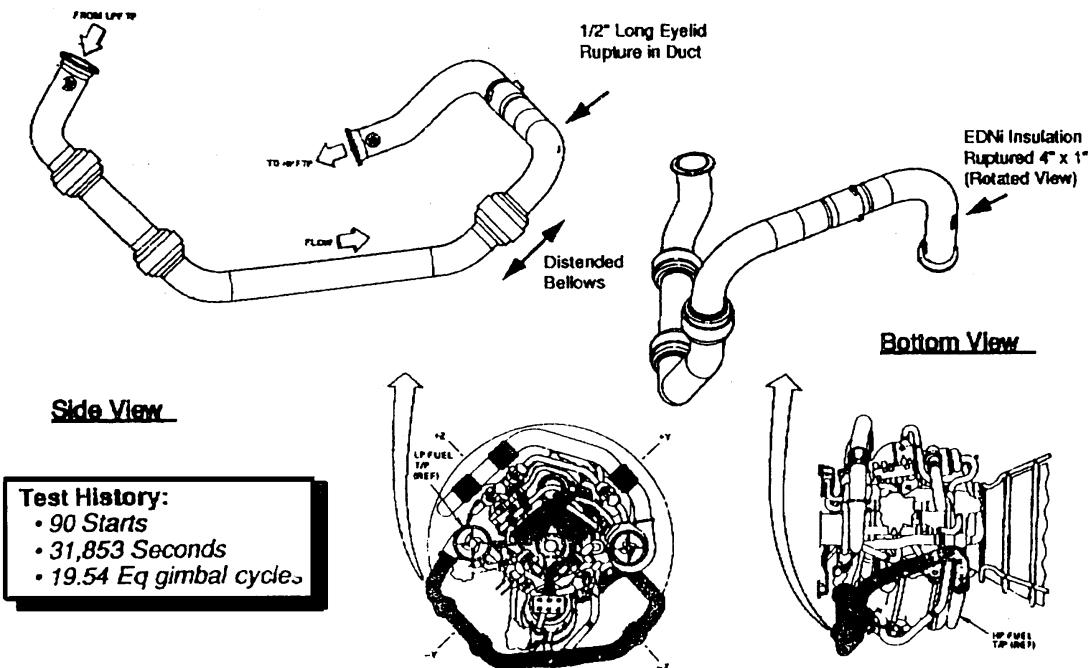


Figure 74. SSME duct.

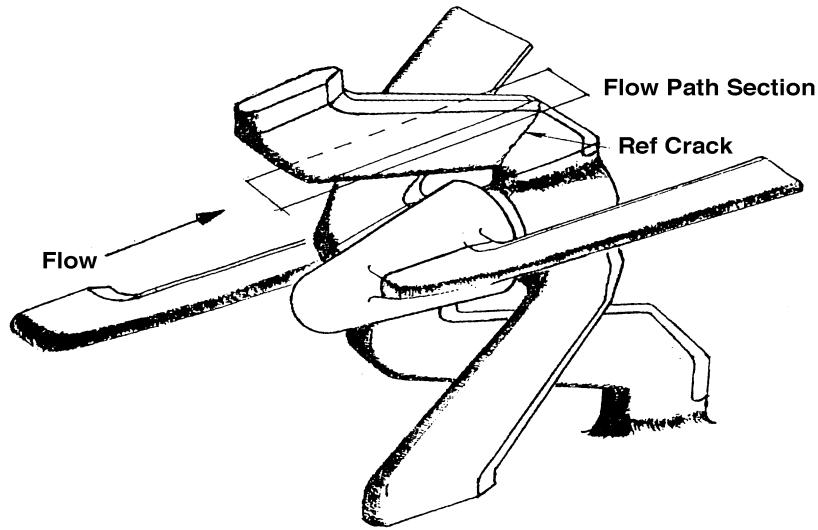


Figure 75. SSME bellows/gimbal.

The preliminary failure was a high-cycle fatigue failure of one of the legs of the tripods. The failed tripod allowed the bellows to expand and the failed part to break a piece loose. This part flowed down stream, impacting the line elbow creating the rupture and thus the leak. After impacting the duct, the failed part came to rest on the fuel flowmeter (figure 76). Though the bellows had passed $7\frac{1}{2}$ h of vibration testing to the 97-percent level in each axis (each independently run) without fatigue failure, the

ENGINE 2206 LPF DUCT BELLOWS FAILURE PROBABLE FAILURE SCENARIO

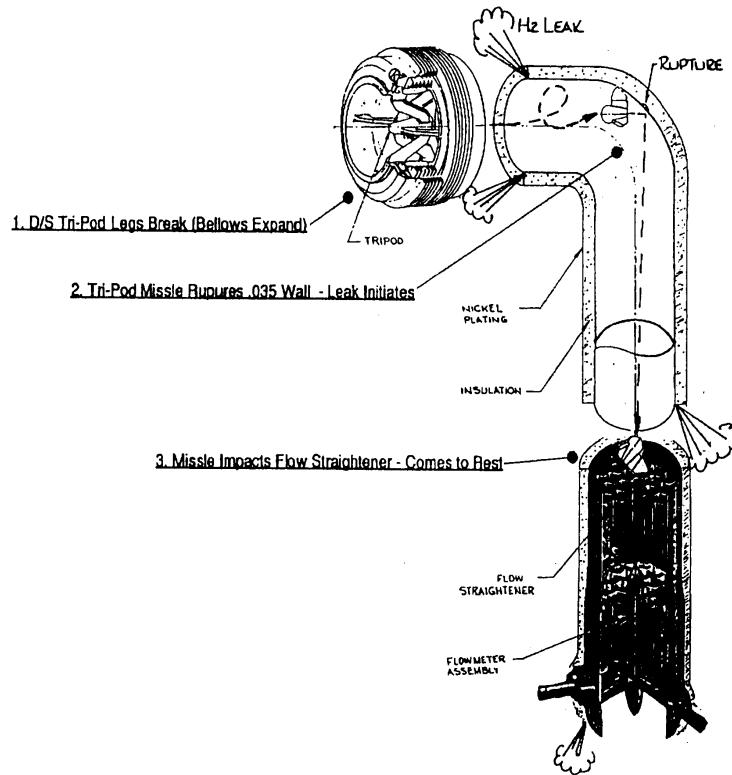


Figure 76. SSME impact failure zone.

part failed in hot fire. One possible cause was due to the vibration criteria used for the vibration testing that did not include internal flow excitation because it was based on hot-fire-measured mechanical data at the duct ends and other locations on the engines and its parts. This is one limitation of mechanical testing because it is not practical to simultaneously test flow and mechanical vibration environments simultaneously in component verification test programs. Other possible causes, such as excessive gimbal angle, etc., were basically ruled out. The real cause turned out to be higher-than-predicted stress due to a dogleg radius that was smaller than the specification required, compounded by a reduced cross section and higher stress concentration (K_t) due to discrepant fabrication. One other failure of the gimbal occurred during DVS testing. The ball and sockets galled, and one tripod failed. This problem was corrected for the flight bellows. All bellows have straight shields inside the bellows to eliminate flow-induced instabilities. Criteria were developed during the Saturn program for these shields, thus no flow instabilities of bellows have occurred on the SSME program.

Another lesson learned from the SSME audit was that environments should be well understood, which is not always possible due to complexities. In this case, everything was done basically correctly otherwise. Obviously if the system could have been designed with robustness, many problems would not have occurred, but this was precluded by weight constraints and performance requirements.

References 29 through 66 discuss some of the above and other resulting problems. Other problems are covered in the appendix. It is clear that early decisions and arbitrary constraints determined the design, its operational performance, and cost, reducing the potential for flexibility in design. Coupling that with the high-performance system, costly marginal systems evolve. A study of the numerous problems documented in the matrix (by projects), the items selected examples in references 29 through 66 and this report, demonstrates the high cost and limited margins of the hardware. Very little flexibility exists beyond the standard missions until the redesign efforts shown are completed. How one is to achieve robust optimized systems in the present austere environment is not clear; however, one can evaluate any part of the process from a system viewpoint and make improvements. That is the minimum that must be done.

(10) SSME Conclusions. The SSME is clearly one of the Nation's outstanding engineering achievements. Although it has experienced many development problems, they are not out of proportion to the technology readiness level at the beginning of the program, coupled with the extreme performance requirements it was driven to achieve, including operating at 109 percent of the original design thrust in order to make up for some of the performance losses created by other shuttle elements. Through this evolutionary history, the engine has been greatly improved. A good example of this is shown by the trending analysis of the high-pressure pumps vibration levels. Figures 77a and 77b show the Rocketdyne lox pump responses, clearly showing the effect of the damping seal introduction. Figures 78a and 78b are the same type plot for the Pratt & Whitney pump. Notice the sharp change with the vibration problem fix in mid-1992. The same basic trends have occurred for the flight pumps (figs. 79a and 79b).

The final SSME, including its new high-pressure turbopumps, should be completed by the end of 1997 and will eliminate many of the problems and concerns of this wonderful high performance engine.

Using SSME, the value of a sound ground test program has shown real merit. Future programs will have to have smarter ground test programs that have less tests but garners the information through better instrumentation and more inclusive test parameters based on design of experiments type statistics, and so forth.

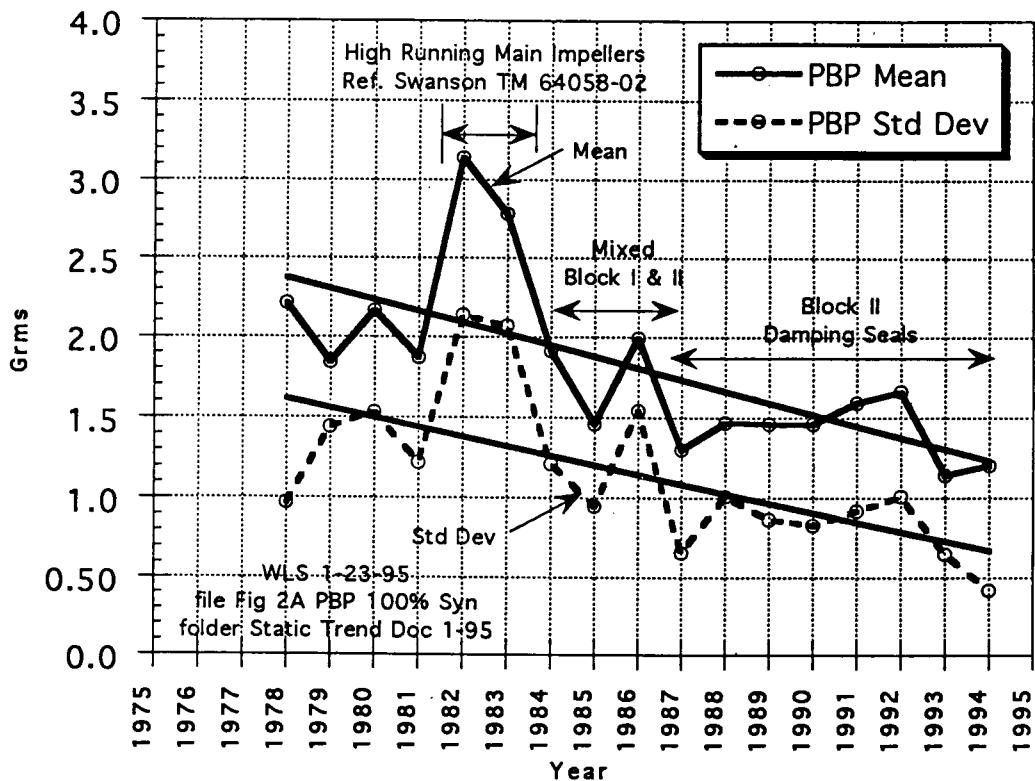


Figure 77a. SSME Rocketdyne lox pump 100-percent trend data.

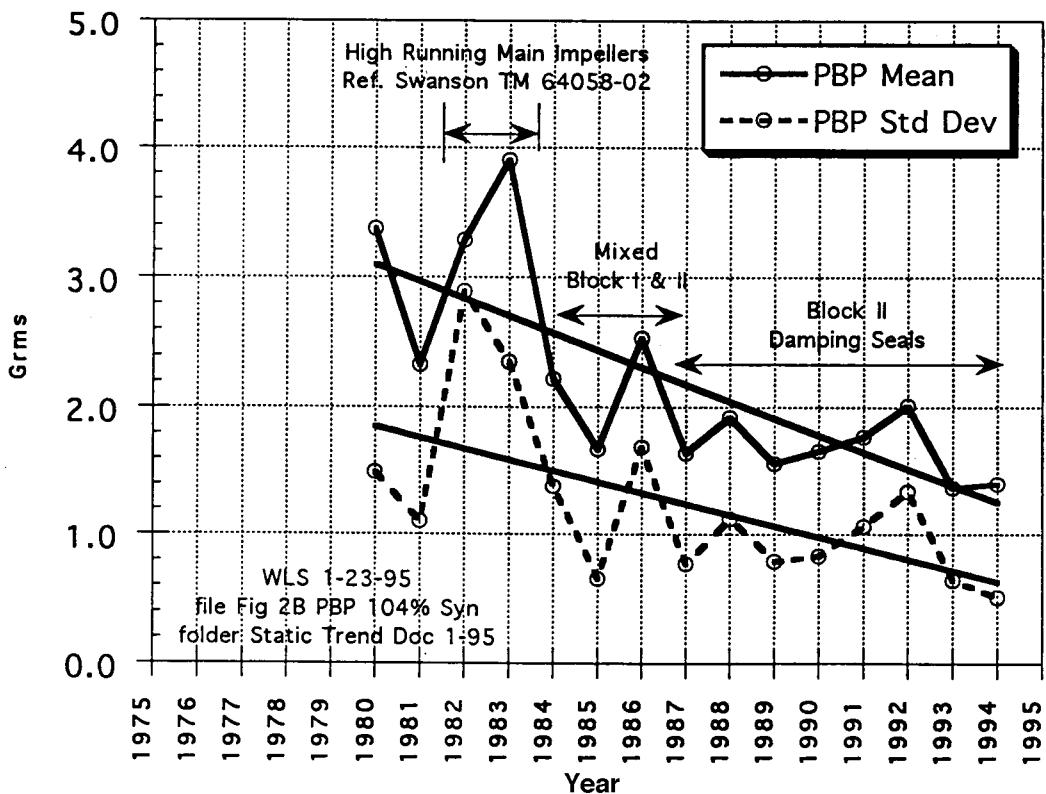


Figure 77b. SSME Rocketdyne lox pump 104-percent trend data.

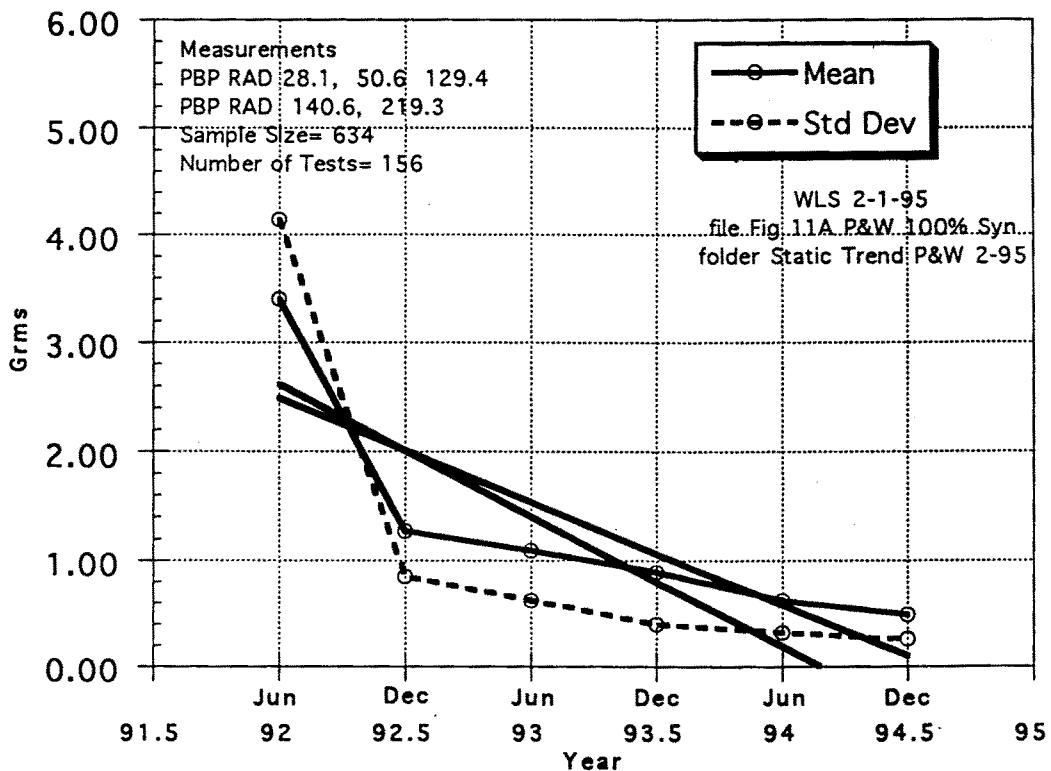


Figure 78a. SSME P&W lox pump trend data unadjusted.

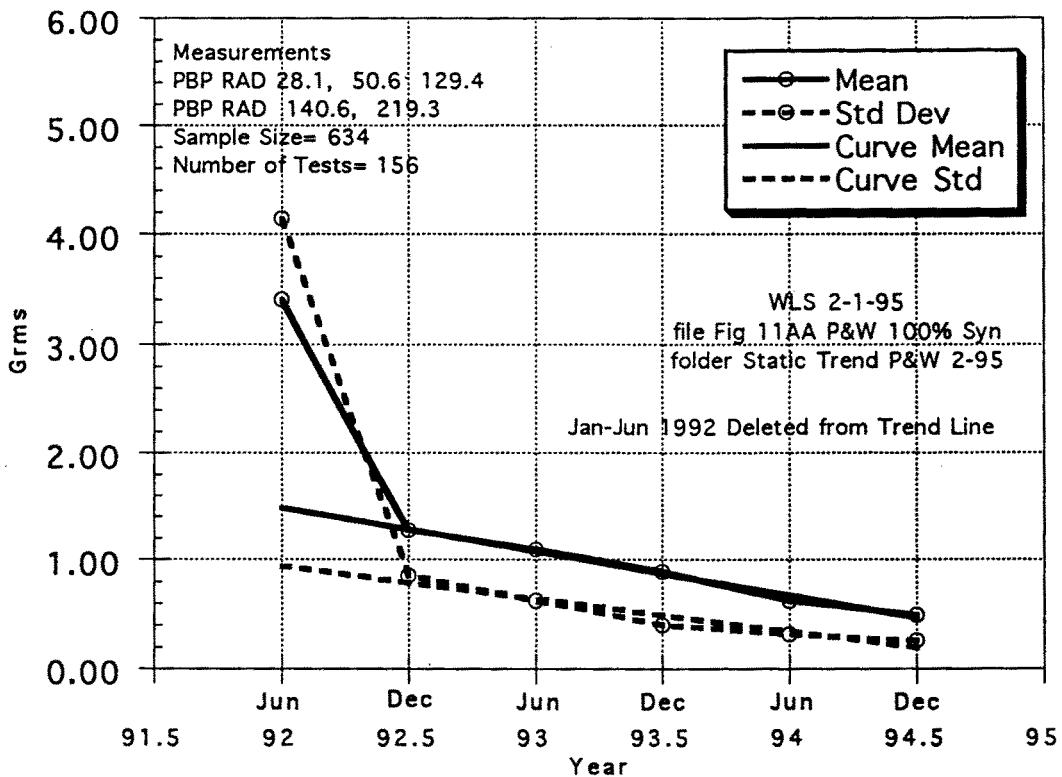


Figure 78b. SSME P&W lox pump trend data adjusted.

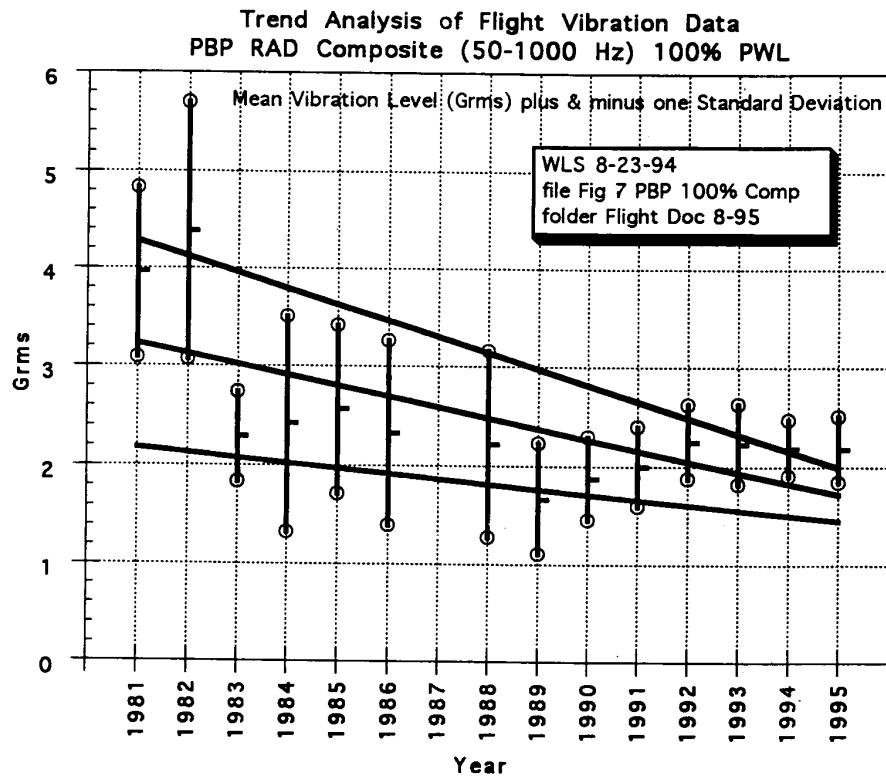


Figure 79a. SSME flight pumps trend data unadjusted.

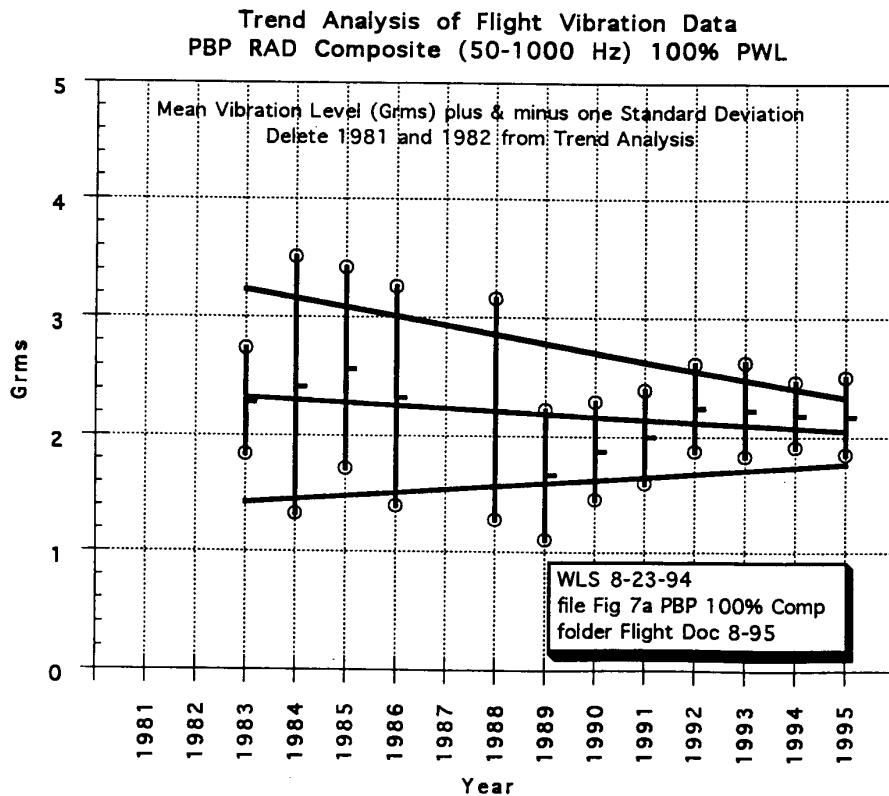


Figure 79b. SSME flight pumps trend data adjusted.

3. Solid Rocket Motors/Boosters

a. Characteristics

(1) Baseline. The SRM has evolved to the current configuration, through two earlier versions that actually flew on the shuttle for several flights, plus two configurations (the filament wound case and the ASRM) that were canceled before completion. The original motor that flew early space shuttle flights was replaced with the lightweight, high-performance motor (HPM) in order to gain vehicle payload capability to orbit. This motor flew until the STS-51L failure when it was replaced by the redesigned solid rocket motor (RSRM) to eliminate the O-ring problem that caused the STS-51L accident. The HPM is shown in figure 80. It is 146 inches in diameter, 1,500-inches (125-feet) long, and has a dry weight of 149,000 lb. Fully loaded, it weighs 1,260,000 lb and has a maximum thrust of 3,600,000 lb, with a maximum internal pressure of 980 lb/in². It is composed of 11 steel segments, including forward and aft domes. The segments are joined using tang and clevis factory joints at the factory in Utah to form four approximately equal-length segments. The four casting segments are shipped by rail to the launch site. The motor is assembled at the launch site using the field joints and is then assembled with the SRB parts, then with the other shuttle elements. The SRB parts include at least the aft skirt, the forward skirt, the recovery parachutes, control gyros, SRB-to-ET attach struts, and separation motors. The total system is shown in figure 80 and the combined dry weight of the SRM and SRB is 193,000 lb.

The RSRM was essentially the same as the HPM, but had features added that solved the problems that led to the STS-51L failure. The thrust profile (same as the HPM's thrust profile) is tailored to maintain maximum dynamic pressure constraints and maximum acceleration during the latter part of the burn time. The first redesign change was in the field joint configuration, which went from a single clevis to a double clevis and an additional O-ring (fig. 81). It also had J-flaps designed in the insulation for the field joint that provided additional joint sealing. This J-flap was made by extending the propellant-to-motor insulation and changing the end configuration (fig. 82). Other improvements were made during the down time such as the nozzle configuration, and the ignitor.

The SRB system was designed to be recovered using a parachute system composed of a pilot, drogue, and three main chutes. The main chutes went through two disreefings in order to ensure optimum performance and to reduce deployment loads. The concept was not to impact the prime SRM/SRB hardware design for recovery loads, but to accept hardware attrition if excessive damage occurred. Figure 83 shows the recovery system package for flight. Once recovered, the system had to be thoroughly cleaned of the salt water and other impurities, inspected for damage and flaws, proof tested again for flight safety, and then refilled with the solid propellant. This was necessary due to the harsh reentry thermal and acoustic environments, water impact, and towing loads, etc.

During the 1980's, with the need to increase performance, the SRM programs started the design and development of a filament-wound case. This program went through extensive development, including a structural test of many specimens and the case. Various problems, including cost and a field joint design susceptible to the STS-51L failure mode, led to program cancellation.

(2) ASRM. Based on the recommendation of the Rogers Commission following the STS-51L accident, the development of an alternate SRM was started. Advantages of the new start were used to enhance shuttle performance. This program was canceled for two basic reasons: (1) the RSRM reliability was demonstrated to be high, and (2) the NASA budget needed the ASRM funds to meet goals and conduct other programs.

The ASRM (fig. 84) was also 125-ft long and slightly larger in diameter than the RSRM at 150 in. With a weight of 1,220,000 lb of propellant and 134,000 lb of structure, the total weight was 1,350,000 lb. It had a longer burn time and higher performance. The thrust profile was tailored, as was the RSRM, and burned for 140 s. The field joint was a different concept, being a bolted configuration (fig. 85). The gimbaled nozzle was also redesigned to solve some of the problems associated with the RSRM nozzle, such as erosion pocketing.

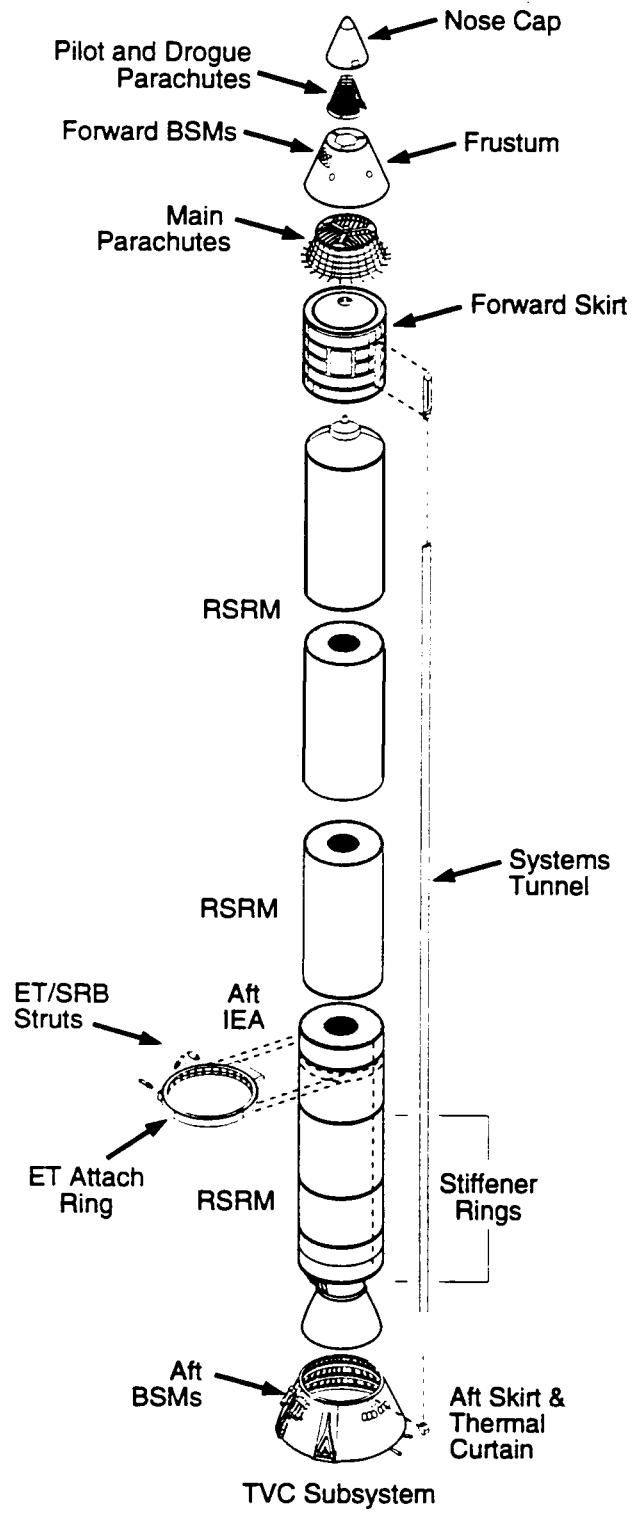


Figure 80. SRM HPM configuration.

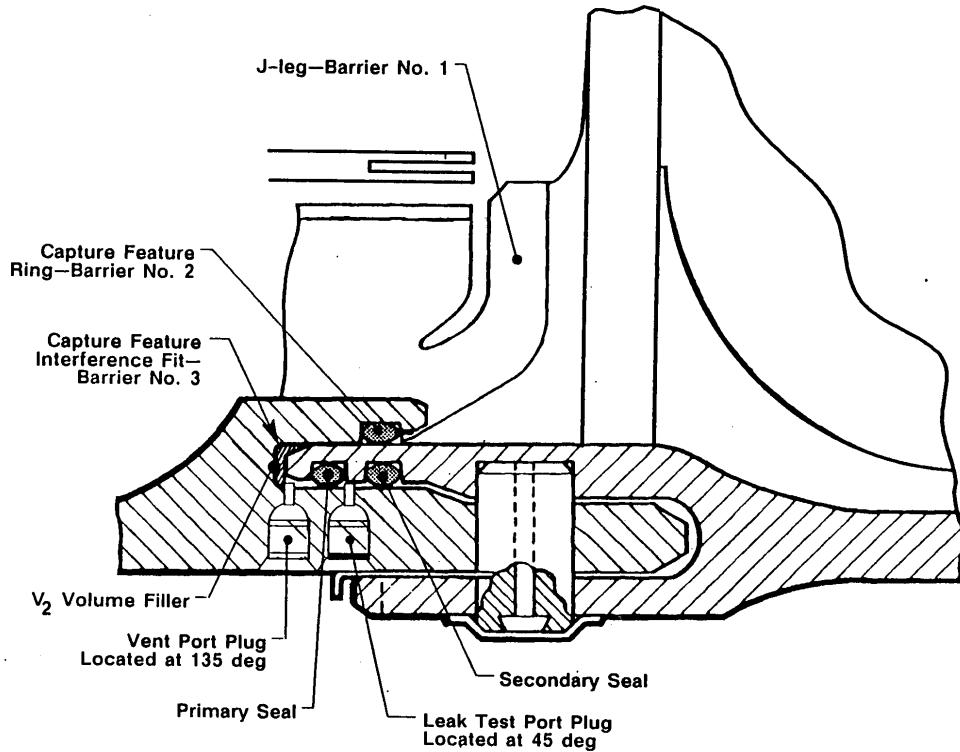


Figure 81. SRM joint redesign.

RSRM Field Joint Insulation (Cont)

- Comparison of CDR design to PDR and HPM designs

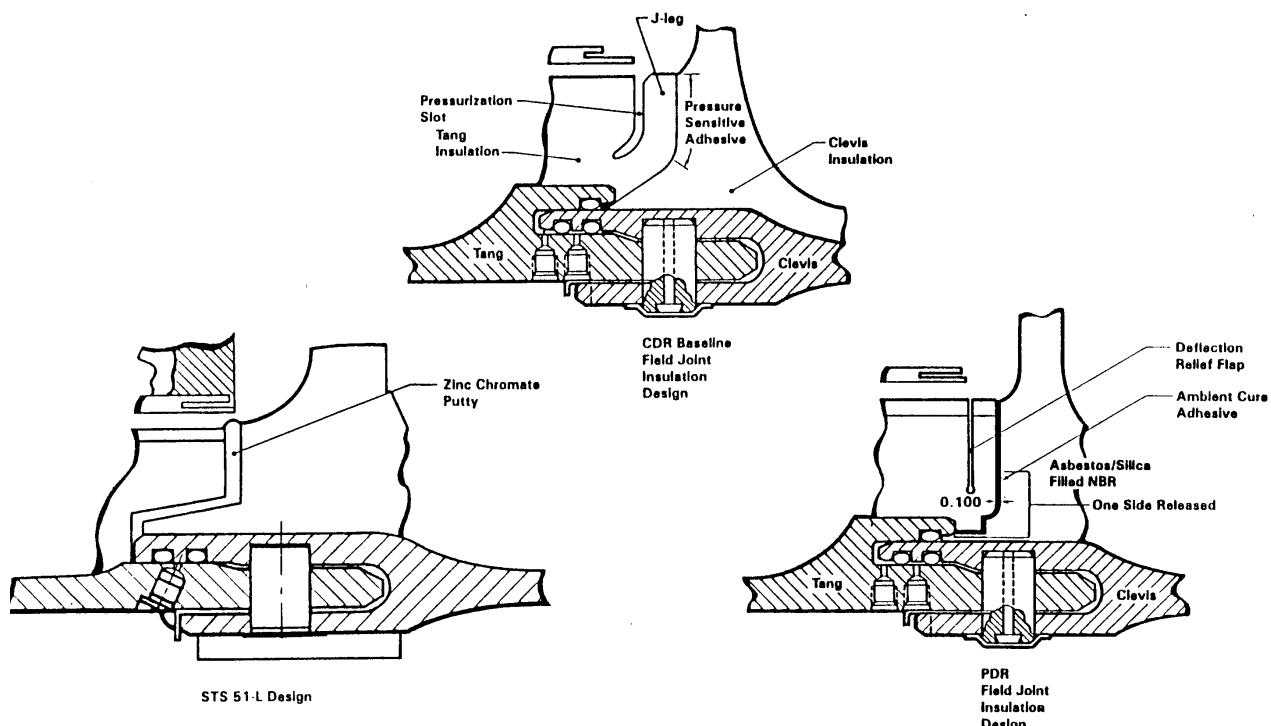


Figure 82. SRM J-flap.

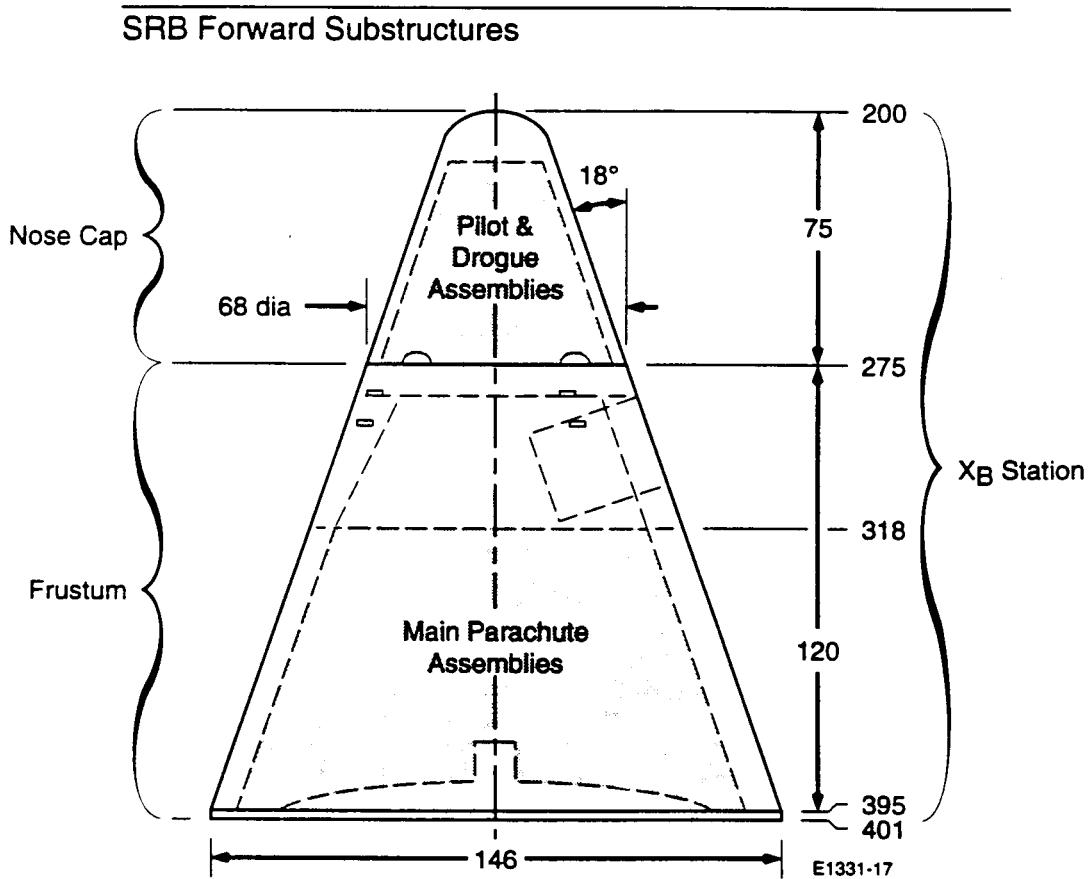
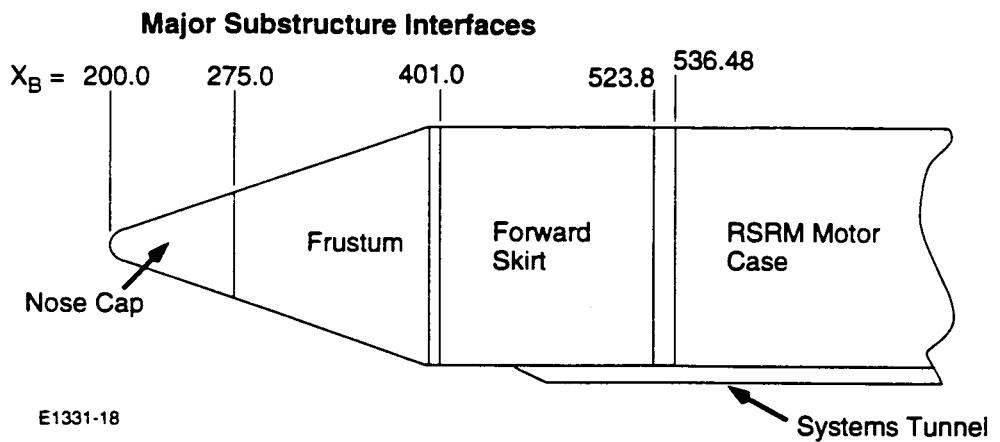


Figure 83. SRB recovery package.

THE ASRM

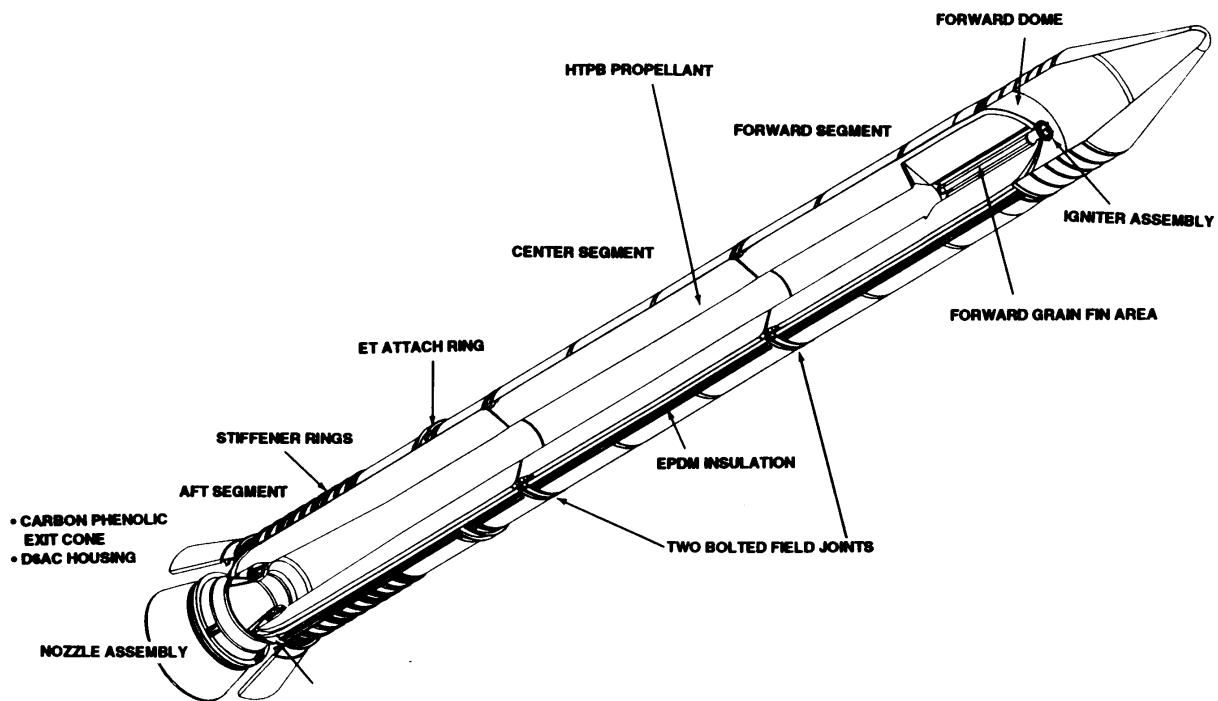


Figure 84. ASRM configuration.

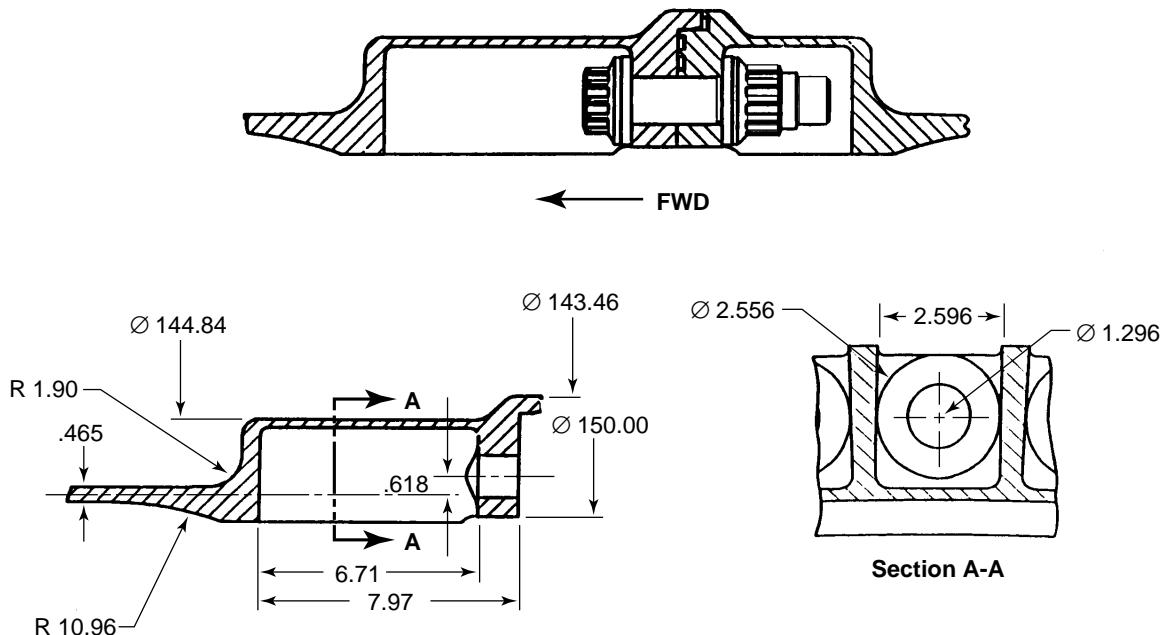


Figure 85. SRM field joint.

b. Evolution/Creativity/Innovation. Designing the SRB/SRM to maximum/minimum loads instead of time-consistent loads was an approach that saved redesigns and cost as the shuttle system evolved and loads increased. This approach early in the program was optimal since the dry weight of the SRM/SRB to payload to orbit was approximately 8 to 1. This meant that approximately 8 lb of structural weight would only penalize the performance 1 lb. Late in the development program as loads increased, time consistent loads could be used to reduce stresses, thus no redesign or weight increases were required as the loads increased.

In the process of reducing operational costs, it was decided to recover the SRB's. Recovering these elements meant that several technologies had to be expanded and applied to this mission phase. Reentry aerodynamic heating drove many aspects of the thermal protection system (TPS). Special testing techniques were required to quantify the induced environments and verify the TPS. Reentry aerodynamics were in the same category due to the large and fast-changing angle of attack. Reentry acoustics, due to the tail-first reentry and the long cavity coupled with the large varying angle of attack, were in the same category. Water impact on the booster was a major technological challenge due to both the speed and angles of impact. Additionally, the water entry angle created air pockets around the motor that collapse with large forces. Understanding and characterizing the sea state was a fundamental part of the water impact loads. Various type drop-test programs were created, including the need for pressure scaling to handle the cavity collapse. The Naval Ordnance Laboratory (NOL) pressure facility was used for this. The system problem of dealing with various initial conditions, from SRB separation (shuttle trajectory variations, separation motor variations, etc.) coupled with the SRB mass properties variation, the atmosphere characteristics, and the parachute system, led to the Monte Carlo techniques to define the conditions for parachute deployment, acoustic and thermal. A technique (innovation) was developed for accomplishing the reentry aero acoustics testing in a continuous-flow wind tunnel where the angle of attack could be varied (swept) real-time. Spectral analysis was accomplished, dictating the angles of attack where the acoustics were tuned and amplified. Once the designed angle of attack was determined, the system dwelled on that angle of attack and produced the design data.

Describing the dynamics of solid propellant was just as technically challenging in that its characteristics were very nonlinear (viscoelastic) as a function of propellant temperature, bore pressure, and oscillation amplitude. For application in analysis, a mechanical analogy was developed based on propellant coupon testing.

The ablative nozzle of the RSRM presents a major technological challenge in that it is a polymeric composite that chemically decomposes from thermal and structural response. This problem is nonlinear, interactive, and multidisciplinary in nature, which requires a totally new innovative approach to formulate the problem that couples the equations of energy, mass transport, and momentum, and to solve the equations simultaneously.

The use of the double clevis field joint along with the third O-ring for the RSRM was unique and practical since it allowed the use of all the case material already bought for the program and provided a joint that opened no more than 5 mils, whereas the SRM joint opened up to 35 mils. The case-to-propellant J-seal over the field joint was also innovative.

Dynamic modeling of the redesigned solid rocket booster (RSRB)/RSRM was very complex due to several factors: (1) nonlinear propellant stiffness, (2) high internal pressure stretching and ballooning the case, (3) pressure effects on the radial case stiffness, and (4) interface constraints (SRB-to-ET and SRB-to-MLP). Numerous nonlinear viscoelastic propellant tests were run to characterize the propellant modulus. Not only was the motor tested as a part of the full-up and $\frac{1}{4}$ -scale dynamic test, but it was tested $\frac{1}{4}$ scale in three different configurations as an element: (1) free-free; (2) horizontal on the SRB-to-ET attach struts and forward thrust post, unpressurized; and (3) configuration No. (2) pressurized.

c. Problem Examples. Many problems occurred during the development of the SRB/SRM, RSRB/RSRM, and the ASRM until program cancellation. A few were unique and will be highlighted here.

(1) Reentry Problems. The separation of the SRB's from the shuttle vehicle through the recovery in the ocean has experienced many problems. This phase of the SRB is very complex and has numerous interactions with the launch portion of the shuttle flight. In fact, this is a classical systems interaction/design problem. The SRM was not designed for the reentry/recovery events in a classical sense, but is assessed for attrition against launch (ascent) design requirements. This allows pushing the margins close, accepting some attrition and refurbishment for reflight as long as it can be shown to be safe for the next flight with a factor of four on life. The separation, reentry, water impact, and recovery are fundamental to successful recovery and reuse.

Problems have occurred in the areas of reentry acoustics, reentry thermal, parachute timing, parachute loads, parachute deployment (extraction from the nose cone), water impact loads, and SRB tow back. The most extensive loss was flight STS-4 where the water impact parachute release g-switch separated one leg of each chute at nose cone separation. Both SRB's were lost due to the resulting high impact velocities. The malfunctioning of the switch was due to external shock loads from the pyrotechnics for separation coupled with marginal g-switch settings.

As has been mentioned, the reentry phase is highly coupled to the launch phase. To illustrate this coupling, several different areas and problems that have occurred during flights are presented. In order to do this, various components of the SRB need to be understood. The SRB is composed of the SRM, which includes the ignitor, the nozzle, and the propellant. The motor is built up from 11 different segments, 4 of which are assembled at the factory (factory joints) into 4 resulting segments. These 4 segments are assembled at the launch site (fig. 86). The SRB is composed of all the additional parts integrated with the SRM to produce the thrusting elements, etc. (fig. 86). The additional parts are

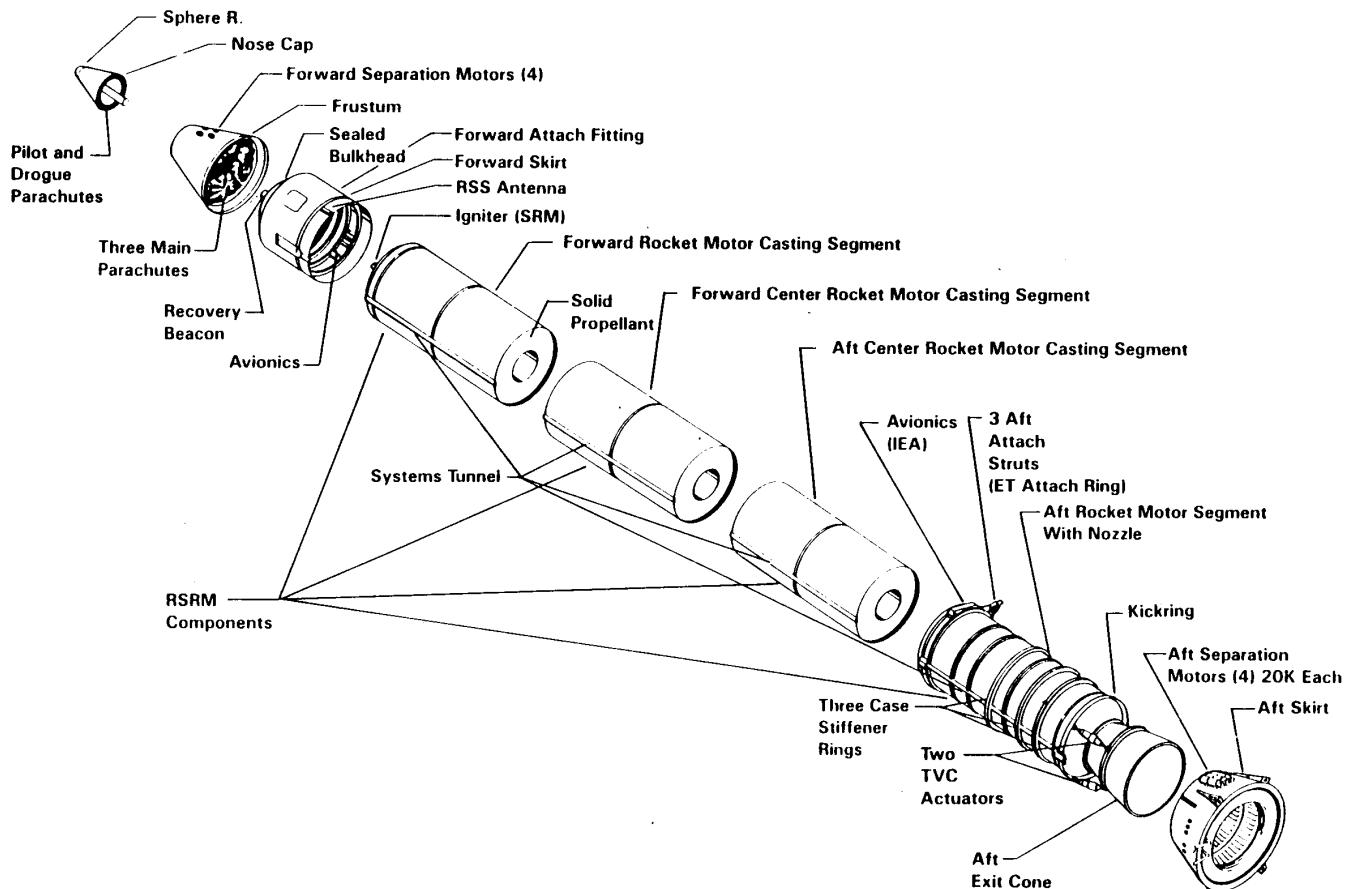


Figure 86. SRM segments and SRB elements.

parachutes, aft skirt, APU system, rate gyros, separation motors, range safety system, tunnels, SRB-to-ET attachment struts, forward skirt, nose cone, and SRB-to-launch pad attachments and separation bolts for the shuttle system. In addition, a thermal curtain is attached between the aft skirt and nozzle to protect all components contained in the aft skirt during ascent flight. This curtain is designed to break away during water impact in order to lower loads and not damage the nozzle.

The thermal curtain design has been interesting. In wind tunnel testing to derive reentry acoustics environments, it was discovered that the curtain would flutter and break up. During development flights, it was decided not to design a fix for this problem, due to complexity and cost, but to accept the failure during reentry, as long as ascent could be verified to be safe. The curtain has failed during reentry on every flight, evidenced by the jump in internal temperatures that are measured inside the cavity between the skirt and nozzle. Insulation is used to protect these parts for this environment increase. The heat loss is a function of angle of attack and the separation apogee. Due to the variation in the SRB separation conditions, angle of attack, and H (altitude), there have been three flights with hydrazine fires caused by reentry environments failing the hydrazine line. This fire increases the normal reentry thermal environments, damaging specific components. The ASRM further aggravated this problem due to its higher performance, longer burn time, increasing velocity at separation by 20 percent. Since this is a factor of 2 in terms of thermal environments velocity cubed, major concerns existed for redesigning the system to handle these environments.

Figure 87 attempts to capture this coupling between the various events. It starts with the ascent trajectory and its variations due to mission differences and to parameter uncertainties. These data become input data for the reentry trajectory analysis. This is usually accomplished using a Monte Carlo statistical analysis approach. The reentry trajectory determines the loads during parachute deployment, reentry, and water impact. Reference 39 discusses the water impact loads problem.

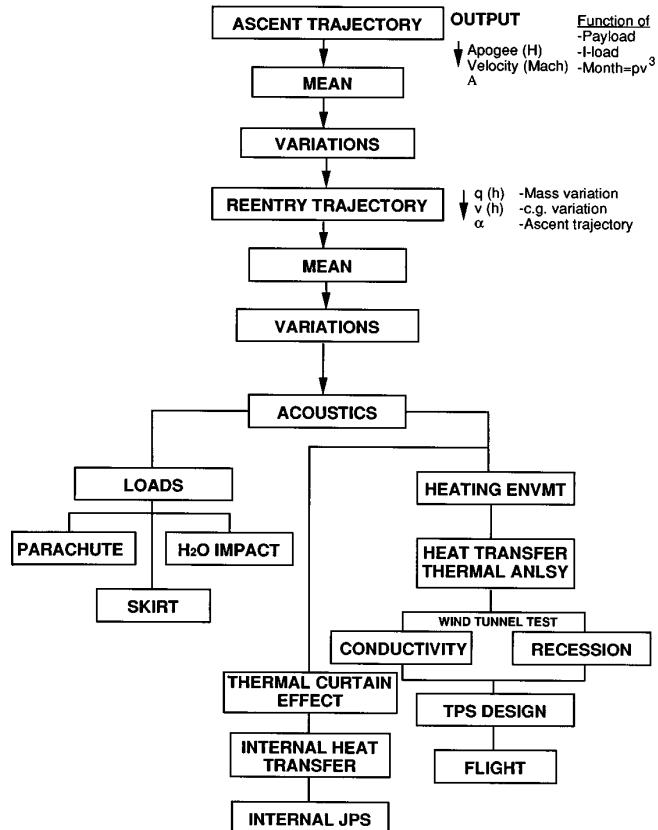


Figure 87. Recovery trajectory interdisciplinary coupling.

The parachutes have had loads problems associated with sequencing, reefing, and deployment out of the forward skirt. Reentry acoustics were a major design concern. The reentry acoustics are caused by an edge tone across the nozzle that resonates with the motor cavity. The SRB reenters the atmosphere tail first and broadsided, with varying angles of attack that induce these edge tones. Two distinct areas of thermal environments exist, the external and the internal after the thermal curtain failure due to flutter. Several flights have had instrumentation that has quantified these data.

(2) Nozzle O-Rings. In the summer of 1995, the ablative nozzle sections were found to have blued or scorched O-rings between the nozzle segments. Figure 88 is a cross section/schematic of the RSRM nozzle showing the assembly points and the O-rings. The assembly points are called nozzle internal joints. When these blued or scorched O-rings were found during disassembly and refurbishment after a shuttle flight, it was decided to delay the next shuttle flight until the problem was understood. A team was formed to investigate, understand, and recommend a solution to the problem. This was accomplished in record time.

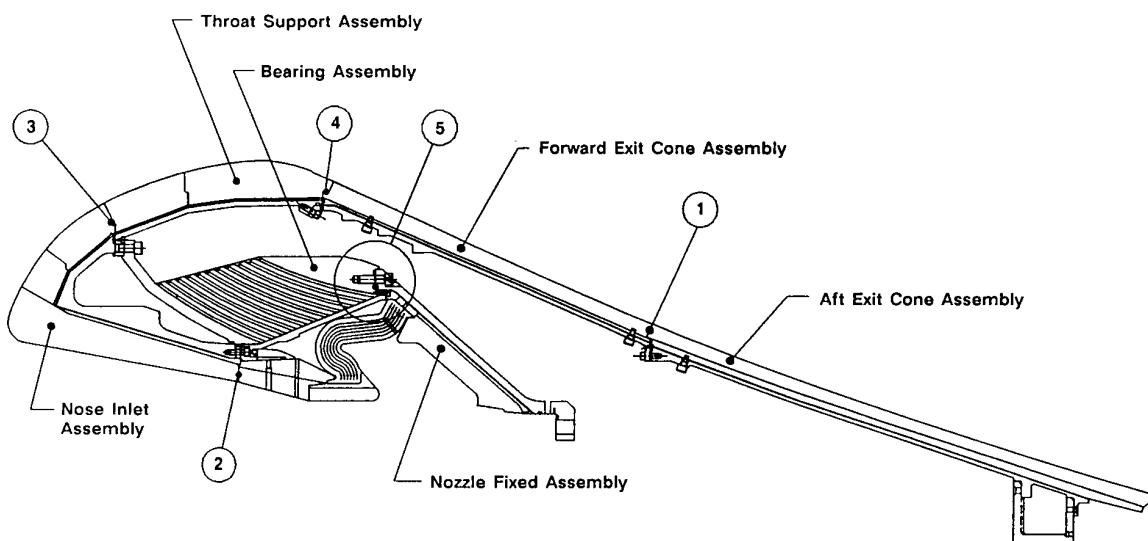


Figure 88. SRM nozzle configuration.

The problem was caused by compressed air trapped during the close-out/transition process in the vicinity of the sealant backfill (RTV) close-out/transition locations. It was believed to have been aggravated by a process change to deeper RTV back fill, which increased the gas path incidence and severity. Where the air was trapped created a blow hole in the RTV that provided a gas path to the O-ring (fig. 89).

The team came up with a procedure change that would eliminate the problem on future builds; however, nozzles were already assembled and in the field and on the shuttle that needed repair. A procedure was developed and certified to take out the original RTV and reinstall it, eliminating the close out blow holes above the joint inflection points. This repair was implemented for shuttle vehicle STS-69 RSRM's on the MLP.

(3) Aft Skirt. One major problem that has occurred on the SRB was two failures of the aft skirt during structural testing at approximately 1.28 times limit load, thus not meeting the 1.4 safety factor criteria.

The space shuttle SRB aft skirt failure illustrates the limitations of analysis/modeling. Early shuttle loads analyses conducted, using simplified models of the launch pad and the SRB skirt, produced a set of loads thought to be accurate for the prelaunch SSME thrust buildup phase of launch. It was understood that a major skirt load would arise from the vehicle weight combined with the SSME

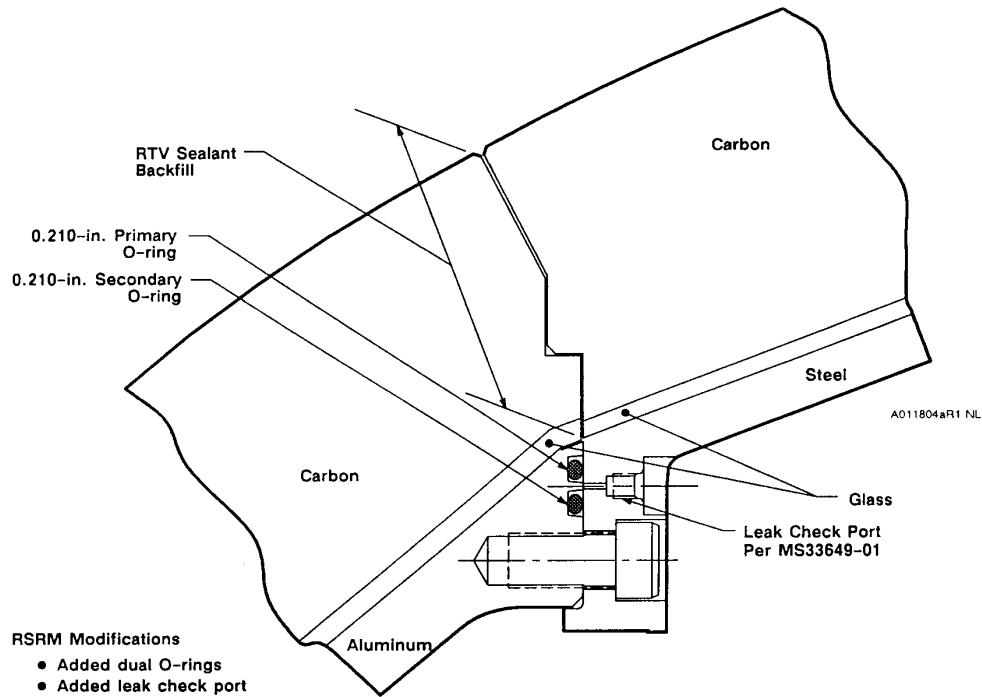


Figure 89. SRM RTV at nozzle joint.

thrust-induced load (figure 90). At full thrust, the four holdown posts away from the orbiter are loaded in compression, not only from weight but also from the vehicle bending due to the SSME thrust. What was not understood was the sensitivity of the local weld stress near the holdown post to the pad stiffness. Figure 91 is a stress field plot of stress amplitude versus circumferential location for one-half of

• **Aft Skirt Post With Axial, Radial & Tangential Load**

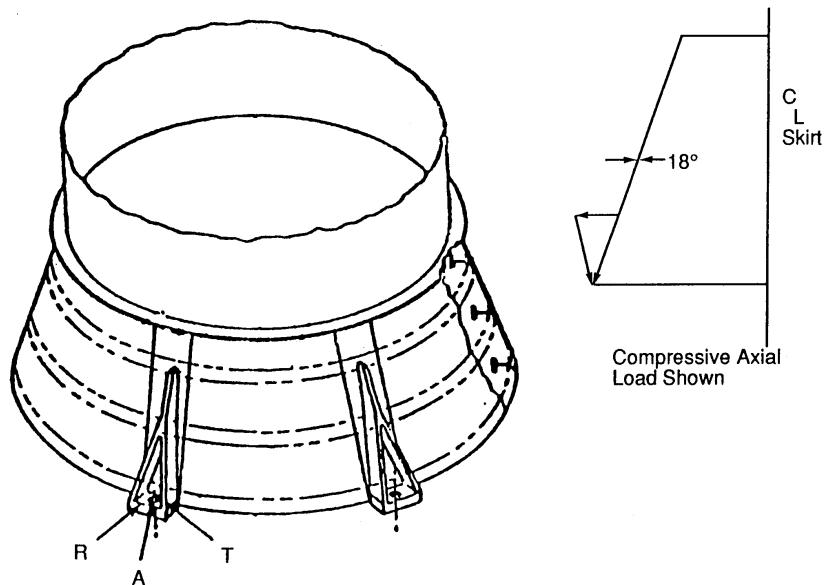


Figure 90. SRB aft skirt.

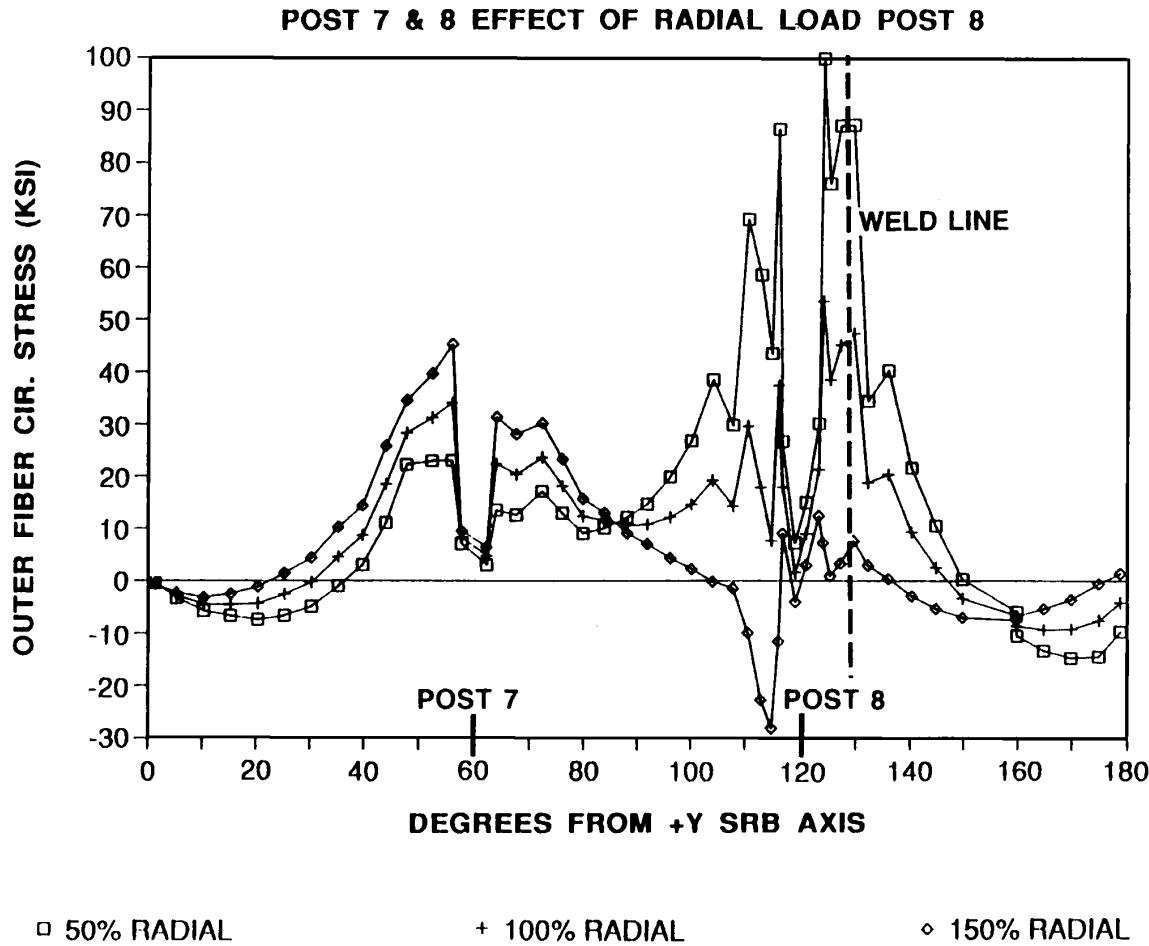


Figure 91. SRB radial load sensitivity.

the symmetrical skirt showing the large peak at the weld for different load variations. Figure 92 is a detailed finite element stress analysis localized around the weld failure point, showing the large stress intensity at the weld near the bottom of the skirt.

Part of the loads sensitivity of the aft skirt is due to the holddown mechanism, which is composed of the spherical bearing, bushing, and aft skirt shoe. The bushing is offset so that the mating of the SRB to the pad could be enhanced. Since the aft skirt load is a function of the radial displacement of the skirt due to the post deflection, this symmetry can be used to prebias the skirt post inward, reducing the effect of the applied load and thus increasing the structural margin. The approach is currently being used on shuttle flights to increase the margins.

In addition, a lot of effort has been expended by United Boosters Technologies, Inc. (USBI) and MSFC engineers to design and verify a fix to relieve the stress concentration caused by the radial bending at the forging weld interface. The load concentration, as seen on figure 92, shows that the radial movement of the very stiff holddown post forging causes the bending in the skin at the weld near the post. In order to distribute this load through more of the skin and bridge, the weld (a simple bracket) was designed and built to accomplish this task (fig. 93). A bone yard skirt was taken and cut into four segments to verify the design. Figure 94 shows how a test segment was cut from the skirt including the test setup. The first segment was tested to failure without the fix to verify that the segment would indeed match the two total skirt test failures. With this accomplished, two segments had the fix installed and tested to failure. The fix reduced the stress locally by a factor of two, thus restoring and surpassing the safety factor requirement. To date, the fix has not been implemented for flight but is ready if needed.

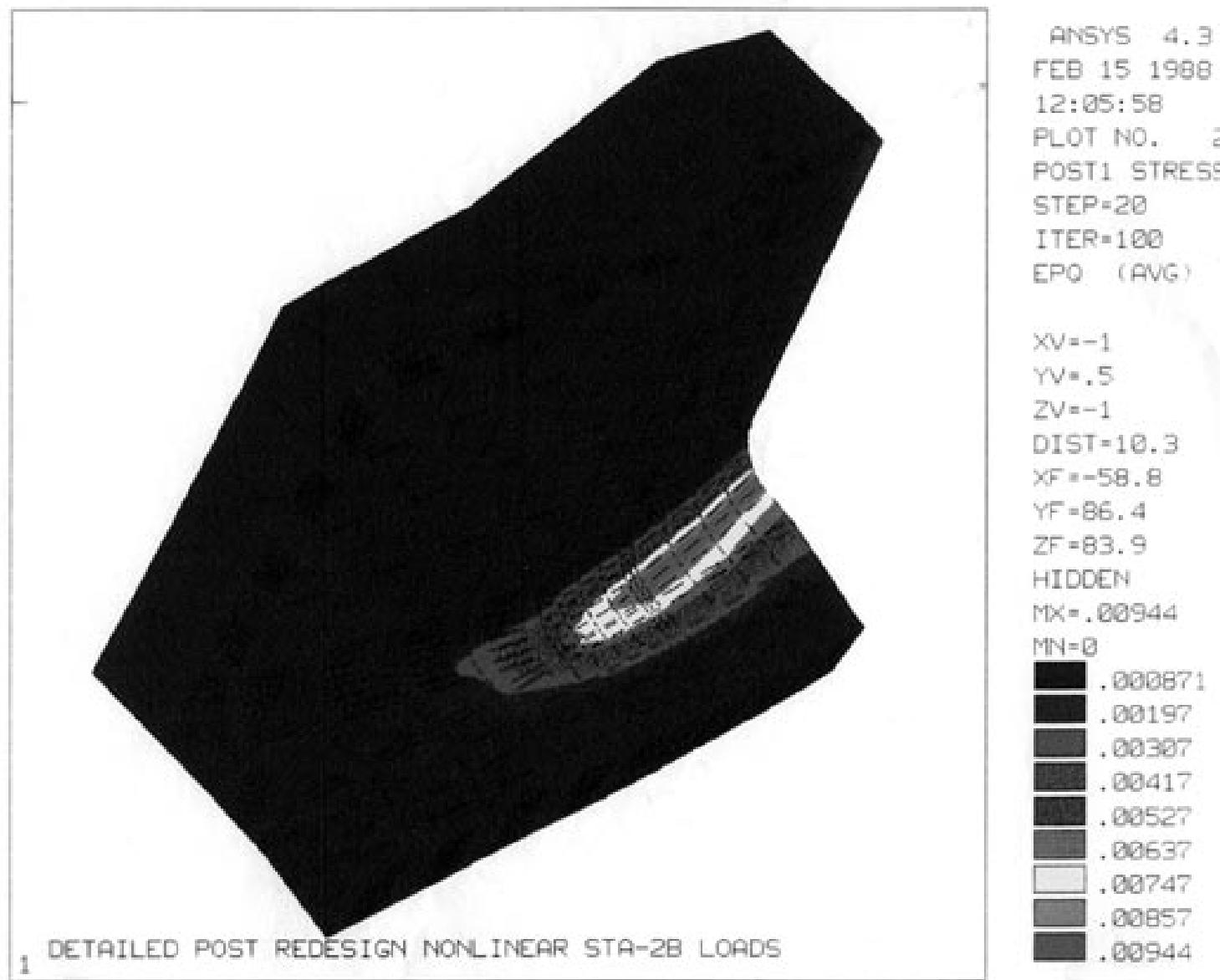
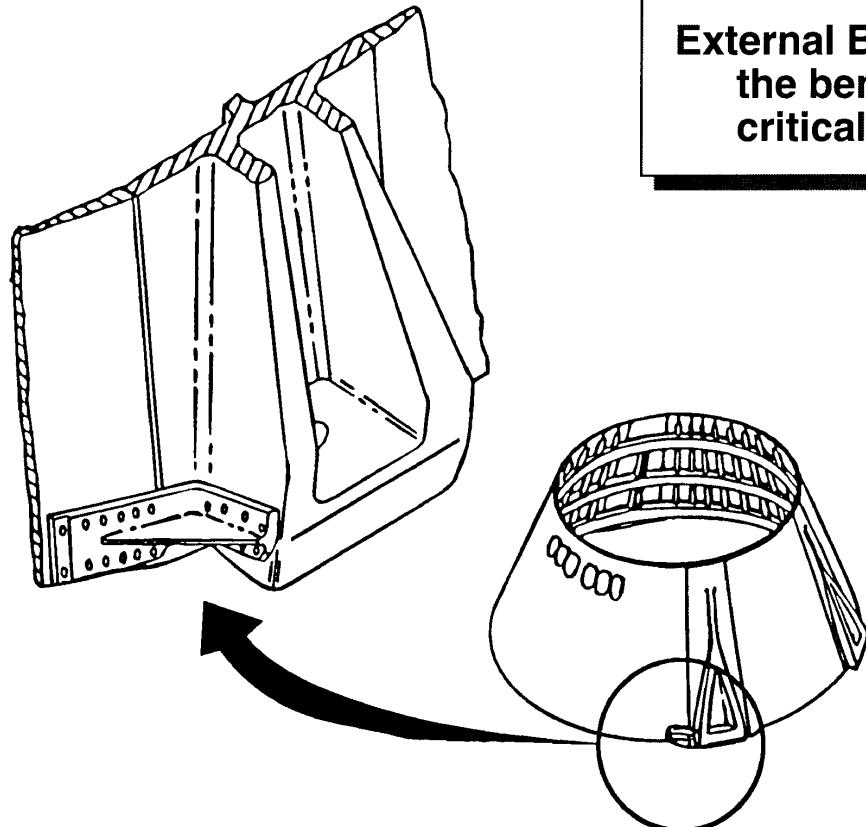


Figure 92. SRB stress concentration.

The External Bracket Modification Concept:



External Bracket is designed to reduce the bending stress in the weld at the critical location.

Figure 93. SRB skirt bracket fix.



SRB

Background

- Multiple options investigated to increase aft skirt holdown post weld factor of safety above current 1.28
- External bracket selected as optimum solution based on detailed analysis
- Component test program conducted to verify effectiveness of bracket
 - Four test articles removed from new non-flight aft skirt
 - Fixture designed and constructed to produce flight (design) strains in weld
 - Predicted weld strain reduction confirmed by tests of external bracket vs. baseline test articles

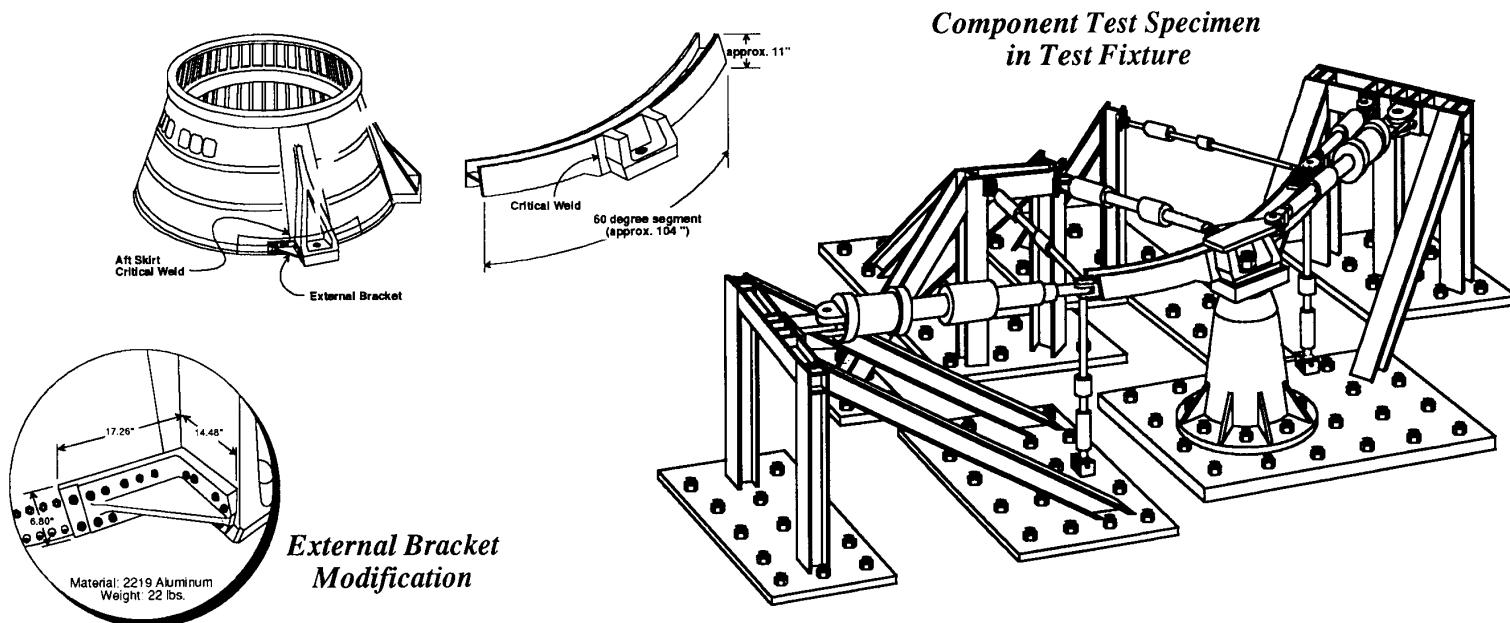


Figure 94. SRB skirt test approach.

(4) Parachute Release G-Switch Failure. A very interesting problem occurred on STS-4. The parachute G-switch activated early on both SRB's, releasing the chute from each SRB, resulting in a loss of both the SRB's. The G-switch was installed to release one of the two parachute risers at water impact, enhancing SRB recovery. In this case, the switches were activated during the time of frustum severance before chute deployment during the atmospheric reentry phase. The severance is accomplished using a linear-shaped charge. Figure 95 shows the frustum chute package and linear-shaped charge. The G-switch was mounted to the SRB skirt after the linear-shaped charge on the IEA firewall, which was mounted on the forward skirt reaction ring (fig. 96). The problem occurred in this manner. The linear-shaped charge caused a shock wave in the SRB skirt, which excited the reaction ring in its first cantilever mode of 85 Hz. This drove the IEA, hence the G-switch. The G-switch natural frequency was near 70 Hz and should have been critically damped. The G-switch response triggered the chute attach bolt, hence the loss of the chute performance and the SRB's. The first three shuttle flights (six SRB's) had no problem; however, STS-4 had the same problem on each SRB. Why the difference? It turned out after much testing (pyro and G-switch response) that all flights were near a problem. These two SRB's exceeded limits because of two factors relating to G-switch damping. The temperature decreased the damping, and the switches were fabricated on the low damping side, but not outside the specifications. This, combined with the expected variation in pyro shock response, led to the problem. No one thought that the G-switch of such a low frequency would respond to the shock, based on the qualification test run without the resonant mode of the support ring, hence the problem was missed. Many months of investigation time were required to find the culprit, although all pieces of data existed prior to the failures. The problem was putting the pieces together in the proper manner. No other G-switch problems have occurred in the other 70 plus shuttle flights.

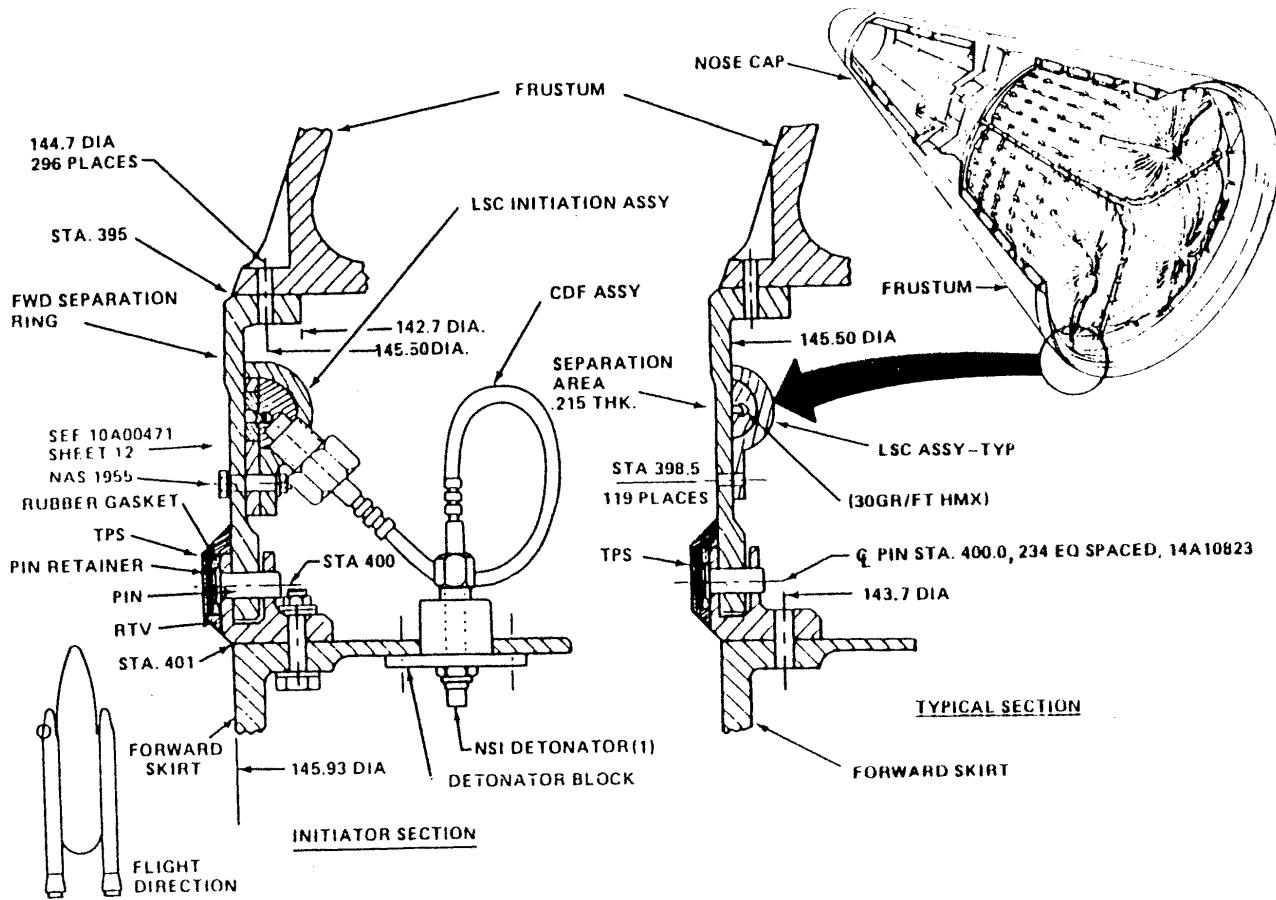


Figure 95. SRB frustum chute.

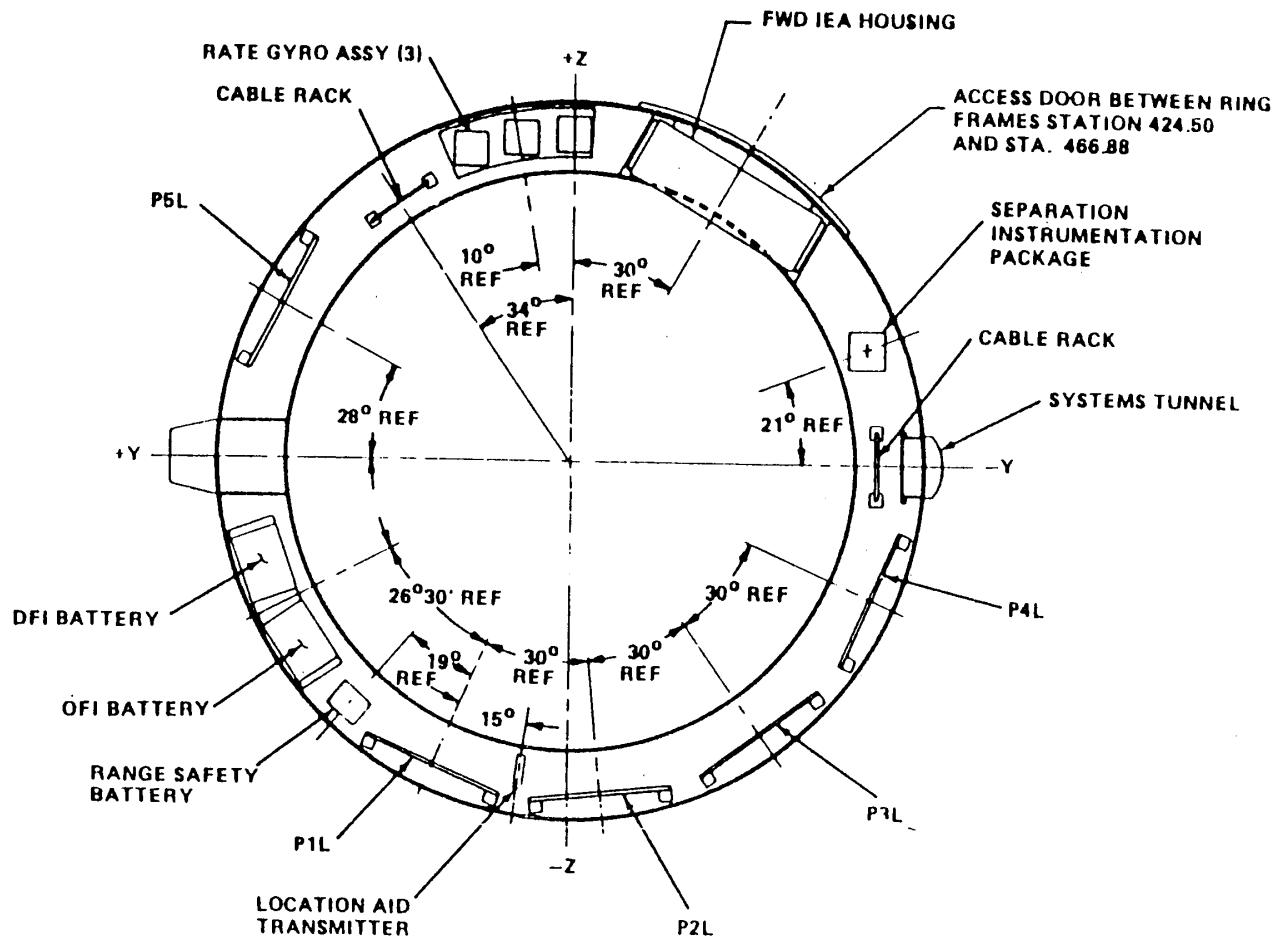


Figure 96. SRB G-switch mounting.

4. External Tank. There were three designs of the ET; the first two have been operational, with the third to fly for the space station missions in order to meet the payload requirements of these demanding missions. The standard ET flew on six of the first seven space shuttle flights. It was replaced with the lightweight tank (LWT) in order to gain approximately 10,000 lb of payload lost in misunderstood phenomena of the system. This tank has successfully flown more than 80 missions. The third tank, super lightweight tank (SLWT), is under development using aluminum-lithium (Al-Li) to gain another 8,000 lb payload in order to accomplish the space station missions.

a. Characteristics

(1) Standard and Lightweight Tank. Since the LWT tank has been the workhorse for the shuttle, the characteristics of the standard tank will not be presented. The major differences in characteristics were the skin thickness. The ET is a major element of the liquid propulsion system. Since it essentially goes to orbit, its weight uncertainties and impacts are a direct impact to vehicle performance; therefore, it is very critical that excessive criteria and margins not be used. It has two tanks and an intertank (fig. 97), consisting of a lox tank that is designed to contain 1,394,891 lb of oxidizer. It measures 655.5 inches in length, with an outside diameter of 331 inches, and it has a volume of 19,563 ft³, with an empty weight of 12,350 lb. It has internal slosh baffles to dampen the propellant sloshing that couples with the attitude control system and antivortex baffles to dampen the vortexes of the flowing (draining) liquid. The tank is insulated with external insulation sprayed on the tank with an automatic system other than for close-outs. The hydrogen tank (fig. 98) is designed to contain 257,500 lb of liquid hydrogen.

DESCRIPTION

The LO₂ tank is a fusion-welded assembly of preformed, chem-milled gores and panels, and machined fittings and ring chords.

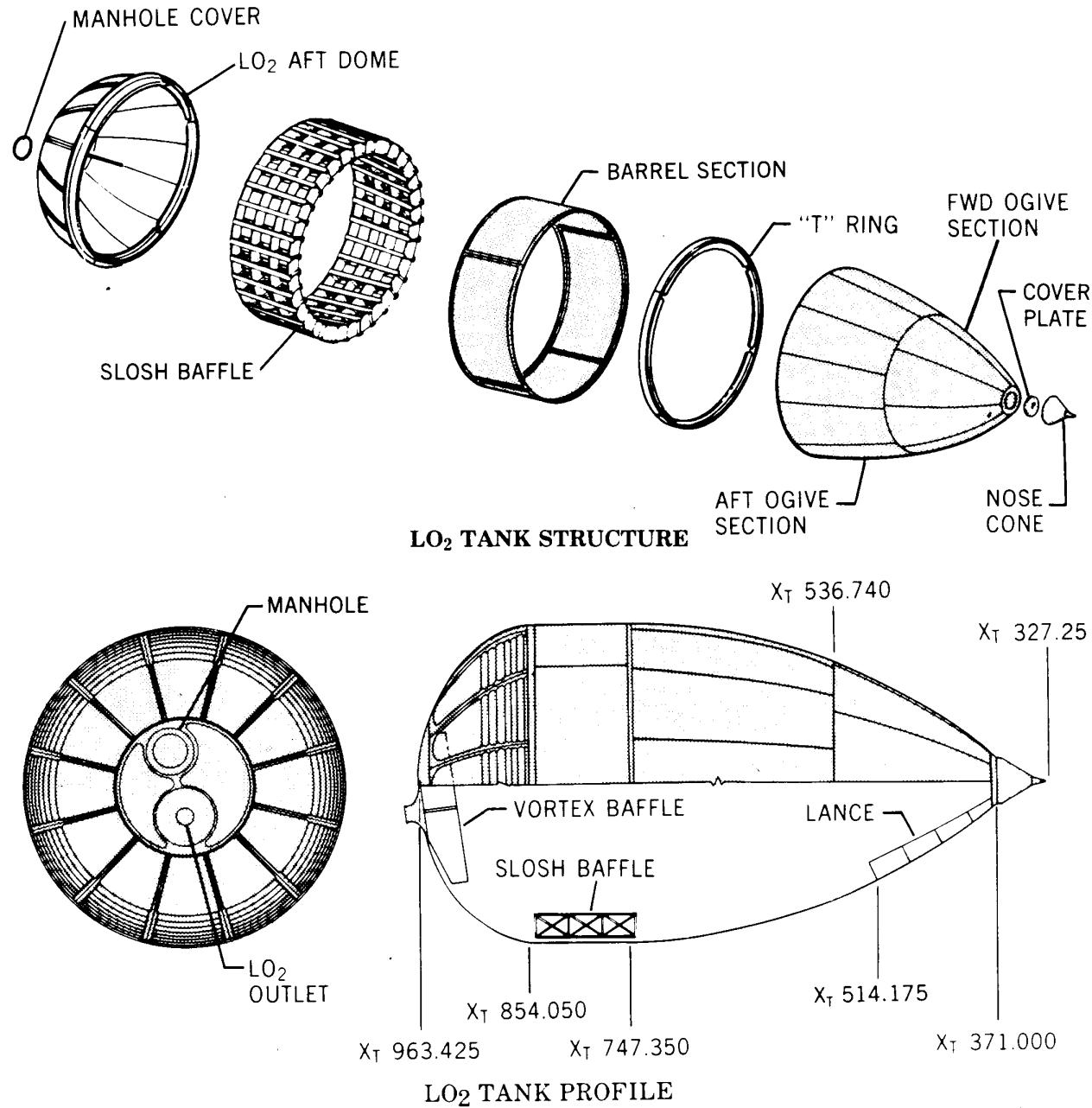


Figure 97. ET lox tank configuration.

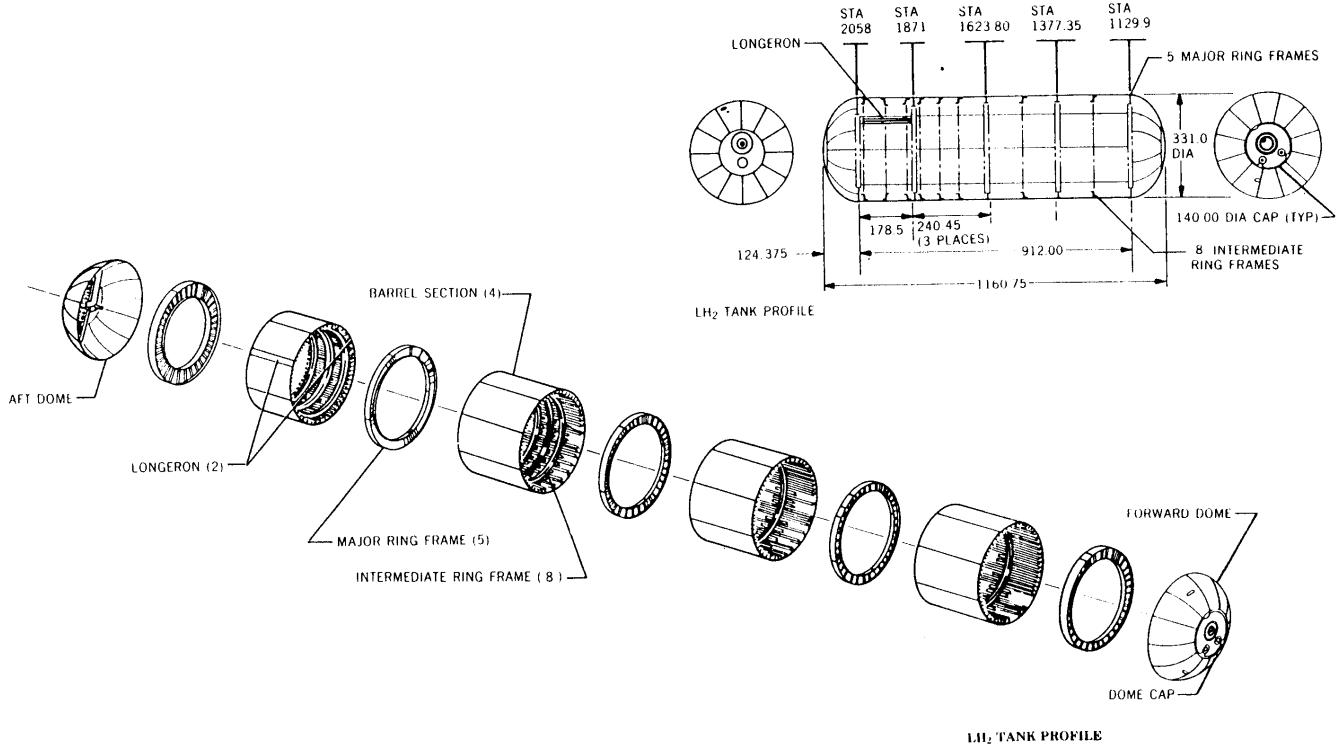


Figure 98. ET hydrogen tank configuration.

It measures 1,160.78 inches long, with an outside diameter of 331 inches, a volume of 53,518 ft³, and an empty weight of 28,000 lb. At the rear section, a large ring frame is installed inside the tank. It takes orbiter thrust loads from the thrusting engines and the SRB moment reaction and transfers them into the tank wall. In addition, there are longhorn stringers in this section, 187 inches long, to further handle the orbiter-induced thrust loads by distributing them throughout the tank skin.

The intertank (fig. 99) is the ET structural connection that joins with both the lox tank and the hydrogen tank to provide structural continuity between these assemblies. Its primary function is to receive and distribute all thrust loads from the SRB's and transfer loads between the propellant tanks. This is accomplished using a massive box beam across the tank diameter, with SRB forward attachments on each end of the beam. The intertank also functions as a protective compartment for housing the operational instrumentation and range safety components. In addition, it provides an integral ET intertank carrier plate assembly that interfaces with the fluid, electrical, and pneumatic system ground facilities, a nonstructural access door, two pressure vent openings, and aerodynamic fairing for the lox feedline, the gaseous hydrogen (GH₂) pressurization line, and two electrical cable trays.

The intertank cylindrical structure consists of eight 45° mechanically joined panels (two thrust and six external stringer-stiffened skin panels), a main ring frame, four intermediate ring frames, and the SRB beam assembly with two forged thrust fittings. It is 220 inches long, has a 331 inch outside diameter, and weighs 12,200 lb.

The pressure tank is made essentially of 2119 aluminum (Al). The process takes the forgings and rolls them into large panels, which are then machined to the basic stress field in order to save weight. The panels are shaped then welded, using large forms to maintain shape. There is over one-half mile of welds in the total tank.

DESCRIPTION

The intertank is the ET structural connection that joins with both the LO₂ and LH₂ tanks to provide structural continuity between these assemblies. Its primary functions are to receive and distribute all thrust loads from the SRB's and transfer loads between the propellant tanks. The intertank also functions as a protective compartment for housing the operational instrumentation and range safety components. In addition, it provides an integral ET intertank carrier plate assembly that interfaces with the fluid, electrical, and pneumatic systems ground facilities, a nonstructural access door, two pressure vent openings, and aerodynamic fairings for the LO₂ feed-

line, the GH₂ pressurization line and the two electrical cable trays.

The intertank cylindrical structure consists of eight 45-degree mechanically joined panels (two thrust panels and six external stringer-stiffened skin panels), a main ring frame, four intermediate ring frames, and an SRB beam assembly with two forged thrust fittings.

The use of the intertank makes it possible for the ET to have separate propellant tank bulkheads, avoiding the design complexity and added operational constraints associated with a common bulkhead configuration.

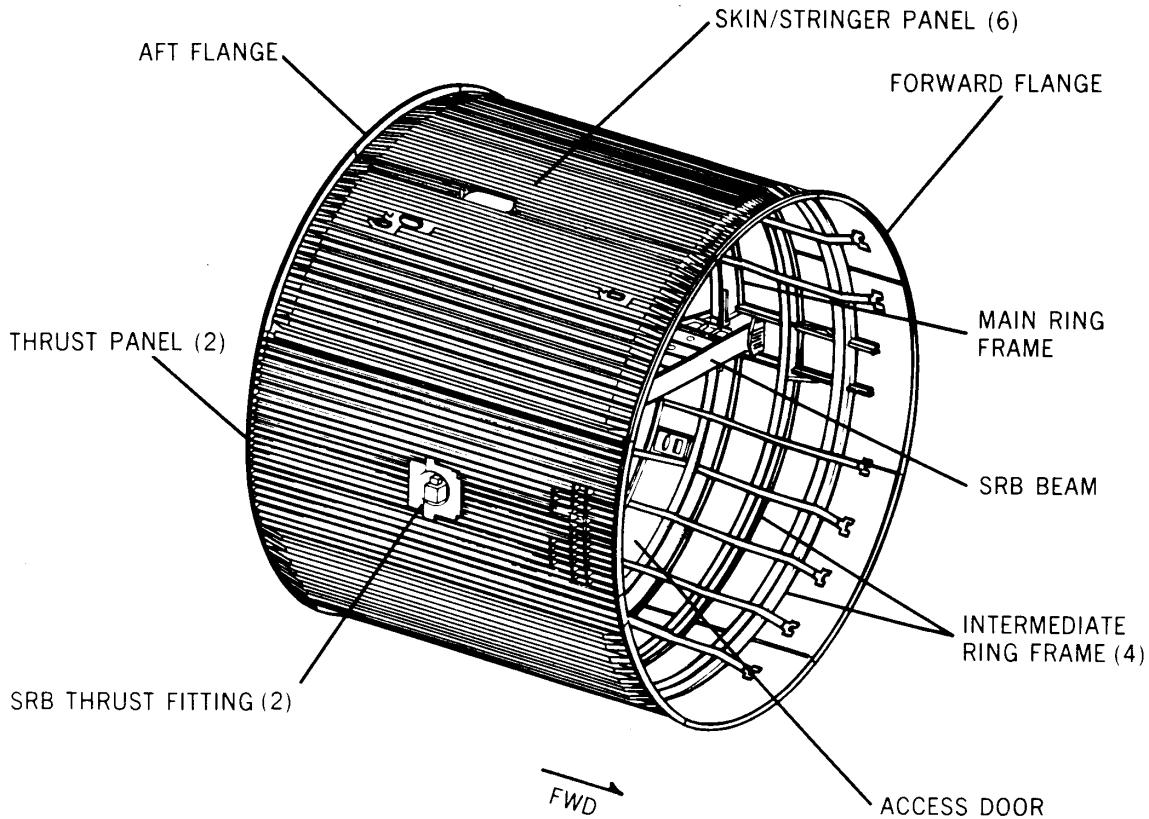
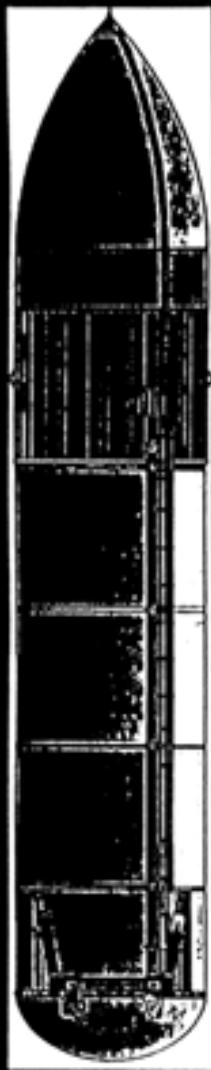


Figure 99. ET intertank.

(b) Super Lightweight Tank. The decision to join with the Russians for the design of the *International Space Station (ISS)* created the need for a 51½° orbit inclination. The shuttle, in its present configuration, cannot deliver the complete *ISS* elements into that orbit without performance enhancements. A major step in performance enhancements can be made by substituting aluminum-lithium (Al-Li) for the current 2219 Al in most of the tank structure. Also, further gains can be made through the use of an orthogrid structure milled into the tank panels. Approximately 8,000 lb of payload enhancement can be achieved with the SLWT. The tank's basic geometry does not change; however, the material does, as well as going to the orthogrid as mentioned above. Figure 100 shows the changes in the fuel tank. Figure 101 depicts the intertank. Figure 102 shows the lox tank changes. The critical design challenges for SLWT have been: (1) welding and weld repairs, (2) achieving a proper fracture toughness, and (3) ratios of ultimate to yield materials properties. All these will be discussed.

SLWT Changes to LH₂ Tank



- Dome Cap and 11 of 12 Gore Panels
- Substitute Al 2195 for Al 2219
 - Resize membrane thickness and weld land width
 - One Gore Panel will remain Al 2219

- Major Ring Frames at Sta. 1129, 1871, 2058**
- Thickness reduction for Al 2024 webs
 - Sta. 1129 outer chord substitute Al 2195 for Al 2219

TPS

- Minimize excess SOFI thickness

Barrel Panels

- Substitute Al 2195 for Al 2219 and replace T-stiffened stringer panels with orthogrid pattern
- Eliminate seven (7) Z-Frames

Ring Frames Sta. 1377 and Sta. 1623

- Substitute Al 2195 for Al 2219

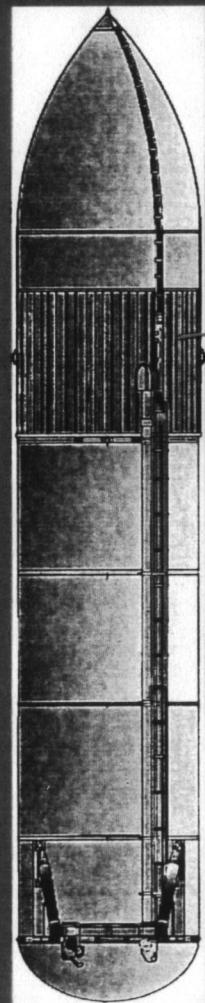
Forward Dome Gore Panels

- Substitute Al 2195 for Al 2219 and resize membrane thickness

- [Solid white square] = Al Li 2090, 2195
- [Hatched square] = Other Redesigned Parts
- [White square with black border] = No Change

Figure 100. Fuel tank changes for SLWT.

SLWT Changes to Intertank



Skin/Stringer Panel

- Substitute Al 2090 for Al 2024 and Al 7075 on skin panels, doublers, butt straps and sheet metal stringers
- Resize thickness where possible

Thrust Panel

- Resize thicknesses of Al 2219 plate

Primer

- Remove primer from detail parts and anodize
- Prime exterior only

Thermal Protection System

- Spray and machine TPS thickness to tighter tolerances

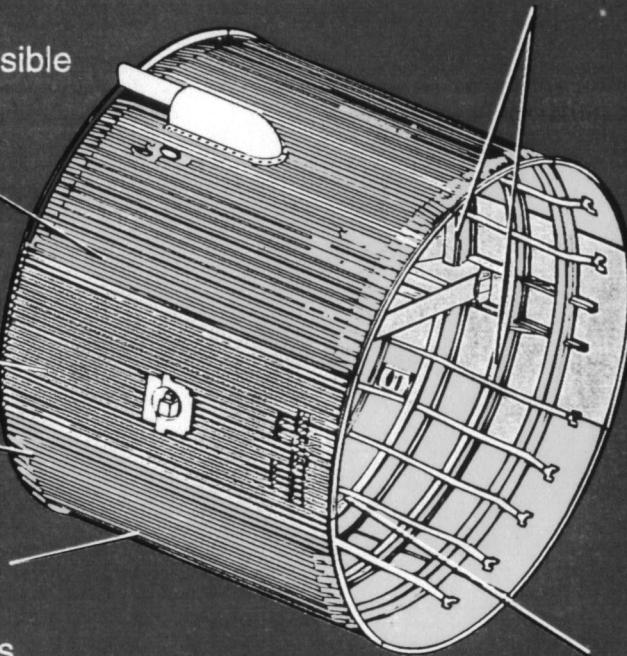
■ = Al Li 2090, 2195

■ = Other Redesigned Parts

■ = No Change

Main and Intermediate Ring Frames

- Reduce thickness of Al 7075 beaded webs
 - Substitute Al 2090 for selected Al 7075 flat webs, chords and stiffeners
 - Resize thickness where possible

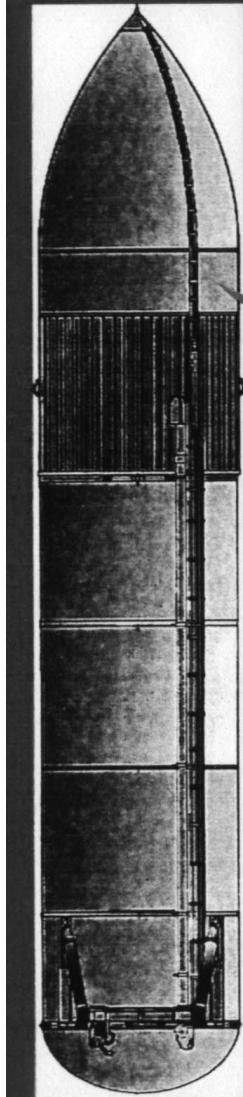


Range Safety System (RSS)

- Delete RSS panel and mounting structure
- Relocate RSS components to SRB Beam

Figure 101. ET intertank.

SLWT Changes to LO₂ Tank



Aft Dome Cap and Gore Panels

- Substitute Al 2195 for Al 2219
- Resize membrane thickness and weld land width

Slosh Baffle Outer Chord

Remove one (1) baffle tray and resize remaining Al 2024 webs

Barrel Panels

- Substitute Al 2195 for Al 2219 and resize membrane thickness and weld land width

Sta. 745 T-Ring Outer Chord

- Substitute Al 2195 for Al 2219

Sta. 852 Outer Chord

- Substitute Al 2195 for Al 2219

Forward and Aft Ogive Gores

- Substitute Al 2195 for Al 2219 and resize membrane thickness

TPS

- Minimize SOFI thickness on tank sidewalls

= Al Li 2090, 2195

= Other Redesigned Parts

= No Change

Figure 102. ET SLWT lox tank changes.

As stated previously, the tank is a prime part of the liquid propulsion system in that it contains level sensors, the pressurization system, etc., as shown on figure 103. It also contains the tumble valve for ensuring that the ET tumbles and breaks up as it enters the atmosphere and that resulting pieces stay within a safe footprint in the Atlantic Ocean and do not impact land.

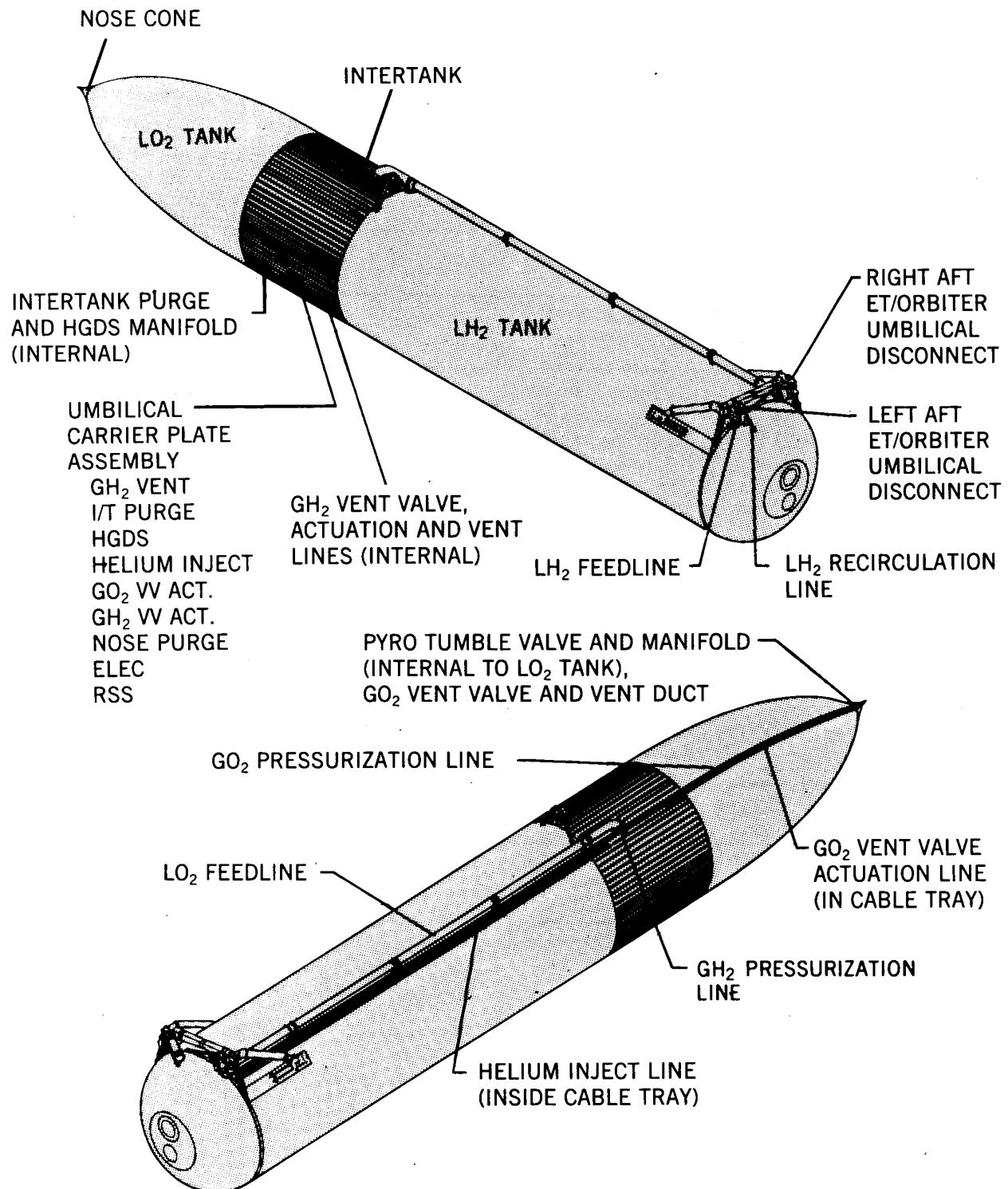


Figure 103. ET pressurization system.

b. Evolution/Innovation. Many innovative processes, as well as design concepts, had to be developed in order to meet the stringent performance goals without compromising safety since the space shuttle is a manned vehicle. The first of these was the fracture control plan and its implementation, which is required to ensure that no failure (tank rupture) would occur during flight. There are several aspects to this plan. (1) NDE techniques had to be verified for this application and expressed as a typical probability of detection (POD). The welds are dye-penetration inspected and x-rayed. (2) Each tank is proof-tested at ambient, which not only screens for critical flaws in the structure, guaranteeing a flight without failure with a life factor of four, but also tests the structure to near ultimate when the cyro properties are taken into account. (3) The tank is designed for leak-before-burst for nearly all welds. Where it is not, or where it is not properly proofed, the rewelds are x-rayed again. (4) Substantial analog testing of each skin thickness with welds, where a flaw is grown by fatigue to critical flaw size, then is tested through mission profiles until failure. At least a factor of four must be obtained for the lowest of the family tested. (5) A Government and contractor fracture control board, composed of stress, fatigue, materials, and NDE experts, reviews each reported flaw for flight safety. (6) All flaws found during manufacturing (x ray and dye penetration) are repaired before final assembly and proof testing.

Probably the most innovative design feature of the LWT was the reduction of the standard safety factor from 1.4 to as low as 1.25. The reduced factor was based on the degree of knowledge of the loads, i.e., longitudinal acceleration and internal pressure loads. If the load included all these loads, then a factor of 1.25 was used. If the load was mixed with 90 percent well known and the other 10 percent not well known, then a factor of 1.29 would be used. This approach, along with milling the skin to approximate the stress field, increased vehicle performance by 10,000 lb.

Robotics automation of spraying the insulation on the tank produces a more uniform coverage and less potential for debonding. The same uniformity has been achieved for much of the average welding using robotics.

c. Problems. A tank problem matrix that lists many of the problems experienced is given in the appendix; however, a few of the major and typical ones are discussed below.

(1) One problem was the aft hydrogen tank rear section oil canning during dynamic testing. During full-scale dynamic testing of the all-up shuttle vehicle to verify the dynamic modes calculation, the rear of the hydrogen tank started oil canning as the SRB's rolled against the struts that connected them with the tank. The cause, when understood, was quite benign and would cause no problem. Prior to filling the tanks with cryo fuel, the overall tank is several inches longer than it is when loaded with cryo and it shrinks due to the lower temperature. In order to compensate for this shrinkage, the SRB struts are canted 7° so that when the tank cyro shrinks, the struts are perpendicular to each element. This 7° angle was enough to roll the large ring frame in the aft of the tank. When perpendicular, the rolling does not take place.

(2) A problem related to the above is the pinch load introduced by the cyro shrinkage radially and longitudinally putting a large load into the tank structure. Early in the program, the struts were pre-tensioned to reduce and balance this load. Later, the tank was designed to handle the load. The thermal deformations are stored in the structure as potential energy, which is released as a part of the lift-off dynamics.

(3) To save cost and handling, it was decided to replace the tank nose cone made of Al with a composite nose cone. Two problems occurred with this design. The first was caused by a lack of understanding of the coefficient of expansion difference between the composite and the Al to which it was to be mated. The loads introduced would fail the nose cone when propellant was loaded. The second had to do with mating the holes with the top of the lox tank. In both cases, it was a breakdown in systems focus, where each discipline did its own thing without communicating with the others.

(4) A very interesting problem occurred during dynamic testing of the lox tank with various levels of propellant fill. When the tank was full to two-thirds full, the damping of the dynamic modes

was basically zero; however, for less full tanks the damping was above $1/2$ percent, which is typical for this type structure. Figure 104 shows the observed damping versus fill level. The cause was a tuning between the fluid and structures modes—a true hydroelastic effect. This modal damping was used in the control/sloshing analysis that sized the antislosh baffles.

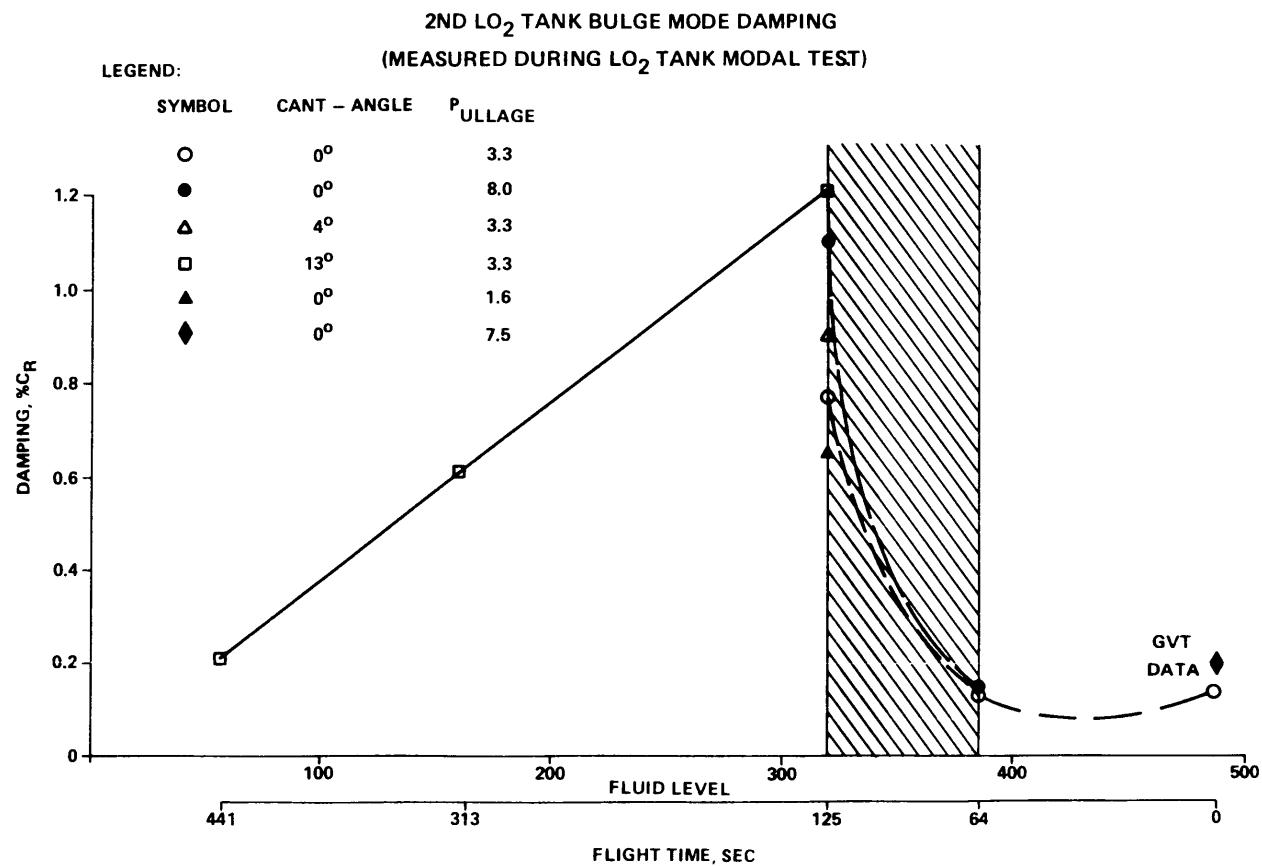


Figure 104. ET hydroelastic mode damping.

(5) Another problem was buckling. Development of propellant tanks has generally had to deal with buckling instability. Buckling is a very difficult problem to analyze since small eccentricities greatly magnify the instability in a nonlinear manner. The ET development, although keenly aware of the buckling potential, had a buckling occur during testing. This was due to the tank having to trim as much weight as possible to gain shuttle performance (1 lb of tank equals approximately 1 lb of payload to orbit). The first occurred during the mated ground vibration test program (MGVT). It occurred while draining the lox tank (water) in order to simulate a flight condition (fig. 105). Operational procedures have been changed so that this buckling is never experienced on the pad during launch operations. Figure 106 is a sketch of the buckled area.

(6) Two major problems have occurred on the SLWT during development: welding and weld repairs, and achieving the fracture toughness desired for fracture mechanics safety. Special techniques have had to be implemented for the welding and weld repair processes that include not only speed and proper forms and clamping, but also the use of backside purging, weld sequencing, and weld planishing to reduce residual stresses. Planishing can, however, be a problem in that it can also cause the weld to crack. Today, the fracture toughness is achieved by a very strict process control that deals with the material's chemical makeup and the heat treating, along with a lot acceptance of each ingot/panel. All panels that do not meet fracture toughness requirements are discarded.

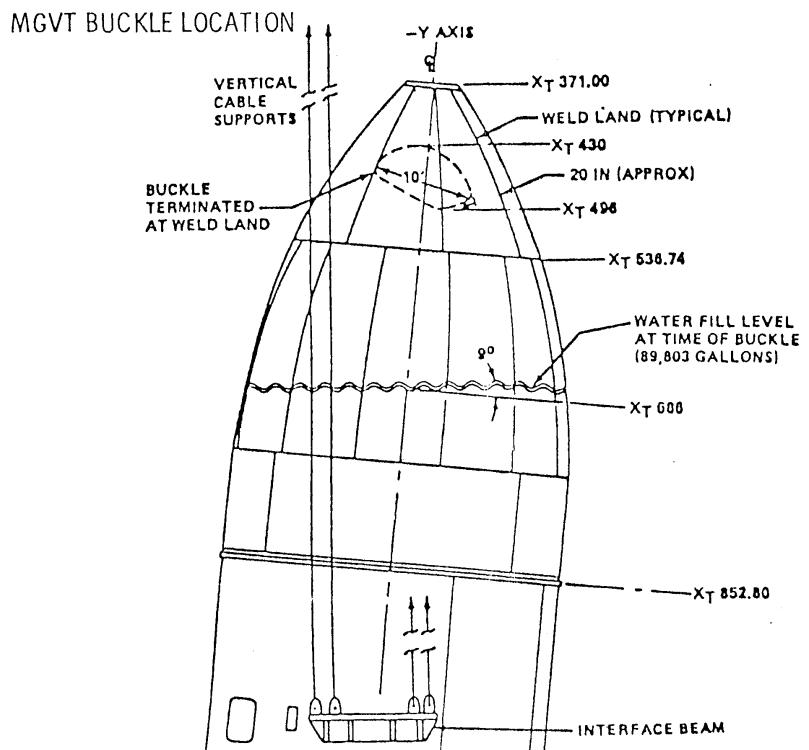


Figure 105. ET MGVT lox tank buckling.

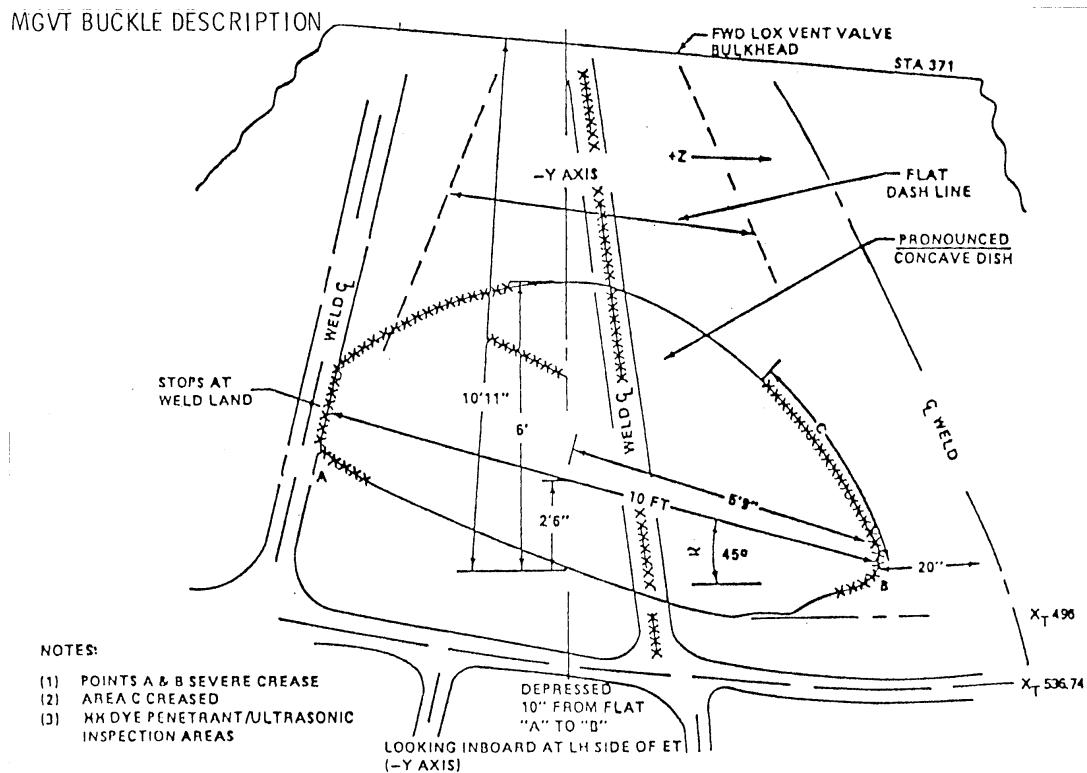


Figure 106. ET MGVT lox tank buckle description.

(7) SLWT Verification Program. Since the basic difference between the LWT and the SLWT was a material substitution with no changes to the basic components, it was felt by the technical community that a shortened test verification program could be achieved. Particularly, it was highly desirable that an all-up cyro tank test to failure be eliminated due to schedule impacts and excessive costs. The elimination of the cyro test program was challenged by two oversight committees, one of NASA origin, the other of Martin Corp. sources. In order to deal with the issue, a blue ribbon team of experts from the various NASA Centers and the Martin Corp. were formed to study the challenge and recommend to the program a cost-effective safe program to proceed with.

The team performed an indepth review of the proposed SLWT test program and compared it with the standard weight and LWT programs. In conjunction with this verification plan, a comparison of all tank panels and parts was made of the thickness, milling, etc., for each in order to determine where prior testing was applicable. Figure 107 shows a comparison of these three test programs (the SLWT program is the one available at the time of the teams formation). The team then established a modified SLWT test verification program. The SLWT structural verification team developed the following philosophies that it used in forming the verification program.

- Verify by test, for each structural element, the integrity of the structure
- Test demonstrate structure will withstand ultimate loads; or test demonstrate structure will withstand limit load and validate analysis accuracy and conservatism used to extrapolate to ultimate load
- Test can be omitted if the factor of safety (FS) > 2.0

Structural Verification ET Structural Test Evolution

SWT	LWT	SLWT
<ul style="list-style-type: none"> • Major Development Tests <ul style="list-style-type: none"> – 10% scale slosh – Forward and aft SRB attach fittings • Subassembly Strength Tests <ul style="list-style-type: none"> – ET/Orbiter interface hardware – LO² slosh baffle • STA Program <ul style="list-style-type: none"> – Intertank static strength – LO² static strength – LH² static strength – LO² Modal survey • Ground Vibration Test Program <ul style="list-style-type: none"> – Full scale ET • Components Qualification <ul style="list-style-type: none"> – Static strength – Vibration • Secondary Structure Verification <ul style="list-style-type: none"> – Static strength and capability – Vibration • Proof Tests <ul style="list-style-type: none"> – LO² tank – LH² tank – Propulsion lines 	<ul style="list-style-type: none"> • Development/Verification Tests <ul style="list-style-type: none"> – LH² skin stringer panels – 2058 frame stability • Interface Hardware Bench Tests <ul style="list-style-type: none"> – Forward and aft SRB fittings – Forward and aft SET/Orbiter hardware • LWT-1 Influence Coefficient Test <ul style="list-style-type: none"> – 2058 frame Stiffness • LWT-2 Limit Load Test <ul style="list-style-type: none"> – Aft LH² barrel and frame • New/Changed Components Qualification <ul style="list-style-type: none"> – Static strength – Vibration • Secondary Structure <ul style="list-style-type: none"> – Static strength and capability • Proof Tests <ul style="list-style-type: none"> – LO² tests – LH² tank – Propulsion lines 	<ul style="list-style-type: none"> • Element Component Tests <ul style="list-style-type: none"> – Orthogrid panels – Frame webs • ALTA Verification Tests <ul style="list-style-type: none"> – Flight equivalent loads – Flight configuration barrel – LH² orthogrid panel stability – Proof test • SLWT-1 LH² Limit Load Test <ul style="list-style-type: none"> – Similar to LWT-2 test – Influence coefficient test – Barrel panel stability – Aft dome pinch load stability • New/Changed Components Qualification <ul style="list-style-type: none"> – Static strength – Vibration • Secondary Structure <ul style="list-style-type: none"> – Static strength and capability • Proof Tests <ul style="list-style-type: none"> – LO² tests – LH² tank – Propulsion lines

Figure 107. ET structural test evolution.

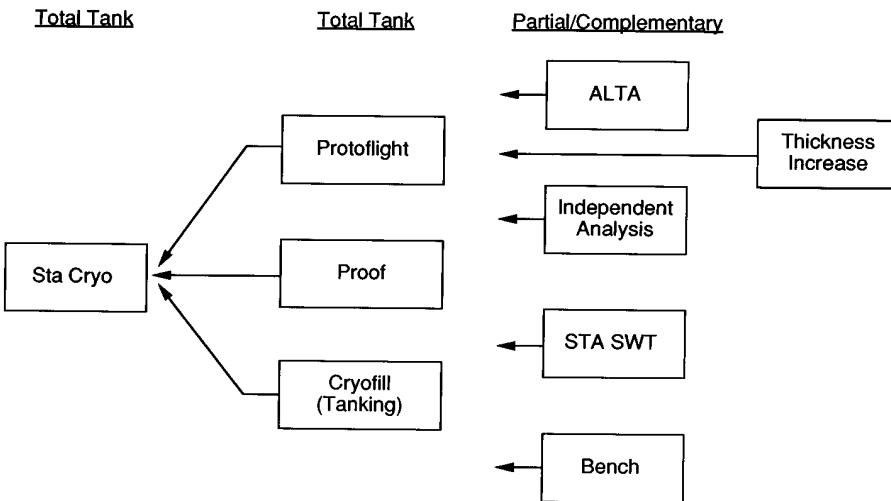
- Test not required if similar, more critical, structural element has been tested or LWT flight verified
- Test article will be built on production tooling with production processes
- Test article will be fabricated from material acceptable for production hardware
- Test completion is precursor to flight
- Cyro effects will be verified by material coupon testing, data gathered from the SWT testing, analysis, and Kennedy Space Center (KSC) tanking test.

Implementation of this philosophy is accomplished by a variety of methods that include:

- Membrane thickness increase—SLWT thickness is maintained at flight-verified LWT thickness in select stability critical regions.
- Independent Analysis—MSFC is performing independent analysis of the intertank, and Langley Research Center (LaRC) is performing independent analysis of the lox tank; provides increased confidence in design adequacy, particularly in areas where factor of safety >2.0 criteria are used.
- SWT Structural Test Article (SWT STA)—The SWT STA demonstrated loads with cryogenic temperature and temperature transients when loading; data and experience if being built upon.
- Component Test—A series of 12 groups of components tests being performed to demonstrate or verify substructures; examples include hydrogen tank orthogrid panel compression tests, intertank skin-stringer/joint compression tests, stress concentration tests, and intertank crossbeam web tests.
- Aluminum Lithium Test Article (ALTA)—A structural test article consisting of a 20-in barrel representative of barrels 3 and 4, Z panels of barrels 1 and 2 of the hydrogen tank, a hydrogen tank aft dome, and a lox tank aft dome; demonstrates orthogrid panel stability to ultimate and capability loads, stability of lox tank aft dome, and 113 percent of limit load in tension for forward hydrogen tank barrels and aft dome.
- Proof Test—Performed on each hydrogen and lox tank; demonstrates strength of each tank pressure wall to 113 percent or greater of limit load.
- Prototypical Test—Performed on each hydrogen tank; demonstrates stability of tank barrels 1 and 2 in the vicinity of the longeron and aft dome stability to 115 percent of limit load; demonstrates strength of longeron region including welds of adjacent panels.
- Tanking Tests—Demonstrates prelaunch conditions to limit load.

Figure 108 shows this in a flow block diagram format as used by the team to assess the risks associated with the approach. The innovative approach added to the program twofold: first, the hydrogen tank proof test was augmented to include prototypical test at 115-percent external loads to verify stability; second, the increase in material thickness to match LWT for buckling. In addition, LaRC and MSFC are conducting independent buckling analysis. Figure 109 is a matrix of the total plan that ensures a safe tank for flight. The concern over no cryo tank test was handled by conducting a series of cryo coupon tests and extrapolating the data to flight conditions. This is possible since the basic material characterization change is in strength and modulus, thus the resulting thermal-induced stresses and the basic tank configuration did not change and were shown to be low.

Verification Risk Assessment Approach



- Assess risk of not performing complete STA tests for individual parts
- Assess risk of combined element and total tank tests versus STA tests

Figure 108. ET test verification flow.

Neil Otte

Super LightWeight Tank (SLWT) Structural Verification

SLWT Lead Engineer
Structures and Dynamics Lab

Verify the integrity of each structural element by test.

- Demonstrate the structure to ultimate load, or alternatively, demonstrate the structure to limit load and validate the analysis accuracy and conservatism to extrapolate to ultimate load.
- Test is not required if similar, more critical hardware has been test or LightWeight Tank (LWT) verified.
- Test can be omitted if Factor of Safety ≥ 2.0 .

	Maintain SLWT or LWT Thickness	Independent Analysis	Component Test	ALTA	Proof Test	Protolfight Test	Tanking Test
LO2 Tank							
Stability	X	X		X			
Strength					X		
Intertank							
Stability		X	X				
Strength	X		X				
LH2 Tank							
Stability				X		X	
Strength					X	X	
TPS			X				X
Interface Hardware	Unchanged						

An aggressive, test based, verification program has been established and is being executed for the SLWT program. The program builds upon the existing test base of the External Tank Structural Test Articles. Testing completed to date has been successful.

A challenging verification program lies ahead of us.

Successful completion of the program established will assure a SLWT which is safe to fly.

Figure 109. ET test matrix.

In summary, the SLWT has an aggressive test-based verification program that is being executed and includes:

- Proof test for strength of the lox and hydrogen tanks
- ALTA and protoflight testing of each flight tank for stability
- ALTA thickness increases and independent analysis for stability of the lox tank
- Component tests and independent analysis for stability of the intertank
- Component tests for frames and substructures.

Successful completion of the program established will assure an SLWT that is safe to fly.

(8) Insulation debond on the nose of the lox tank was a major problem early in the program. It was believed to be a debris source that damaged the frail orbiter reentry heat protection tiles. This problem was caused by the thermal gradients between the TPS and the substrate that was being driven thermally by the cryo lox temperatures. Through proper layering of materials and better bonding, the problem was solved. It serves as a good example of how an insignificant problem on one element can lead to a serious problem on another element and to the system.

G. Hubble Space Telescope

1. Characteristic. The Hubble Space Telescope (HST) is the world's largest and greatest orbiting optical telescope.⁶⁸ Its accuracy and precision are unprecedented. Even with the problem of aberration of the mirror, which has been partially corrected, its contribution to science and our understanding of the universe has grown by leaps and bounds. The pointing accuracy of 0.007 arc s, which is equivalent to focusing on a dime from Washington, DC, to Boston, is unequaled in the optics world. The HST (fig. 110) is composed of the outer protective shell (called the SSM), the momentum wheels that control the vehicle by varying each wheel's speed, and the aperture door for blocking direct sunlight from the instruments, allowing more science time. The inner parts are the heart of the optical telescope assembly (OTA) system composed of the metering truss mounting, the primary and secondary mirrors, the focal plane structure, and the aft truss mounting the scientific instruments. Figure 111 shows an exploded view of the OTA with the scientific instruments. To eliminate thermal distortion in one direction, the OTA frames are built of composites that have near-zero coefficient of thermal expansion (CTE) in the longitudinal direction. The mirrors must be accurately located relative to each other driving this CTE requirement. The light enters striking the primary mirror, which focuses it on the secondary mirror. The secondary mirror focuses the light back through a hole in the center of the primary mirror onto the scientific instruments. The instruments are: (1) the fine guidance sensor, (2) the faint object spectrograph, (3) the wide field planetary camera, (4) the photometer, (5) the high-resolution spectrograph, and (6) the faint object camera. They function to perform the fine pointing and to gather various scientific data for studying the universe (fig. 112). Power is provided by solar arrays in conjunction with storage batteries. The control system uses six rate gyros and two fine guidance optical sensors to provide roll, pitch, and yaw information, and is designed to the observatory, accurately locked to within the 0.007 arc s on a subject for extended periods. This is equivalent to a 1.22-cm offset at a distance of 600 mi (1,000 km). Temperatures within the telescope are controlled actively and passively to assure pointing accuracy and structural stability. The system was designed for maintenance and refurbishment on-orbit at approximately 5-year intervals. The first of these repair missions has already occurred as a result of the aberration problem. During that mission, the solar arrays, batteries, rate gyros, and fine guidance sensor were replaced along with the aberration correction lens.



Figure 110. HST characteristics.

2. Evolution/Creativity/Innovation. The prime technology driver for HST was the very high pointing accuracy, which required several breakthroughs and developments: (1) development of the pointing control logic and mechanism, including the fine pointing instrument (sensor), momentum wheels, and vibration (modal) suppression; (2) development of low-force isolation techniques for mechanical systems such as the momentum wheels (verification of these isolation systems required innovative testing and data evaluation techniques); (3) development of composite materials that would have near zero CTE in the longitudinal direction (you do not get something for nothing, so the low expansion in one direction created deflections in the other directions, creating a major joint design effort); (4) understanding and verification of the modal characteristics for both pointing while on orbit and accurate loads predictions during launch and transportation; (5) accurate and efficient thermal control systems to protect the sensitive instruments and maintain pointing accuracy; (6) long-life mechanical and electrical systems; (7) solar energy system to provide power to operate the system on orbit; and (8) on-orbit servicing.

The solar arrays, supplied by the European Space Agency (ESA), created impulses to the system when going through the day/night and night/day terminator due to thermal snap. Interaction with the fine pointing system was many times more than would have been for the case for healthy solar arrays. Reference 67 discusses this problem and its solution with the new arrays that were installed during the repair mission. Flight data served to identify the troublesome modes, and a new

SPACE TELESCOPE CONFIGURATION KEY DESIGN FEATURES

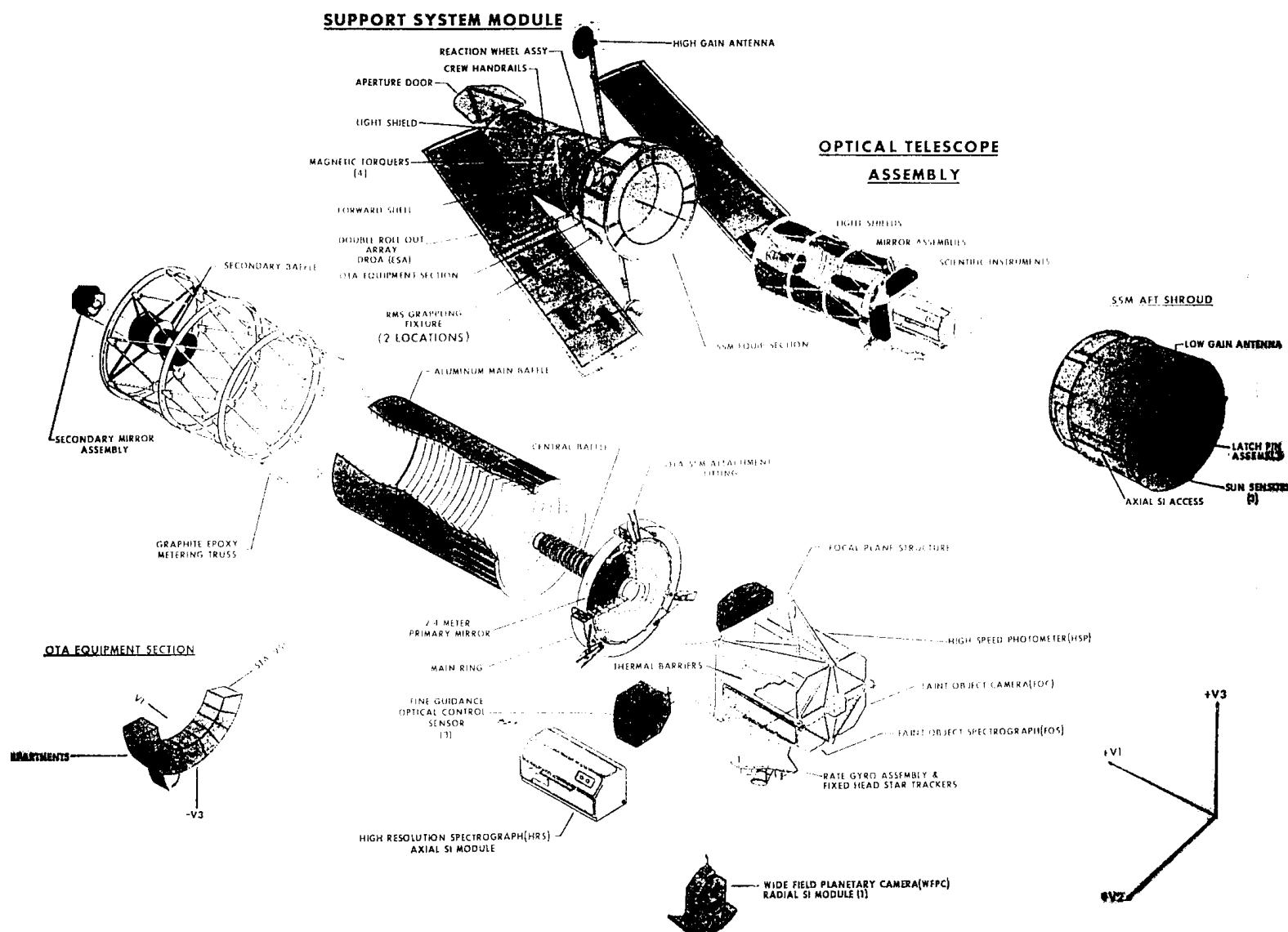


Figure 111. HST exploded view.

HUBBLE SPACE TELESCOPE CONFIGURATION

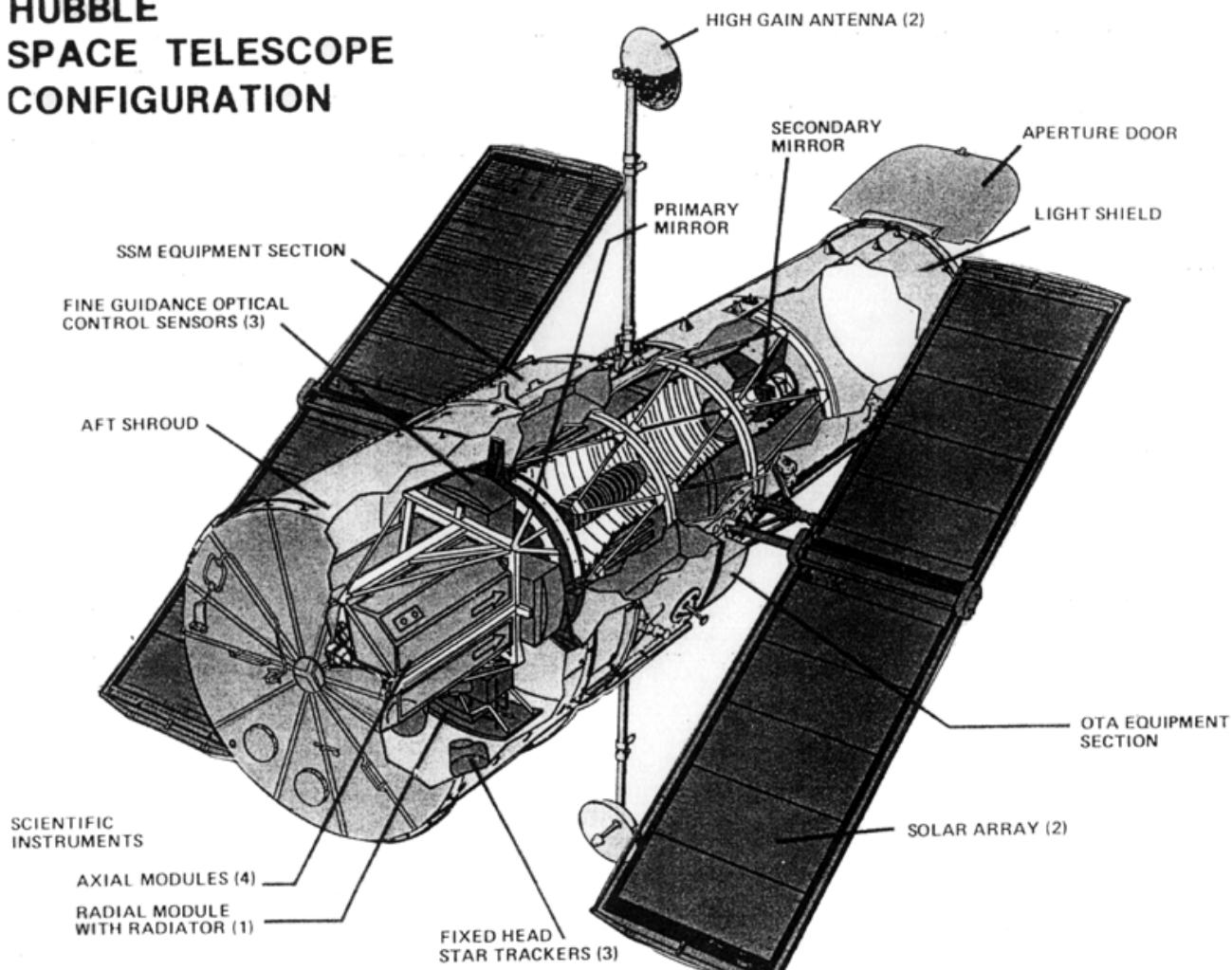


Figure 112. HST solar arrays.

control law/stabilization filter was designed. The up-linked control changes enabled NASA to meet the HST pointing specifications during the 2 years of operation before the repair mission. Over 20 structural modes were suppressed with the new control law. This is the largest number of modes suppressed with one control law and control system to date. The next section dealing with problems will discuss further the items identified and the innovations applied.

Probably one of the great innovative achievements of the Hubble program was the repair mission, which not only required creativity for the mirror aberration solution, but required the best that man could accomplished in space EVA's—first to capture the Hubble, then to repair it. The national coverage and interest in this mission alluded to its complexity and accomplishment.

Another creative achievement occurred late in the ground-verification portion of the program when it was decided that the free-free modal test performed on part of the system, which adequately verified the on-orbit configuration, had not verified the modal characteristics for the transportation configuration when Hubble was attached to the orbiter payload bay. To test the total final assembled telescope in the constrained mode would be at least very risky and very costly to the program. This problem was solved by some very innovative workers,⁶⁷ which showed that a low-level impedance test of the orbiter telescope attach points with the final Hubble configuration would adequately provide the missing

data. The technique was demonstrated on simulated hardware, then the Hubble was successfully tested. The data were excellent. The approach was not only a great cost saving (the all-up test cost \$10 M, impedance test cost less than \$1M) but greatly reduced the risk of damage to the flight hardware.

3. Problems

a. Mirror Aberration. The largest impact problem that occurred on HST was the mirror aberration that was caused by improper grind measurement tools that created a small error on the outer portions of the mirror. Figure 113 shows the basic problem and the results it created. The problem did not degrade all the science. The frequency spectrographs were not affected. A problem of this type shows what negative publicity can do to an organization. To fix the problem, an innovative lens was developed that corrected the aberration with a small loss of power. This design was probably the major innovative device for Hubble. The repair mission was also one of the most exciting, with national interest and television coverage. Much training in the Neutral Buoyancy Simulator tank was required for the astronauts in order to achieve overwhelming success. Figure 114 shows images taken before and after the lens correction and shows how well the fix worked.

The "best" focus for a beam containing spherical aberration lies somewhere between the marginal and paraxial focus.

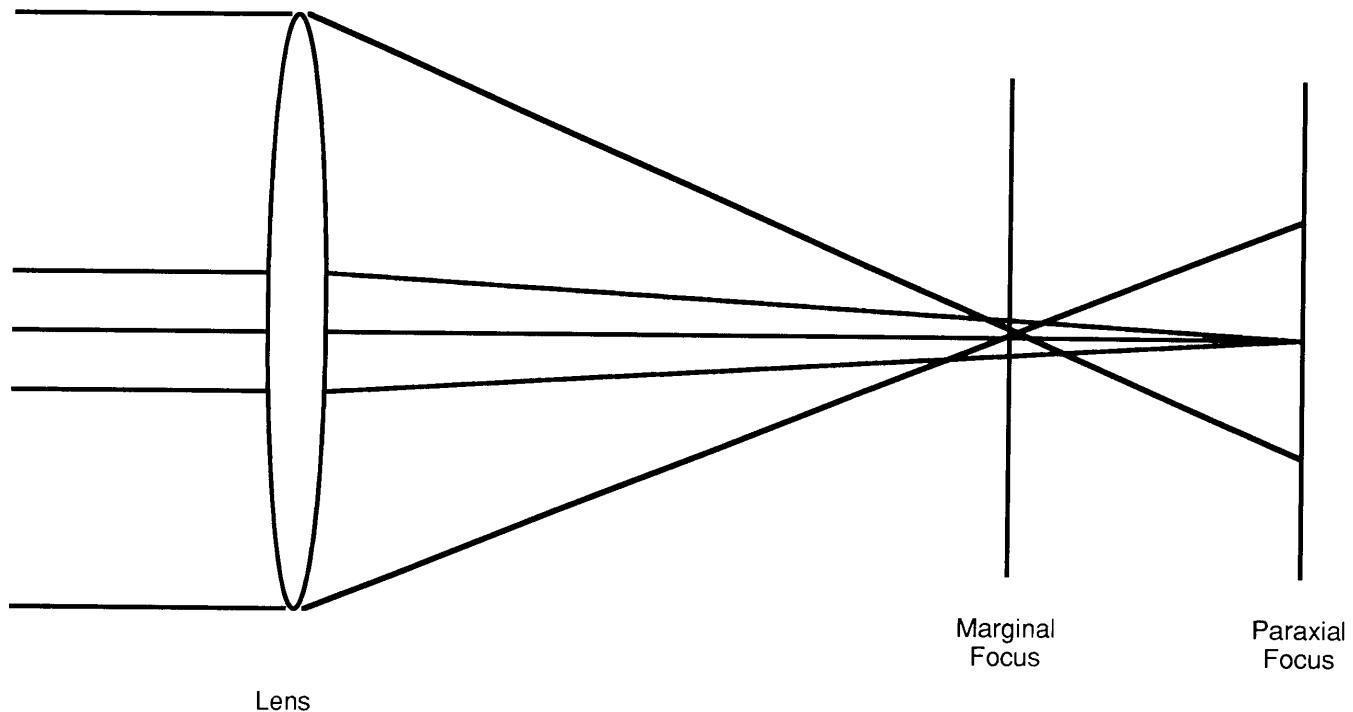


Figure 113. HST spherical aberration.

b. Solar Terminator Excitation. After discovering that Hubble had an aberration problem, it was mandatory that as much science as possible be salvaged until a repair mission could take place. In the process of tuning up the pointing control system, it was clear that every time Hubble moved from day to night, or night to day, a large disturbance occurred, upsetting the pointing system and disturbing the observations.

Nova Cygni 1992
Hubble Space Telescope
Faint Object Camera

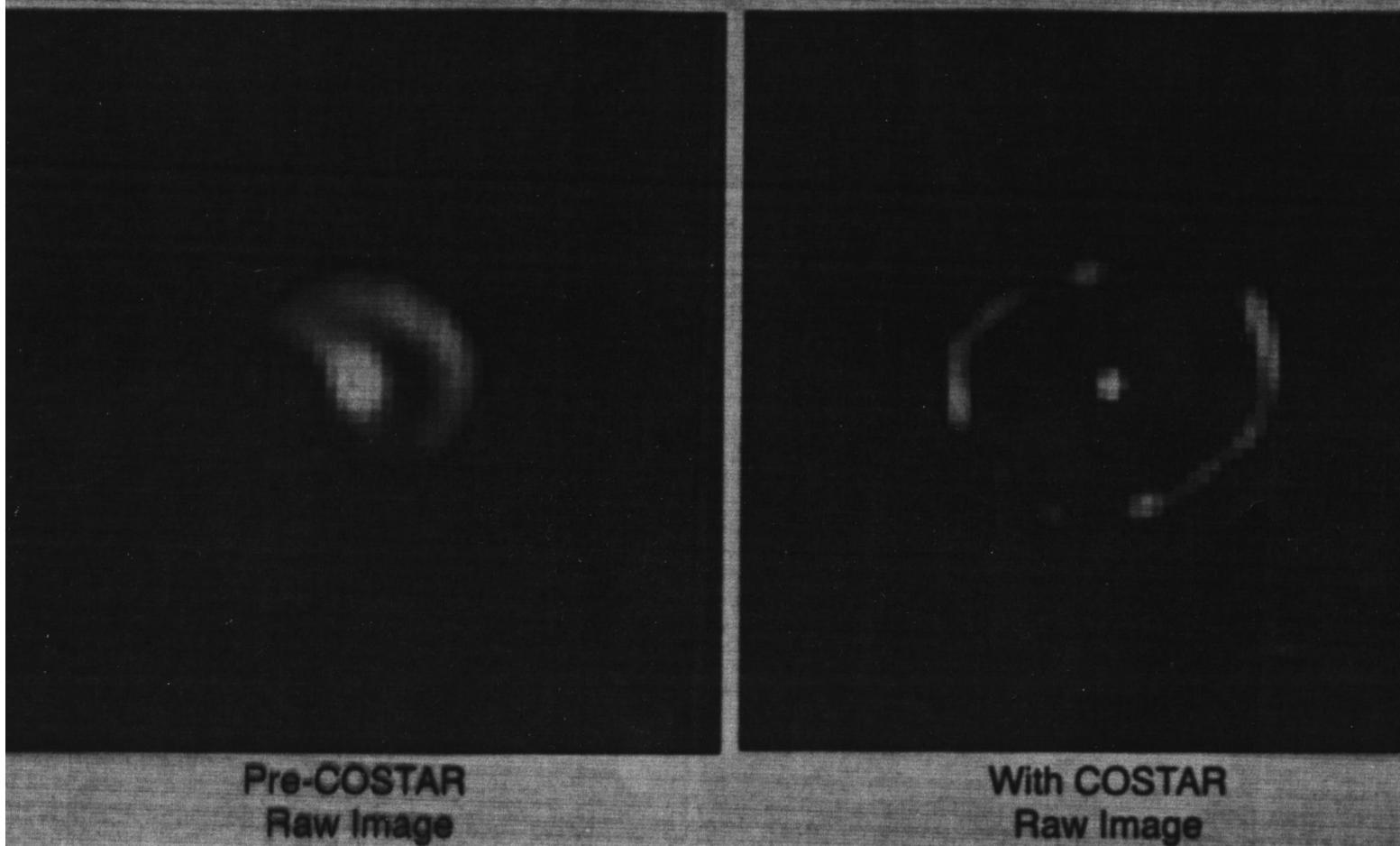


Figure 114. HST picture before and after repair mission.

Two parallel efforts were carried out to understand and solve this response problem. First, detailed simulations of the spacecraft with the pointing control system were conducted along with data evaluation of the flight data in order to develop a control logic software change that could be uplinked and that could solve this problem and save science.⁶⁷ Second, an effort was made through ground test and analysis to understand or isolate the disturbance cause. This information would also help design a fix to be carried up and installed during the repair mission.

The major problem was caused by the solar array bi-stem jerking due to thermal expansion. The thermal deflections of the bi-stems resulted in mechanical binding and slippage, which occurred as the telescope passed in and out of direct solar radiation. This sudden mechanical response induced torquing disturbances to the HST's pointing control system. The ESA, the solar array vendors, conducted thermal analysis and tests to address these deflections and the sudden mechanical response that had not been predicted.

Once the problem was identified, ESA first conducted detailed thermal analysis, which basically confirmed results found during the design phase. In addition, MSFC and Lockheed conducted parallel analysis with almost the exact results for the temperature predictions. Next, ESA conducted simulated solar radiation testing to evaluate the bi-stem response. Much work was also accomplished at the Marshall Center from the dynamics and control organizations, as well as the thermal analysts, to help understand the problem.⁶⁷ As a result, the understanding of the mechanical response was achieved and ESA designed a replacement solar array that would solve the problem. The faulty arrays were replaced during the repair mission with this new array. No further problem of this type has occurred in Hubble's mission.

The control personnel at MSFC were able to design a control logic that would dampen the oscillation, so that good science could be obtained while waiting for the repair mission to take place. The damping of this disturbance was achieved at a much greater degree than expected, and the software was data-linked up to Hubble, saving the science until the repair mission.

There are many lessons in terms of understanding multidisciplinary interactions as well as testing to the extent possible on ground. The designer should also pay much attention to the design of bi-stem deployable rods, which are necessary (or some variation thereof) for collapsing arrays for transportation.

c. Designing for Low CTE and Other Design Issues. Highly accurate large space optical systems are very sensitive to distortions and misalignments. One of the main sources of these disturbances is the thermal expansions and contractions of the structure in the severe space environment. The approach taken for Hubble was to build the basic truss structure out of composites where the CTE could be made very low in the critical directions. This does not come for free, and many problems resulted. Carl Loy (retired), one of the structural designer's at MSFC, provided a history of these problems. The following are some excerpts from notes he provided the author.

(1) Metering Structure Joints. The first metering truss built by Boeing had a major problem with the joints due to the laminate lay-ups transferring the loads for the low CTE direction into the other planes. If the laminate is oriented such that its plies are perpendicular to, and the laminate makes up a portion of a dimension that is to be controlled by near-zero CTE effects of the composite lay-up, the expansion/contraction of the laminate thickness will tend to overwhelm the low CTE of the structure, compromising the remainder of that dimension (fig. 115). This is because the CTE of the laminate through its thickness is controlled by the matrix material, i.e., the epoxy. The CTE of the graphite fibers perpendicular to the fiber length is also quite high. The solution to the problem was to remove the laminate thickness from the "line of action" in the dimension that is to be CTE controlled (fig. 116). The flight metering truss, the gusset plate that joins the strut in adjacent bays, passes through a slot in the web of the ring. A clip is used to tie the ring to the gusset to maintain the location of the components.

(2) Secondary Mirror Support Structure. The secondary mirror support structure under thermal deflections deformed the alignment between the primary and secondary mirrors. The metering structure

**Strut/Ring Joint Configuration in the
Boeing 3 Meter Metering Truss**

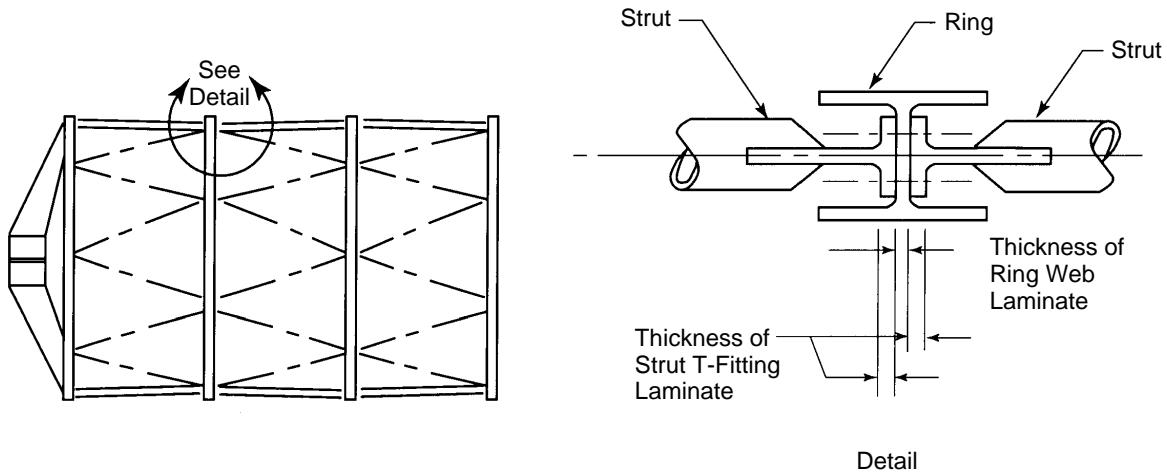


Figure 115. HST shelf.

**Strut/Ring Joint Configuration in the Boeing
2.4 Meter (Flight) Metering Truss**

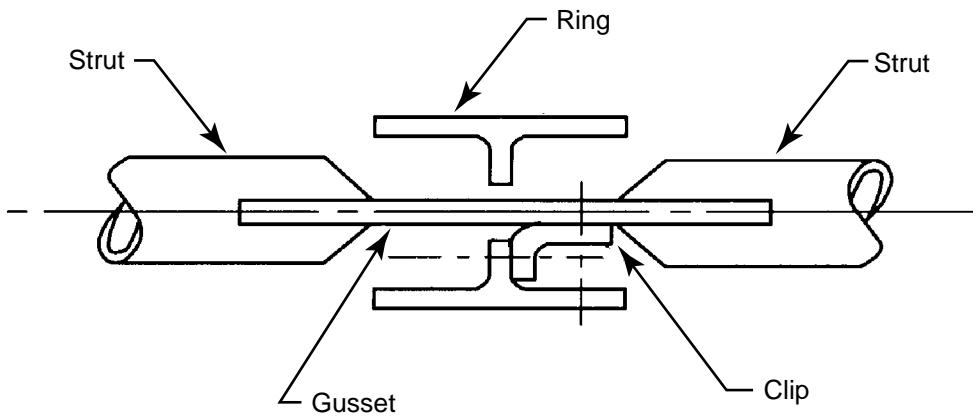


Figure 116. HST metering truss fix.

design with the secondary support structure was located forward of the forward ring of the metering structure (figs. 117 and 118). The primary reason for this structural arrangement was to save weight in the metering structure. However, it was observed that a thermal change resulting in a uniform contraction or expansion of the diameter of the forward ring of the metering structure would move the secondary mirror forward or back along the longitudinal axis of the metering structure. A thermal gradient across the diameter of the forward ring will, in addition to the pumping action mentioned above, also tilt the secondary mirror with respect to the longitudinal axis of the metering structure. Both of these effects were noticed during the tests on the Boeing 3-m metering truss and the General Dynamics half-scale metering shell. The thermal gradient was sufficient longitudinally. The solution to the problem was that the secondary mirror support structure was redesigned so that it was symmetrical, forward and aft, about the plane of the forward ring of the metering structure. That is, no bending or flexing of the support structure will occur due to the thermal expansions/contractions of the metering structure (fig. 119).

**Secondary Mirror Support Structural Arrangement
for the Boeing 3 Meter Metering Truss and the
GD/C Half-Scale Metering Shell**

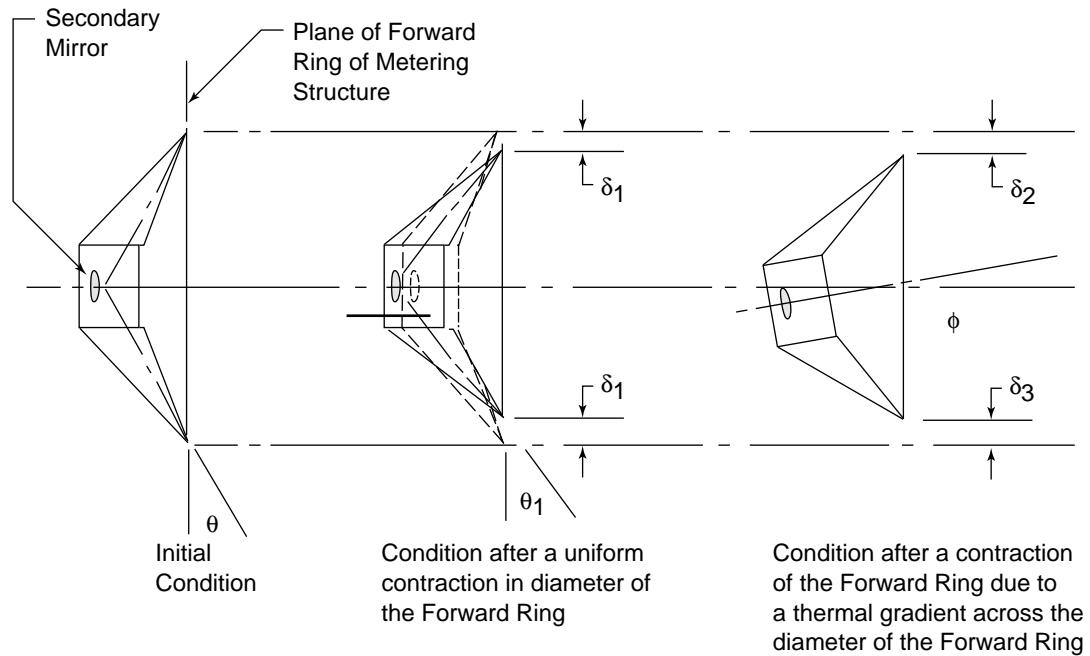


Figure 117. HST mirror deflection.

**Secondary Mirror Support Structural Arrangement
for the Boeing Flight Metering Truss & the Air Force
Modified GD/C Half-Scale Metering Shell**

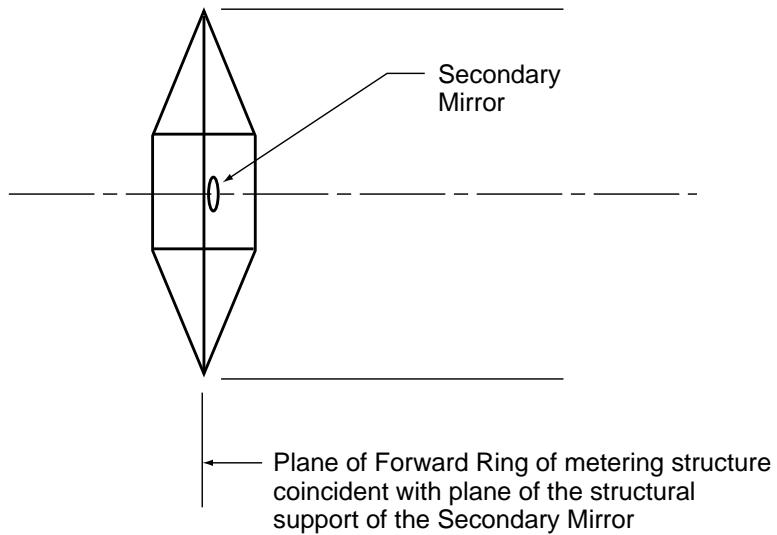


Figure 118. HST secondary mirror.

**Field Splice/Shell Configuration in the GD/C Half-Scale
Metering Shell as Modified by the Air Force**

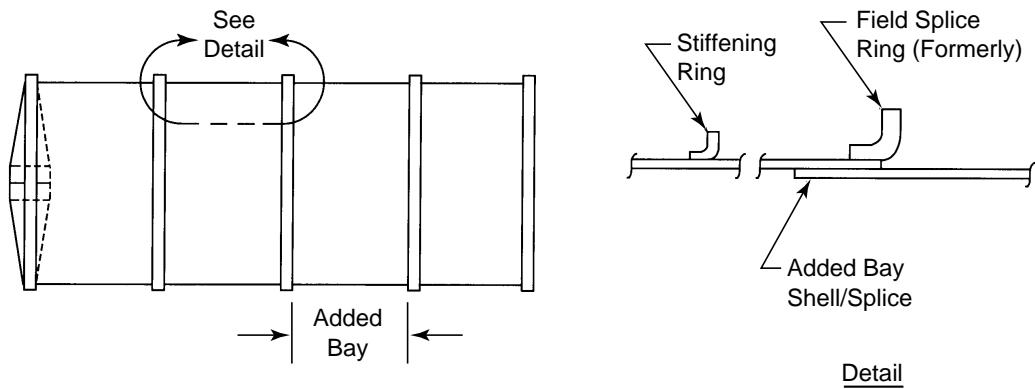


Figure 119. HST modified truss.

(3) "Bathtub" Fittings. The original design had "bathtub" fittings at the fine guidance sensor, and the radial scientific instruments (SI's) points B and C latch locations (the focal plane side of each of these points, i.e., the fixed half of the latch) were bolted to these "bathtub" fittings. All of the fittings failed (three were tested) well below the design load. The failures were all in the length of the edge of the fitting (fig. 120). The failure was due to the fact that the design did not place any graphite fibers in the corner areas, i.e., in the areas indicated by N on figure 120. In other words, there was no graphite reinforcement, but only epoxy, tying the base plate to the gusset plates on either side, where the failure occurred. These failures occurred at such a time in the program that it was deemed feasible not to try and remedy the design, but to replace the "bathtub" fittings with titanium-machined fittings having the same configuration as the composite fittings. This was a good trade since the reinforced composite fittings would weigh approximately the same as the titanium fittings.

**Configuration of "Bathtub" Fitting in
Flight Focal Plane Structure**

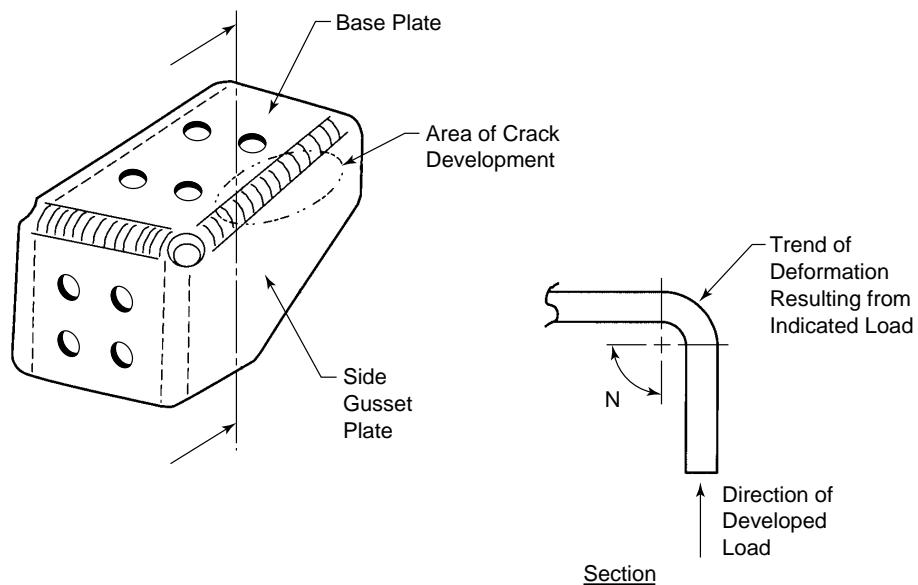


Figure 120. HST bathtub fitting configuration.

(4) SSM Aperture Door and Aft Shroud Bulkhead. The SSM aperture door and the aft shroud bulkhead structures were of aluminum honeycomb construction; both were flat and both were of large dimension in length and width. The thickness of the honeycomb core of the facings is typical, though the facings on the aperture door approach a thinness that requires very careful handling for the large sheets used.

Inspections of these two sandwich structures, after they were completed, indicated an area of debond—critically so in the aperture door—and several “dings” in both structures. The inspection reports, the manufacturing and bonding methods, and the processes and procedures were extensively reviewed by Marshall personnel experienced in all of the affected disciplines. Both of these structures were built in the Lockheed Burbank Aircraft facility, which has considerable experience in building aluminum honeycomb structures; however, the aperture door and aft bulkhead were larger than any honeycomb structure they had previously built. This lack of experience with that size honeycomb contributed to the poor quality of the finished product. Also, the bonding tool used probably contributed to the final quality of the product.

The solution to the problem was to analyze the aft bulkhead and accept it due to the smaller panels created by the internal panels reducing the size of the debonds. The aperture door was rejected as beyond repair, and a new one of much-improved quality was produced. The rejected door was used as a substitute for the flight door in the various test programs until time to ship the Hubble to the launch site.

d. Thermal Vacuum Testing (TVT). Thermal testing of such a large spacecraft with subsystems integrated from many different contractors was indeed a challenge. There was an abundance of knowledge and experience gained during the course of Hubble’s TVT testing. As with any TVT, the amount of instrumentation was limited and continued to be a concern until testing. The successful completion of Hubble’s TVT, followed by math model correlation to the measured data, proved that adequate instrumentation was provided and could serve as a valuable guideline for future testing of this magnitude. The Hubble’s TVT can also serve as a very good example of environment simulation using infrared (IR) lamp arrays. The Hubble program completed design, manufacture, checkout, and implementation of a huge IR lamp array accompanied with thorough documentation. The knowledge and documentation resulting from Hubble’s experience should have direct application to any new spacecraft system-level testing.

The Hubble program completed a rigorous evaluation to specify thermal stability criteria for each test phase of TVT. After lengthy debates, the thermal group successfully obtained agreement to conduct certain thermal balance test segments without interference from electrical reconfiguration of power level changes. These tests were critical to the follow-on correlation to math models. The stability criteria established had to be met before configuration to the next test segment was initiated. This experience indicates that thermal balance tests should be a requirement for thermal control system verification. This experience should also serve as a shortcut for future programs in specifying and justifying criteria.

During TVT development testing for Hubble’s SSE TCS, multilayer insulation (MLI) blanket performance did not meet expectations, which were based on previous testing of the other Hubble hardware. It was discovered that the outer layer material of aluminized TeflonTM, combined with the absence of a netting spacer, is not suitable for the blanket design of small components. The CTE differences between the aluminized layer and the TeflonTM substrata caused the outer layer to shrink at cold temperatures. This shrinkage would compress the blanket at the corners, creating thermal shorts and reducing performance. This prompted a more thorough evaluation test to determine relative MLI blanket performance for small components. As a result, the final blanket design for the Hubble’s SSE TCS used TedlarTM as the outer layer, and DacronTM netting as the sublayer spacer, which gave much better performance for such small MLI designs.

During Hubble’s thermal vacuum test, several failures of the thermal control system occurred. Posttest evaluation eventually traced the failures to faulty thermostats. The failures were specifically

attributed to improper dry gas backfill by the manufacturer. A plan was developed to replace all potentially affected thermostats. This plan included installation steps to revivify heater circuitry after replacement. This particular step was especially important in the absence of further environmental testing. Accurate documentation of this step is very important for future reference.

e. Summary. The Hubble project taught many valuable lessons that have been briefly described in the above paragraphs. Hubble was planned for orbital verification and mission support. Much effort was expended to have all technical experts trained, and to have ground-based simulation of the on-orbit spacecraft, etc., put in place for resolution of slight anomalies. One problem occurred during the initial orbiting and checkout phase—the determination of the spacecraft orientation. Eventually, an abbreviated method was developed to determine spacecraft orientation from telemetry data. Future spacecraft training sessions must have a system approach so that this type procedure would not have to be worked out in real-time.

Despite the various problems discussed here and the ones illustrated in the appendix, Hubble was a very innovative and specialized design, as the current science being collected proves. The ability to make telemetry software changes was a lifesaver and is a standard approach, but must be well thought and designed for. The same is true for the maintenance mission, which required not only the design for line replaceable units but consideration of the labor complexity involved for on-orbit repair in zero g and by astronauts in space suits. The success of the original mission phase, the repair mission, and the ongoing science mission after the repair gives evidence of design and program success. Figure 121 shows the repaired Hubble being deployed from the shuttle.

H. Spacelab

1. Characteristics. The Spacelab system was designed by the ESA to provide a flexible orbiting laboratory within the shuttle cargo bay for conducting various experiments and research in space.⁶⁹ It consists of different combinations of enclosed human working areas called modules (two lengths) and open V-shaped pallets, which expose experiments to space or act as launching/retrieval platforms that are exposed to open space when the shuttle cargo doors are open. The modules have a life support system, experiment racks with power, etc., with space in between for the astronauts to work. Figure 122 shows a basic module and figure 123 depicts a typical pallet. Figure 124 shows some of the basic combinations of modules and racks in the cargo bay. In the modules, the astronauts/scientists have essentially a lab to work in as they do on Earth. The pallet experiments can be controlled from the module or the orbiter crew observation deck.

Spacelab missions are cooperative efforts between scientists and engineers from around the world. Teams from NASA Centers, universities, private industry, Government agencies, and international space organizations design the experiments. Some scientists actually fly and run the experiments in space, with support from scientists and engineers on the ground who work and communicate with those on-orbit via voice, video, and data links. The reusable international vehicle allows them to bring the experiment samples back to Earth for further study.

Many missions have flown using Spacelab hardware in various combinations. The experiments covered the gamut from microgravity to stellar observation. For example, Spacelab flew on STS-9 in 1983, with 73 experiments. The laboratory module flew again on STS-51B, as Spacelab 3, emphasizing low-gravity materials processing. Spacelab 2, on STS-51F, was a pallet-only mission, studying solar science. Spacelab D-1, on STS-61A, was a German mission.

Spacelab became more specialized in the 1990's. Astro missions were dedicated to ultraviolet and x-ray astronomy. This included Atmospheric Laboratory for Application and Science (ATLAS), Spacelab Life Sciences (SLY), United States Microgravity Laboratory (USML), United States Microgravity Payload (USMP), International Microgravity Laboratory (IML), Spacelab J (Japanese Materials Processing), and Spacelab D-2 (a German reflight with upgraded experiments).

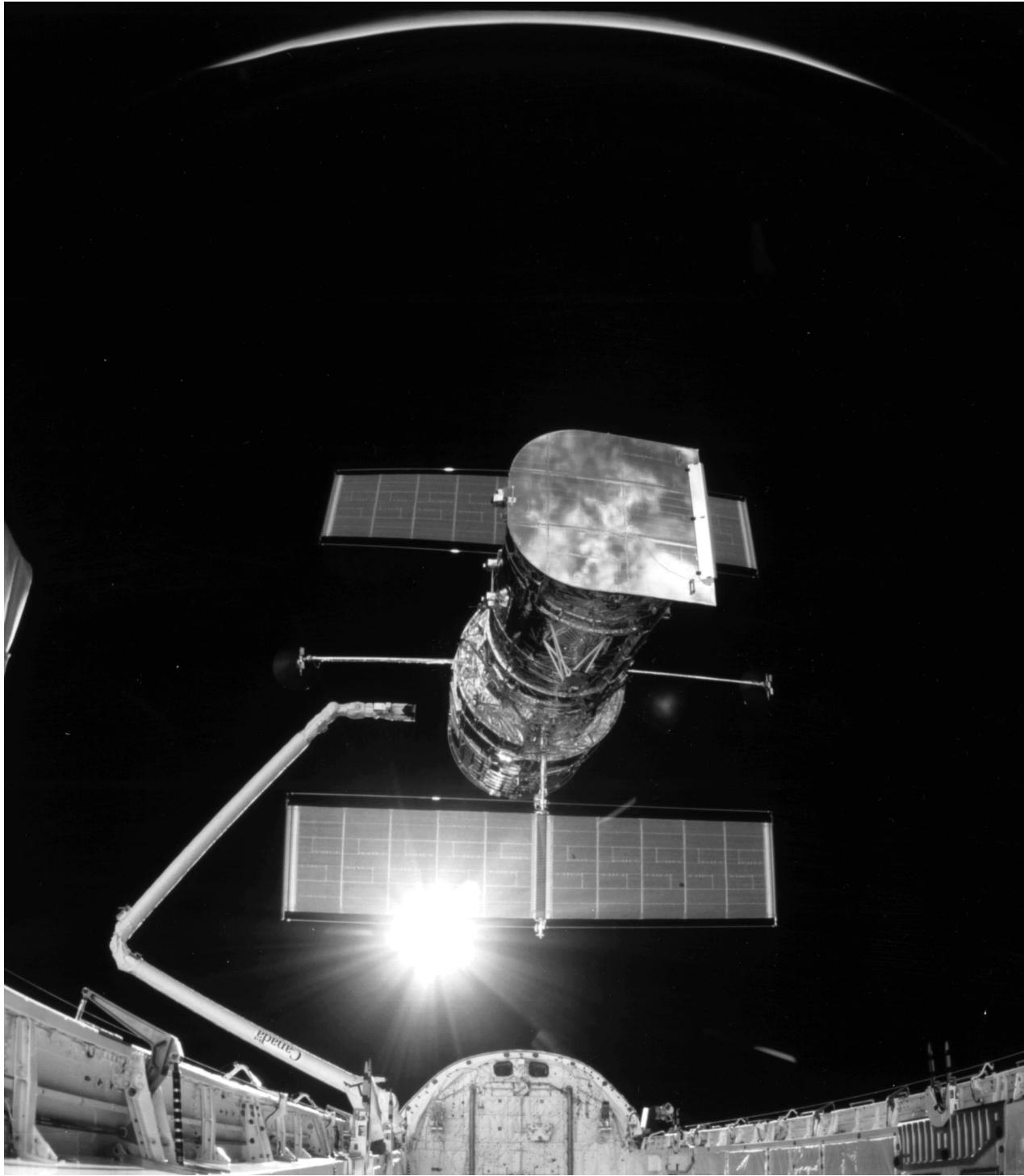


Figure 121. HST deployment after fix.

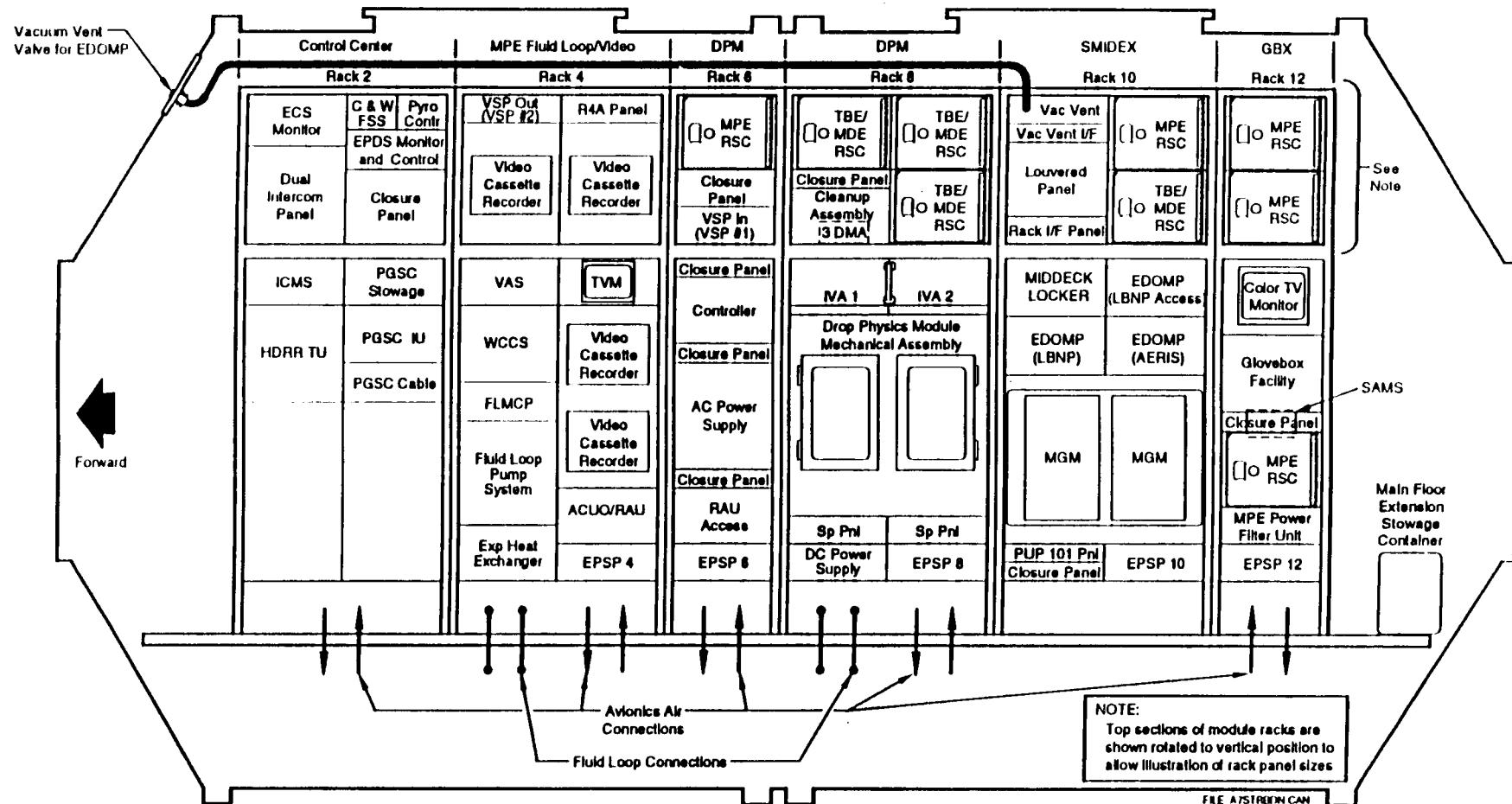


Figure 122. Spacelab module configuration.

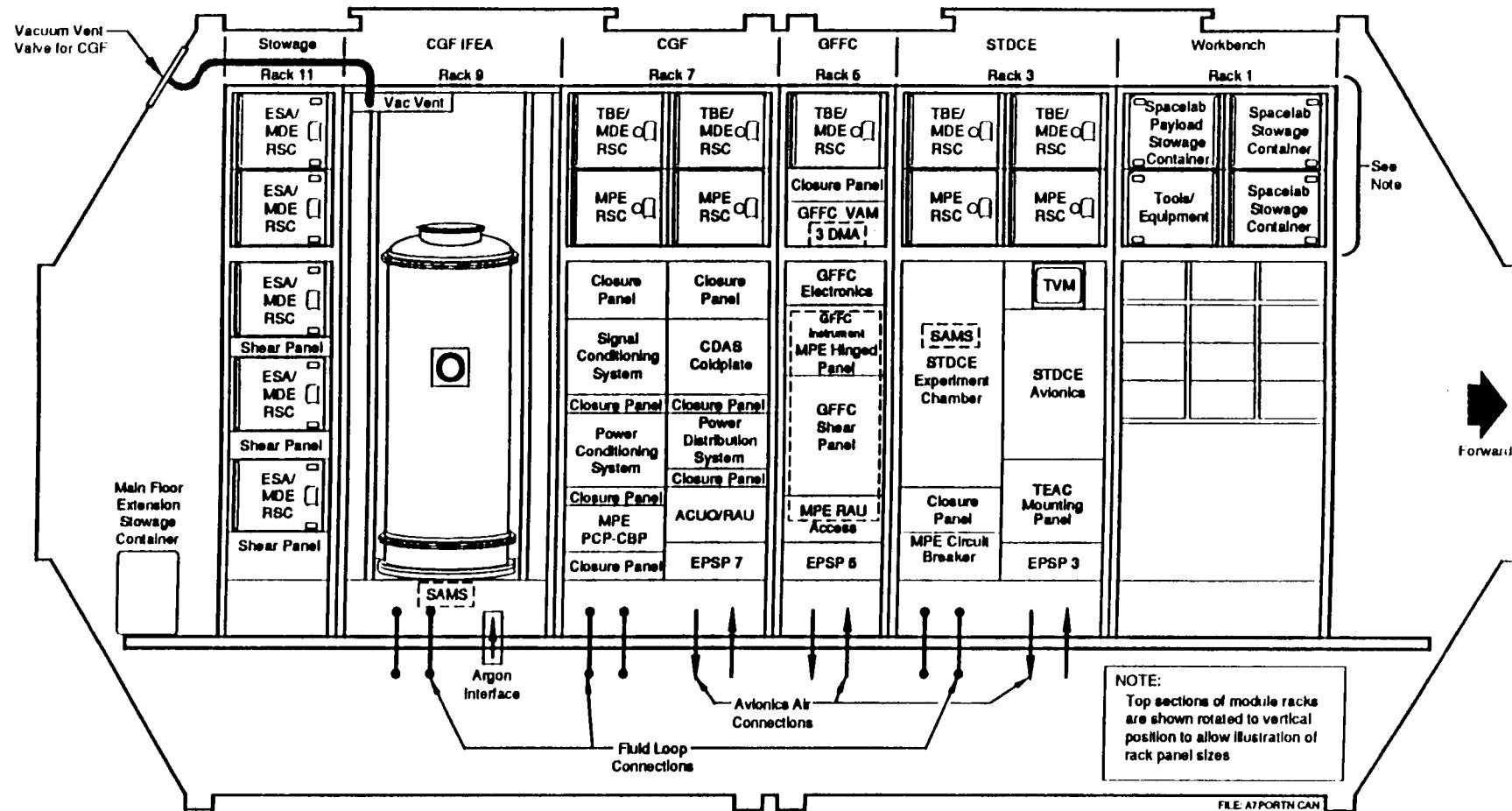


Figure 122. Spacelab module configuration (continued).

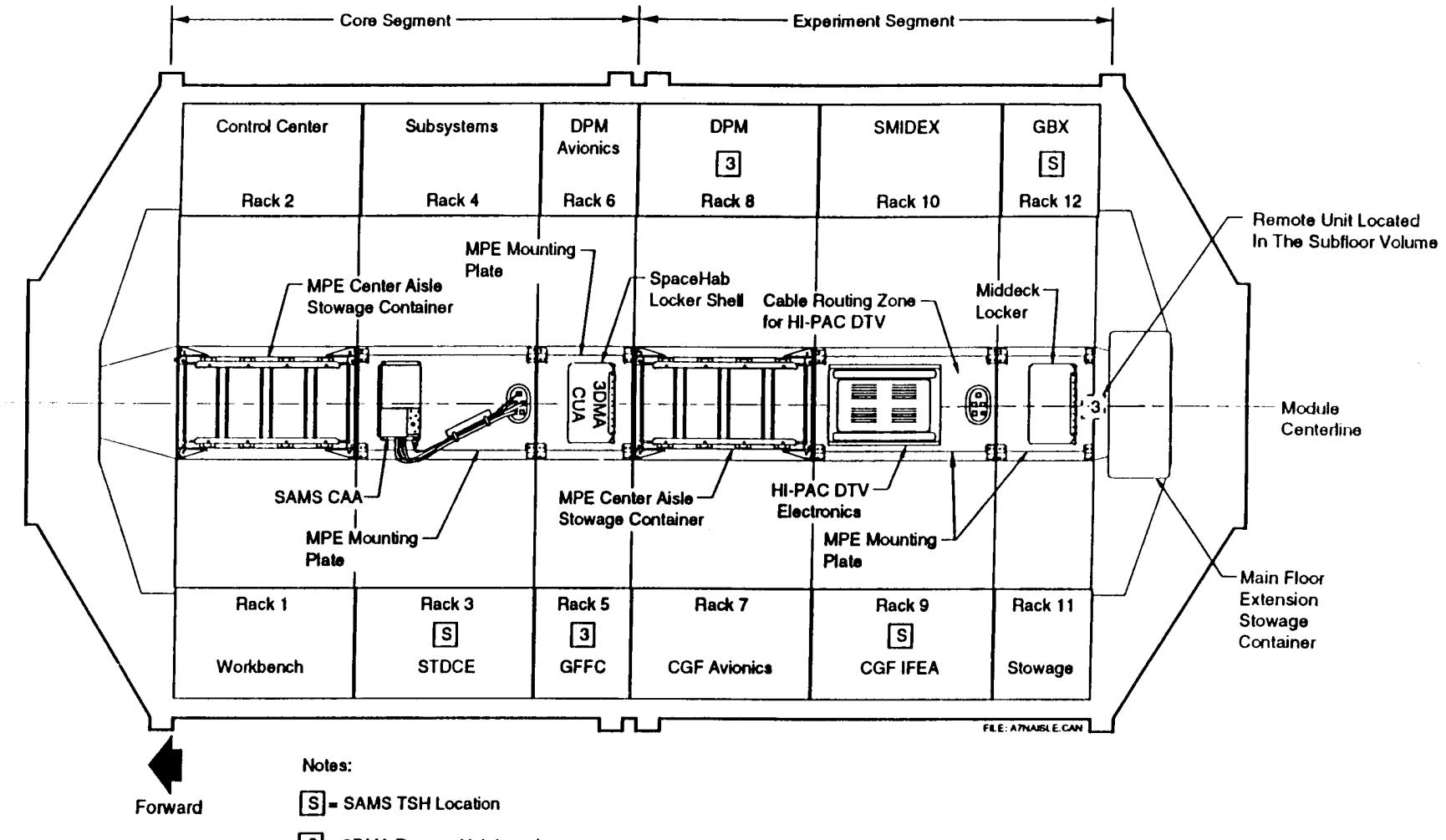


Figure 122. Spacelab module configuration (continued).

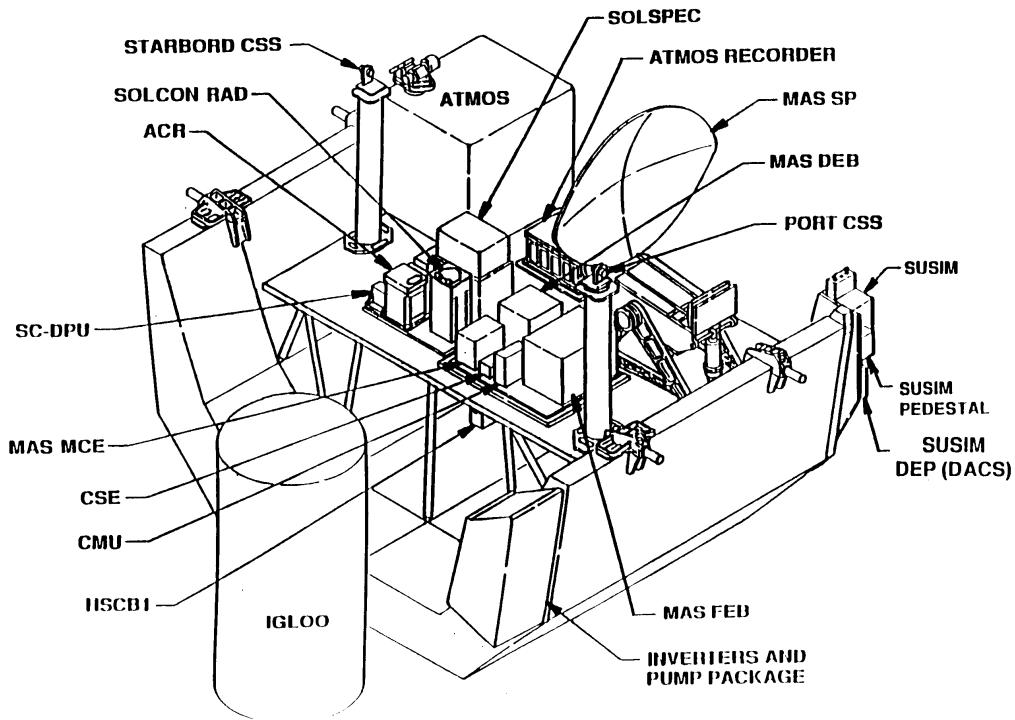


Figure 123. Spacelab typical pallet configuration.

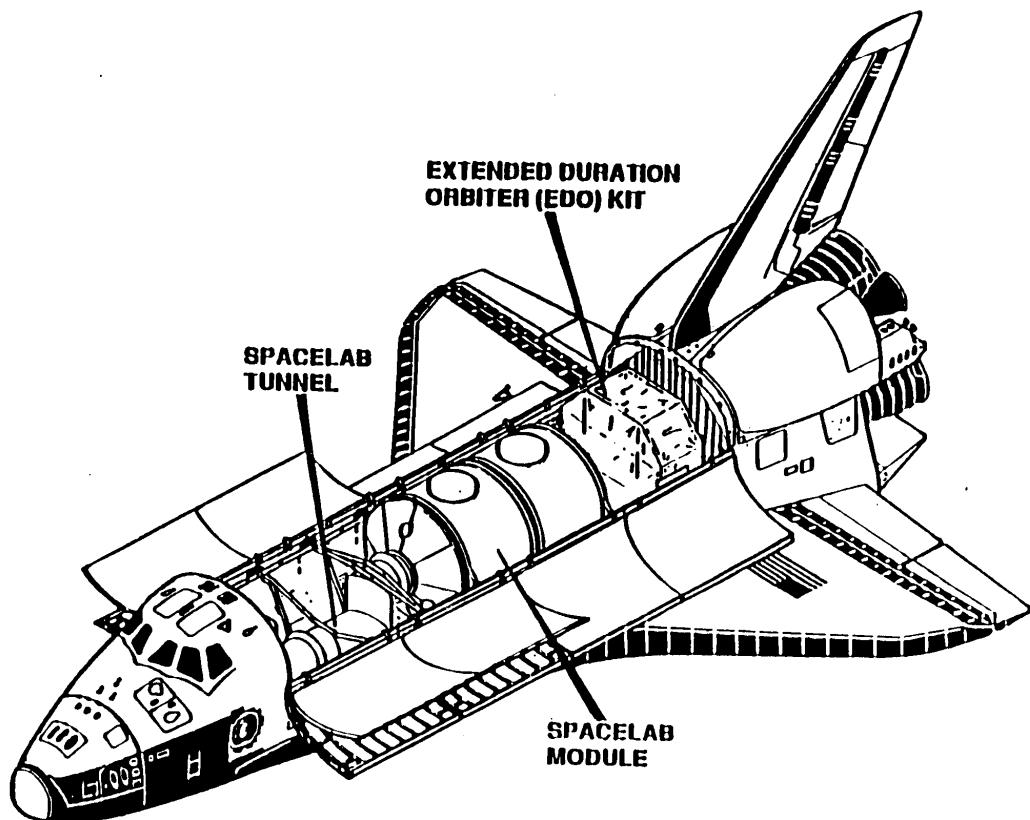


Figure 124. Spacelab module and pallet combination.

Spacelab, as a laboratory, has emphasized microgravity science, life sciences, astrophysics, solar science, space plasma physics, and atmospheric science. It has proved to be a very valuable workhorse, extending the shuttle to be an effective short-term laboratory, and has established an excellent technical base for the upcoming *International Space Station*.

2. Evolution/Creativity/Evolution. The basic idea of a combination of working quarters in space (modules), in combination with the V-shaped pallets, was very creative. The modules themselves, with the various type racks, gave much flexibility in terms of experiments and laboratory setups. Obviously a laboratory must have the capability of heat and cold, vibration, etc. All the materials and equipment must be compacted, stored, and fastened down to survive the launch, reentry, and landing environments, which was no small feat within itself. Delicate observation instruments had to have the same stowed requirements, yet, while in space, they had to be very flexible for pointing control, etc., and to some extent be isolated from the shuttle on-orbit motions.

3. Problems. One problem area for Spacelab was the design and verification of the module racks. There have been two basic problems, the first being the load paths and load carrying capability. The other, the rack natural frequency.

Load carrying capability is best exemplified by the high stress levels that occur for the mission peculiar equipment (MPE)-to-drop physics module (DPM) mechanical assembly interfaces, MPE-to-Spacelab rack interfaces, and DPM microgravity measurement assembly (MMA)-to-Spacelab rack interfaces. The cause of the high stress levels was due to the stiffness and large mass of the DPM in the lower portion of rack 8, especially the MMA. The problem has been solved by maintaining minimum “close fit” interfaces between MPE, DPM, and the Spacelab rack, using match and drill for MMA and epoxy in selected slotted holes in the rack.

In order to ensure that Spacelab equipment and experiments do not tune dynamically with the shuttle dynamics during lift-off and landing, a lower frequency constraint of 25 Hz was placed on major elements, such as racks, and 35 Hz on secondary elements, such as experiments attached to the racks. The racks had trouble meeting their frequency constraint, both analytically and through dynamic verification testing. There were two solution options: (1) redesign the racks to accept the larger loads introduced by the dynamic tuning, and (2) redesign the racks to meet the frequency constraints. The later approach was taken, with minimum design fixes.

A few of the other Spacelab problems are listed in the matrix.

I. *International Space Station*

The design of a long-life space station must deal with several demanding requirements. First, it must be able to withstand the launch environments, which have a different demand than on orbit. Second, since it is not possible to launch the totally assembled station at one time, the station must be designed for on-orbit assembly. Third, the station is to be human-tended, requiring a life support system that has minimum refurbishment and logistic requirements. Fourth, the station must be designed to withstand harsh on-orbit environments such as debris impacts, thermal variations, etc., with minimum refurbishment. Fifth, power must be obtained from the Sun, which requires solar cells, batteries, etc., needing minimal attention. Sixth, it must be a scientific laboratory, with all that implies. Seventh, means of docking and undocking spacecraft, such as the shuttle, must be accommodated with means of transferring crew and supplies, as well as having a crew return vehicle in case problems occur. Also, some experiments can be deployed and retrieved. Eighth, cost is a big driver and must be a major criterion in the design and operations. Finally, it can be a wayfaring station for missions to the Moon and planets.

As a result of these complex requirements, the space station configuration has gone through many evolutions in order to cut cost and better align with the national budget constraints. During this process, not only has the configuration changed greatly, but it has become truly an *International Space Station (ISS)* with hardware being provided by Russia, Japan, Canada, and ESA (fig. 125). The impacts

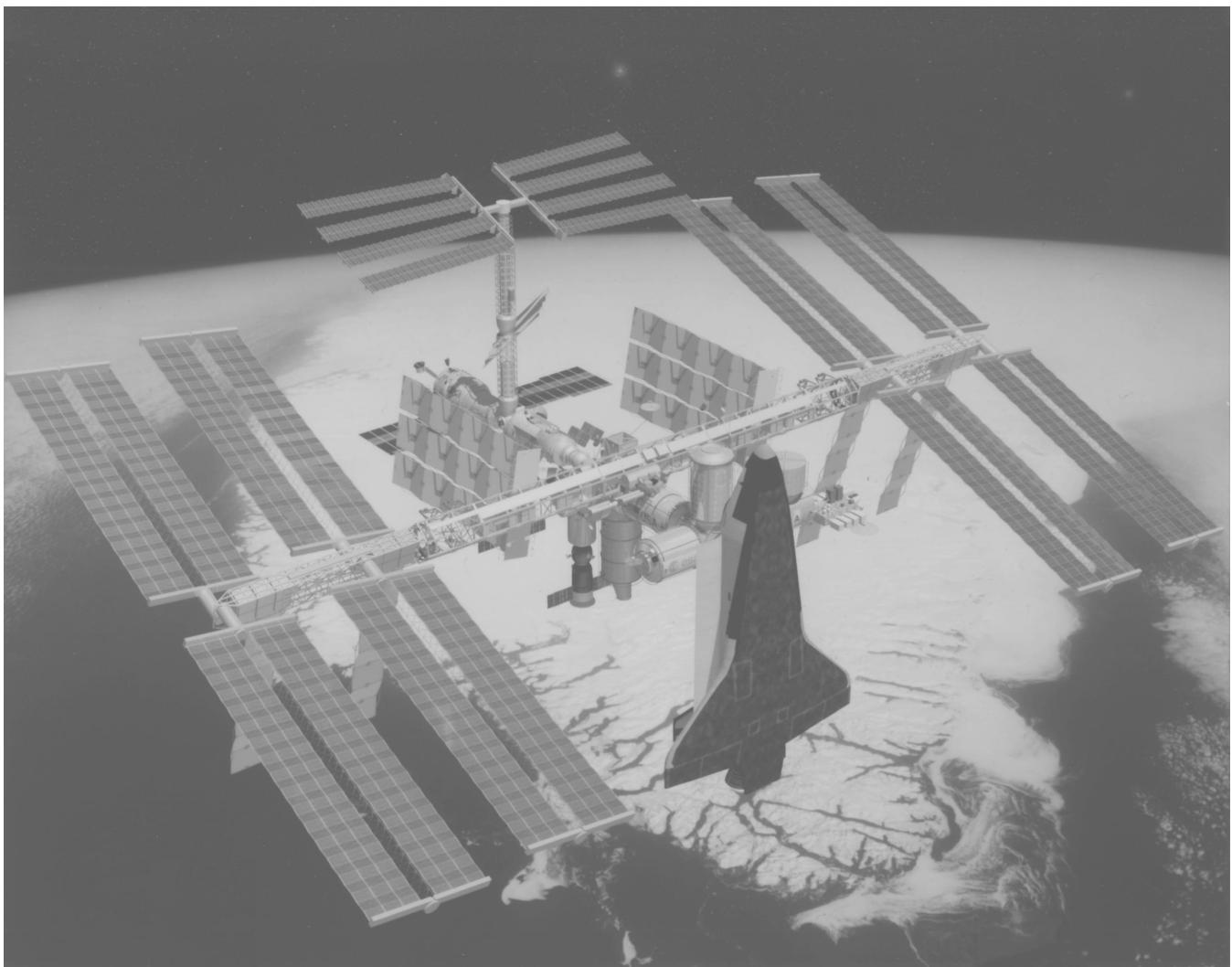


Figure 125. *ISS*.

of these changes have affected not just the station, but have changed how science and operations are achieved, as well as the impacts to the space shuttle hardware required in order to place the desired components into the proper orbit. (These changes were discussed partially under the section on the SLWT.) As a result, the present configuration is now frozen and in final design, with hardware being manufactured. This configuration meets all the basic requirements and is well on the way to operational capability.

1. Characteristics. There are many crucial subsystems to a long-term orbiting station. They include, but are not limited to: (1) life support, (2) power, (3) propulsion, (4) control, (5) science laboratories, (6) docking adapters, (7) structural systems, (8) meteoroid debris protection systems, and (9) crew quarters. Most of these have many subsystems or elements themselves with specialized functions to perform. All these subsystems work together as the *ISS*.

The environmental control and life support system (ECLSS) is composed of several subsystems (fig. 126). The ECLSS functions and subfunctions are shown on figure 127, and include atmosphere revitalization, atmosphere control and supply, temperature and humidity control, water recovery and management, waste management, fire detection and suppression, and other functions such as food storage and preparation, radiation protection, etc. The goal of ECLSS is to provide a healthy, supportable environment in which astronauts can live and work efficiently without costly use and loss

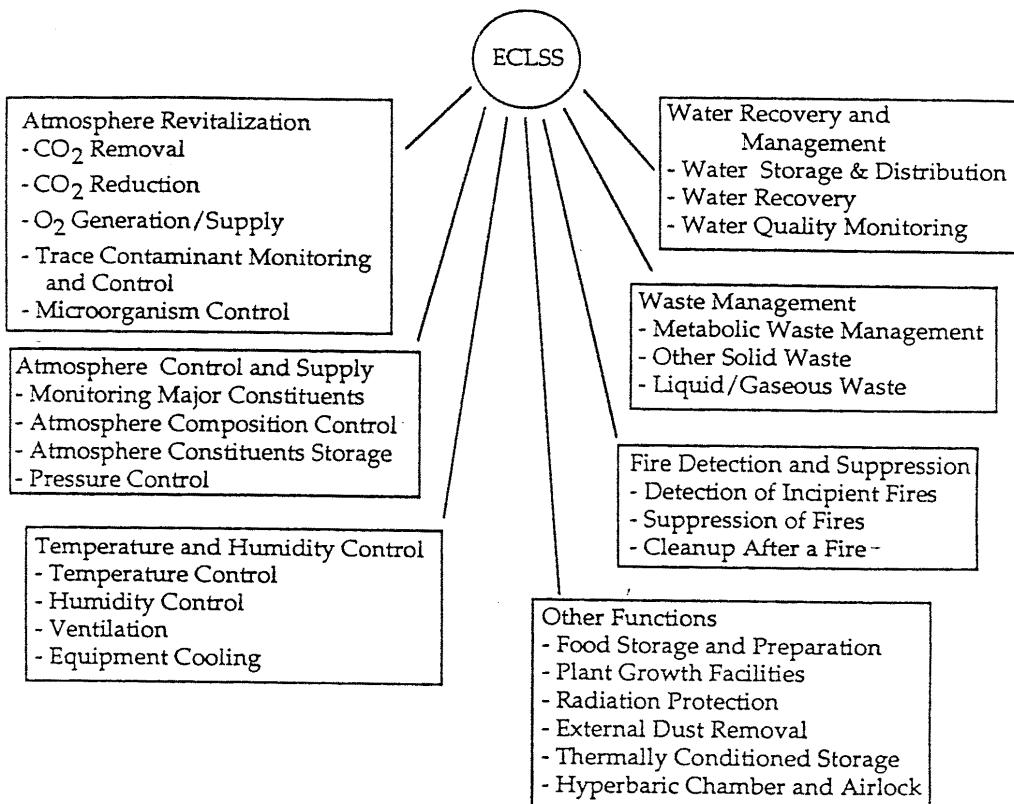


Figure 126. ISS ECLSS.

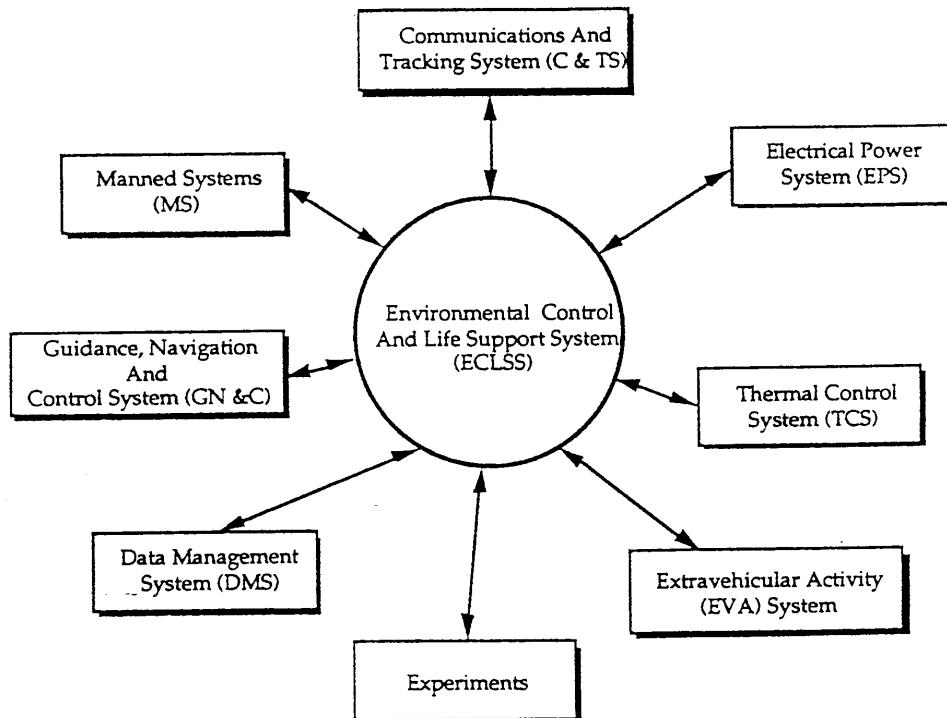


Figure 127. ISS ECLSS function.

(replace) as the current systems used on shuttle do. Instead of carrying enough oxygen, water, etc., for the mission, the system recycles, as much as is economically feasible, comparable to the Earth's system. Figure 128 is the basic *ISS* showing the ECLSS components. The U.S. laboratory module is shown of figure 129. This module will serve as the main control to support the subsequent assembly flights. It includes internal audio system, video system, and internal control and life support systems.

Another very critical subsystem is the meteoroid debris protection system. This system is designed to mitigate debris damage to the station, potential loss of human life, and loss of critical functions and science. The approach deals with the debris environment determination/impacts, impact damage, and loss due to damage (fig. 130). The external protection system is built up of layers of shields/bumpers as shown on figure 131.

Solar arrays provide the energy for the power system, as was shown on figure 125. The system receives the Sun rays, converts it to electrical power, and stores the excess in the batteries. The structural system is composed of the central truss around which the other elements such as the modules are attached. There are docking modules for the space shuttle and Russian spacecraft and the abort crew recovery system. Figure 125 shows many of these systems as well as the country providing the hardware.

The space station is a very complicated system. It is not the purpose of this report to discuss all these characteristics. It must be assembled in space using several launches spread out over a year or two. To utilize the Russian launch capabilities, a 51° orbit inclination led to the shuttle performance enhancements that included the development of the lighter weight external tank. At the time of this writing, all were being accomplished as planned.

2. Evolution/Innovation/Creativity. There are so many innovations that have occurred to date in the development of *ISS*, it is not possible to discuss them all. Obviously, since *ISS* is still in development, many others will evolve.

The meteoroid protection system mentioned previously is truly a creative design. Not only does it protect against impact damage, but it also deals with equipment location to minimize damage, crew escape procedures, etc. Not only has the design been very creative, but it also has the analysis and testing approach. The testing of impact damage determining hole size, leak resistance (rate), etc., for many different protection approaches was completed. For example, figures 132 and 133 show the test setup for measuring depressurization of a module.

Development of the life support system has many innovations in its design. These systems are very complex, composed of chemical, fluid, electrical, hydraulic, control, etc., all linked together in a highly interactive system. To develop and verify ECLSS, long-term hardware tests have taken place from a module loop down to a water recovery system test. This innovative breadboarding has allowed problems to be made visible and the design to be corrected.

The management approach between Government and industry has also been very innovative. There is one prime contractor—Boeing. Integrated product teams are used for design and development. These teams have Government support, with membership and engineering tasks from analysis, to simulation, to testing. Integrating into the *ISS*, hardware from the international partners has challenged the best in innovative management. To date, all is working well, hardware is built, other hardware is proceeding, operational procedures are under way, as well as the on-orbit assembly techniques, etc. Problems have occurred and will continue. The team in place should handle them well.

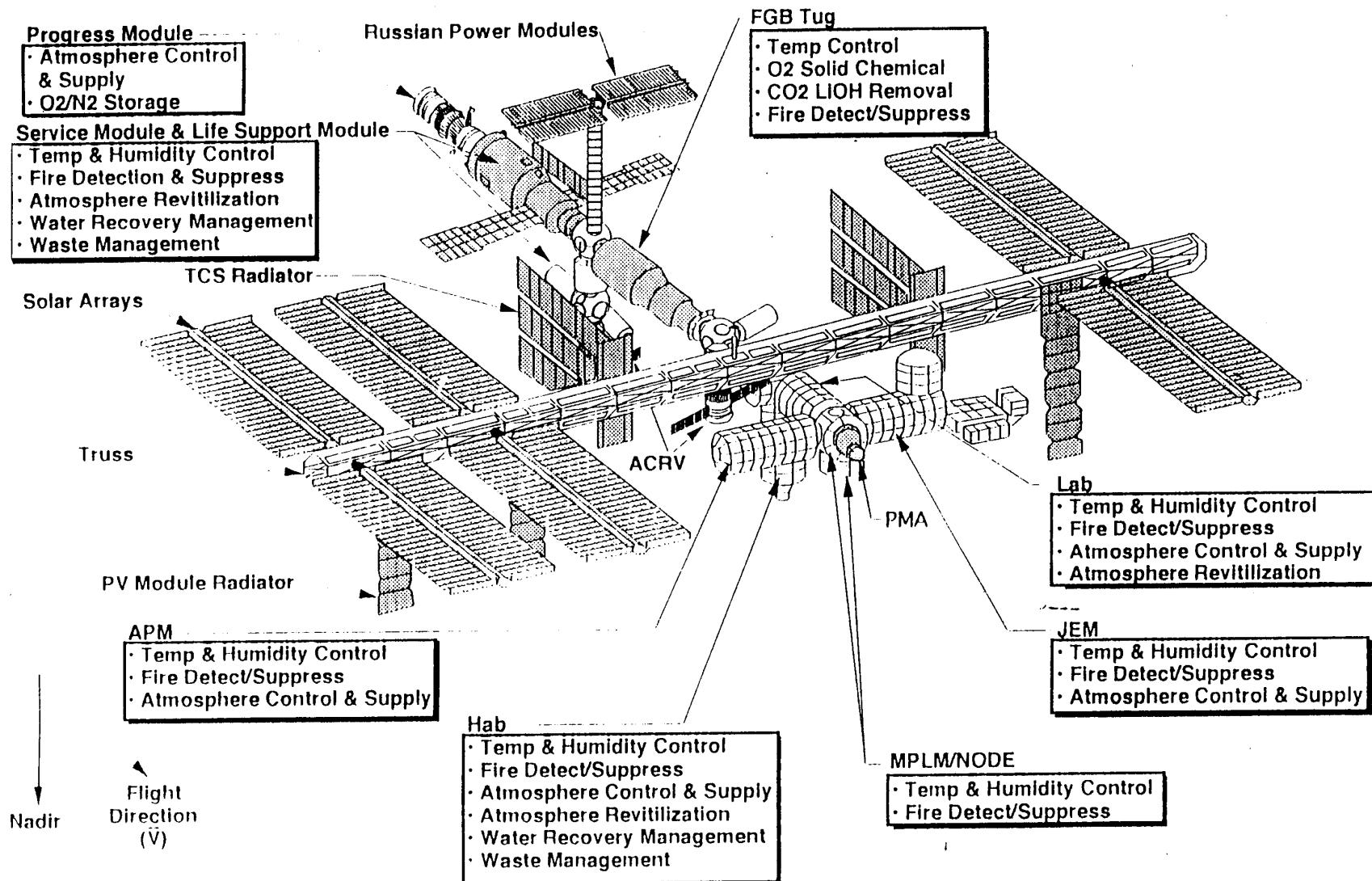
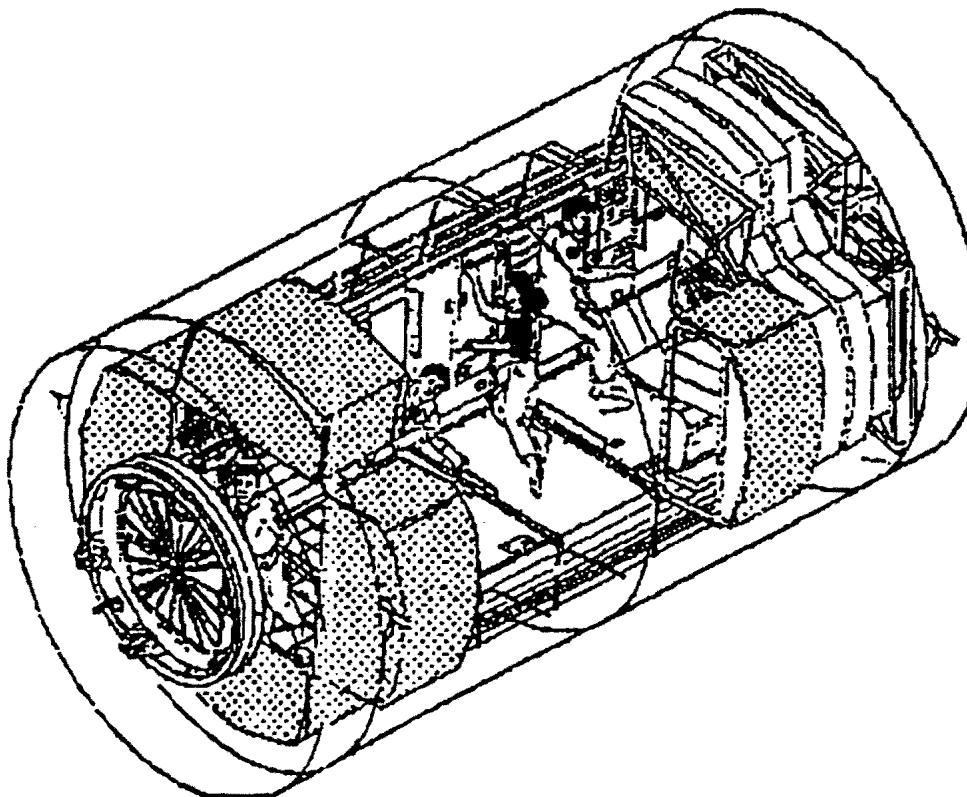


Figure 128. ISS with ECLSS subsystems.

U.S. Laboratory Module

The U.S Laboratory Module provides the main control to support subsequent assembly flights. Active Lab systems include internal audio system, video system, Internal Thermal Control and Life-support system.

Figure 129. ISS U.S. laboratory module.

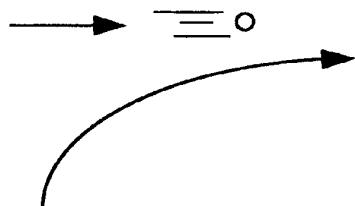
Orbital Debris Risk Mitigation

$$P_{\text{loss}} = P_{\text{impact}} \times P_{\frac{\text{damage}}{\text{impact}}} \times P_{\frac{\text{loss}}{\text{damage}}}$$

Reduce P_{impact}

- Environment Definition

→ **Debris Reduction, Tracking,
& Avoidance**



Reduce $P_{\text{damage/impact}}$

- | | |
|------------------------------|-----------------------|
| • Penetration | • Bondline Separation |
| • Ricochet | • Spallation |
| • Pressure Vessel
Rupture | • Leaks |
| | • Delamination |

→ **Augmented Shields and Materials**

Analysis Tool: BUMPER

Reduce $P_{\text{loss/damage}}$

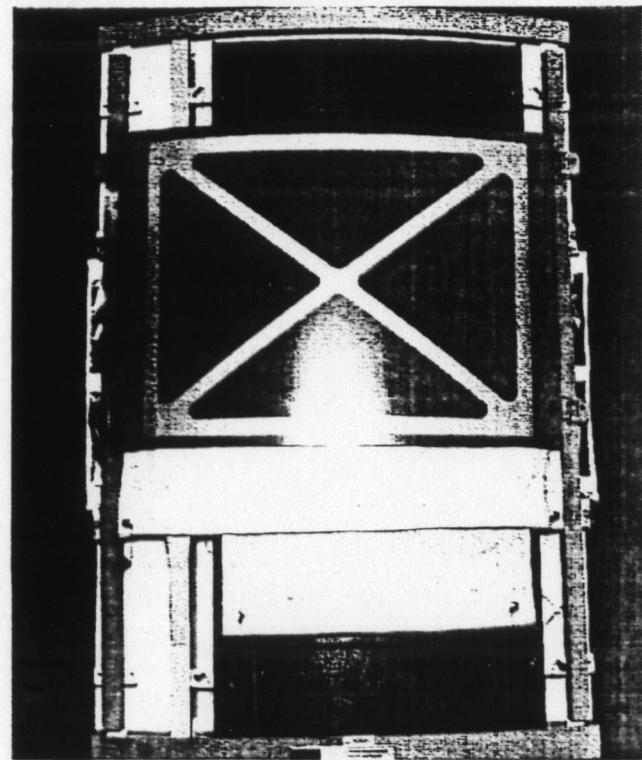
- Critical Cracking
- Control Loss
- Critical Equipment Loss
- Injury Loss
- Hypoxia
- Repair

→ **Equipment Locations**
Internal Spall Blanket
Hatches Open/Closed
Crew Position in Modules
Sealing/Repair Strategies

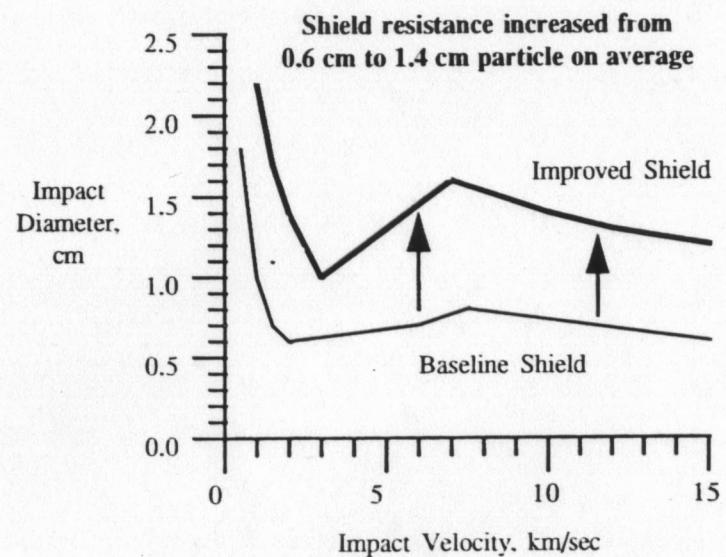
Analysis Tool: MSCSurv
(Manned Spacecraft Crew Survivability)

Figure 130. Approach for meteoroid debris protection for ISS.

ENHANCED MANNED MODULE ORBITAL DEBRIS SHIELD



Ballistic Limit Comparison (45 deg impact)



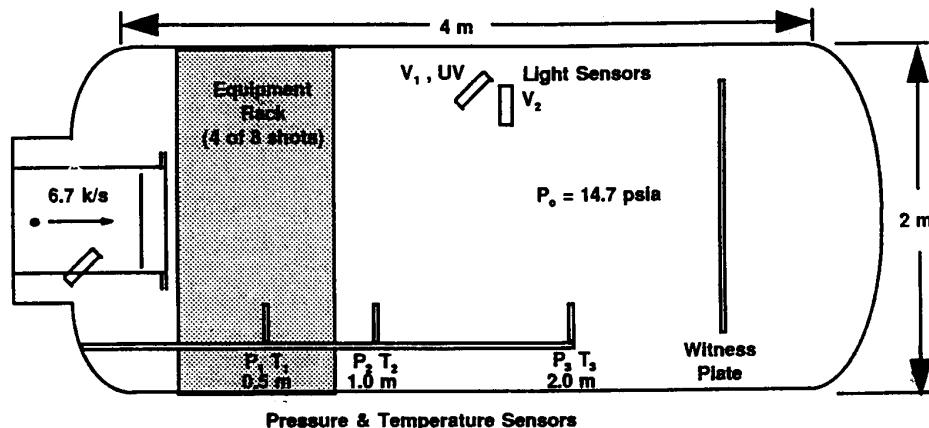
With New Shields
U.S. Lab = 0.9935 Probability of No Penetration for 10 years

Figure 131. ISS debris protection configuration.

Manned Module Internal Effects Following A Penetration

MSFC - University of Alabama in Huntsville

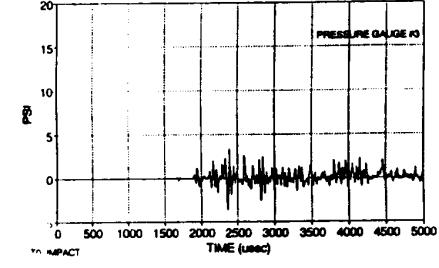
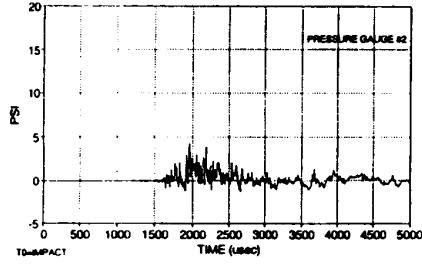
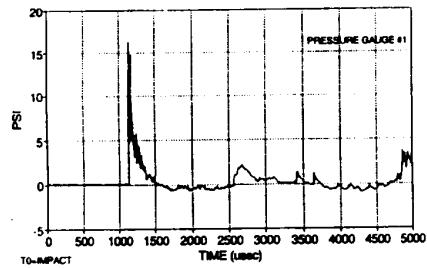
- Eight tests measuring:
 - Overpressure
 - Flash intensity
 - Temperature rise
 - Fragment dispersion.
 - Determines effects of:
 - Projectile energy
 - Shield type
 - Internal equipment
 - Spall blankets
- on hazard levels experienced by crew members.



Overpressure

Test #1 7/18/94
Dia = 0.52 in.
Vel = 6.7 k/s

U.S. Lab Whipple Shield
No Internal Equipment



Temperature Rise

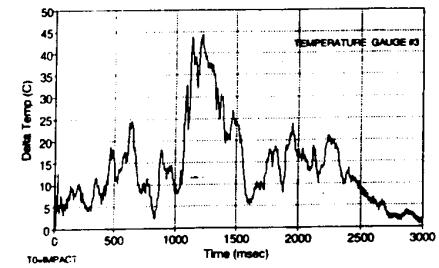
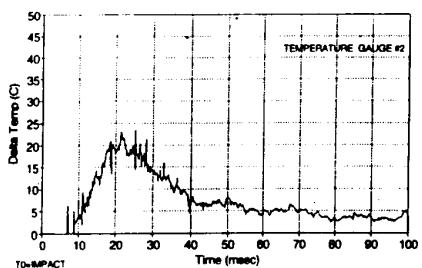
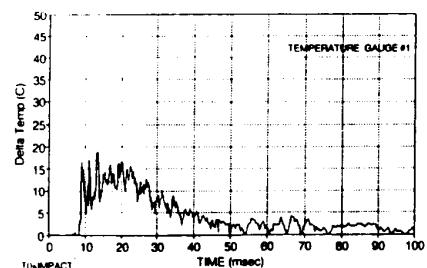
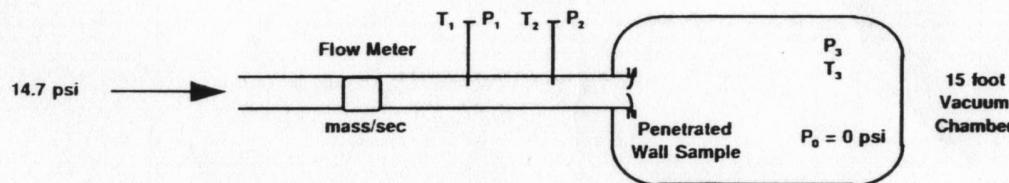


Figure 132. Meteoroid debris testing approach.

Manned Module Through-Wall Hole Depressurization Tests
MSFC Test Area 2



Objective: Measure the discharge coefficient (d_c) for a variety of hole sizes and shapes.

Use measured d_c to determine available escape time.

- 10 Samples
- Hole Sizes range from 0.4" to 3.0"

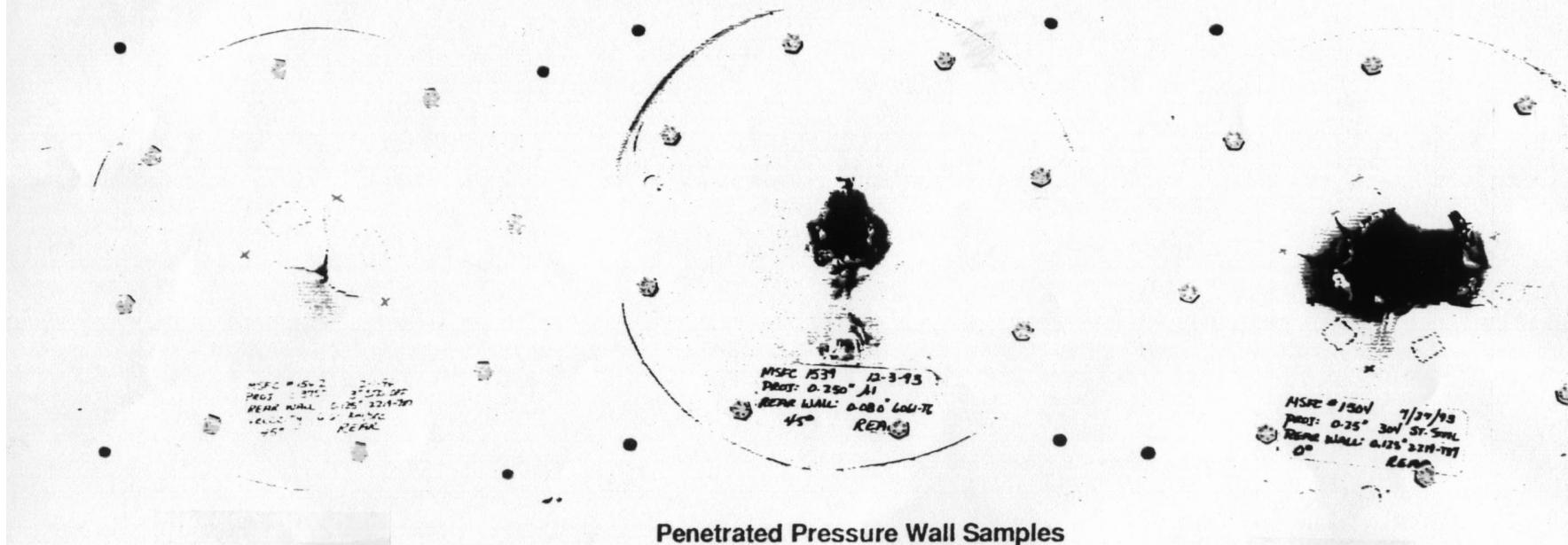


Figure 133. Debris impact results of pressurized container.

3. Problems. Although many problems have occurred so far in *ISS* development, only a few typical ones will be highlighted. As was experienced on the Spacelab project, the racks did not meet the 25-Hz frequency constraint due to its open box construction for crew utilization. The problem was solved by adding stiffening to the rear of the rack and making the utility panel a structural element instead of a closeout, which raised the frequency without compromising the crew access.

The meteoroid debris protection system design was discussed in section 2 and requires no further discussion. The glass window design for the modules experienced fatigue cracking when the sustained stress due to on-orbit delta pressure was applied. The problem was aggravated by the moisture content of the ambient air, causing stress corrosion effects in the glasses' microcracks when under stress. Several things were accomplished to solve this problem: (1) conservative fracture mechanics methodology was employed for the window redesign; (2) proof testing of each pane of glass was required to screen initial flaws larger than those acceptable by the fracture analysis; (3) windows were redesigned such that tension is not on the inside surface of the pane, making it damage tolerant; and (4) the cavity between the primary and redundant pressure panes was vented to space to reduce exposure to moisture.

The development of the life support system has had several problems. These developmental problems have utilized long-term ground demonstration testing of the various subsystems. As a result, the hardware has experienced many development problems. For example, the testing of the Hab/Lab element resulted in cyclic variations in cabin temperature, carbon dioxide (CO_2) partial pressure, and dew point. It was found that the CO_2 removal subsystem was returning air exhausted into the cabin. The solution was to reroute the CO_2 removal subsystem exhaust to upstream of the condensing heat exchanger. Another problem had to do with the overall system complexity and component redundancy, which utilized separate loops for potable and hygiene water reclamation. The solution was to combine the two loops into a single water reclamation loop. The water system in another problem was not completely sterile prior to testing. This was caused by microbial growth in the test system particularly in plumbing dead legs. The solution was to eliminate the plumbing dead legs and sterilize the system with hot water at 250 °F. Water was also entrained upstream of the CO_2 removal subsystem. This was caused by a break down of the silica gel when subjected to liquid water. The CO_2 removal subsystem was modified by adding a layer of Zeolite 13x™ upstream of the silica gel, solving the problem. See the matrix for other problems.

J. Titan Viking

The Titan launch vehicle has been a workhorse not only for Air Force payloads, but also for many NASA payloads. The author worked the Titan Viking program from the launch vehicle standpoint, which dealt with three launches: Viking simulator, German Helosis, and Viking, mainly from the standpoint of a first stage pogo concern/problem. The pogo concern, and thus the pogo working group, was active for approximately 2 years part-time. Jet Propulsion Laboratory (JPL), LaRC, MSFC, and Lewis Research Center (LeRC) were the NASA Centers involved with the Martin Aerospace Corporation.

The vehicle had not only the boost stage with solid strap-ons, but also an upper stage called Centaur. The basic problem that Viking faced had to do with a known pogo oscillation that occurred near the end of the liquid boost stage burn. When the Titan Viking simulator test was flown, this pogo oscillation was of family, creating loads on the spacecraft that were larger than the design loads. Since the basic spacecraft was already designed, this was a major problem. Either the spacecraft had to be beefed up structurally or the pogo problem eliminated. As a result, a blue ribbon team was formed to understand and recommend actions in terms of the problem. The team stayed active until just prior to the Viking launch. The preliminary assessment showed a dynamic tuning of the Viking simulator with the vehicle longitudinal bending mode, reducing the equivalent damping or, said differently, increasing the modal gain in the pogo loop. At that time, it was rationalized that the real Viking would be different; therefore, the program proceeded.

The next Titan launch was the German Helosis payload and was comparable in size to Viking. Since Helosis had no loads problem, it was deemed acceptable to launch the system. The result of the launch was successful; however, the pogo response was larger than for the Viking simulator. The team, along with contractors, spent a lot of time studying the data, running sensitivity pogo and loads studies, and conducting special tests. It was found that the vehicle had stood on the pad for 2 days in a cold wind, which shifted the gain in the propulsion system, changing the pogo response.

When this was understood, it was decided to put an accumulator on the propellant duct, detuning the structural system from the propulsion system, thus eliminating the pogo oscillation. Parallel to this fix design, structural dynamic tests were run on the Viking payload in order to help verify the margins of the Viking system. The pogo fix was designed and installed without impacting the launch date. The flight was successful. No pogo occurred, thus there were no load increases to impact the Viking payload. History shows the final success of the Viking probe.

The message of this activity is very clear. When large energy sources (thrust) are present, coupling dynamically with other systems (structural, fluids, and acoustical), the response is generally nonlinear and is, therefore, very sensitive to small variations. The small shift in dynamics (Viking simulator) from prior systems and the longer pad time in low temperatures (Helosis) are examples of how small changes and the nonlinearities can create large unwanted responses. When these conditions are present, it is always prudent to eliminate the coupling and make the system robust. This was done for Titan Viking, resulting in a successful program.

K. Experiments

NASA and Marshall have flown many experiments, in particular on the space shuttle, mainly using various parts of the Spacelab discussed previously. These experiments have many times been joint ventures with other countries such as Italy, Japan, ESA, Canada, etc., where the system was built by the International partners, flown by the U.S., and a joint experiment conducted. Others were totally U.S. experiments. The following is a partial list of these experiments.

<u>Experiment</u>	<u>Purpose</u>	<u>Supporters</u>
IPS	To develop a common pointing mount for flying observatories in the space shuttle.	ESA/U.S.
SAFE/DAE	To qualify a lightweight solar array with fold/unfold capability and demonstrate on-orbit dynamic test capability using remote sensing.	U.S.
Astro	To give astronomers a view of the universe impossible to obtain from the ground.	U.S.

1. Safe Solar Array/Dynamic Augmentation Experiment (SAFE/DAE). In September 1984, NASA tested the SAFE/DAE on shuttle mission STS-41D. This testing qualified advanced solar arrays and large space structures technologies that have several features which had marked improvements over what was being used at that time.

The solar array was approximately 100-ft long (30.9-m) and 13.1-ft (4.0-m) wide. The total experiment, including the array, totaled 940 lb in weight. The array was a flat-fold flexible substrate solar array sized to output more than 12.5 kW at the base of the wing at 55° centigrade normal to the Sun in near-Earth space. The array blanket consisted of 84 panels, 0.37 by 4 m (14.5 in by 13.1 ft) in size, that were mechanically hinged together to allow easy replacement of the panels. Only two of the panels had active solar cells for the experiment. The other cells used glass sized simulators in order to reduce experiment cost (fig. 134). The wing was deployed and retracted using an AEC-ABLE Engineering

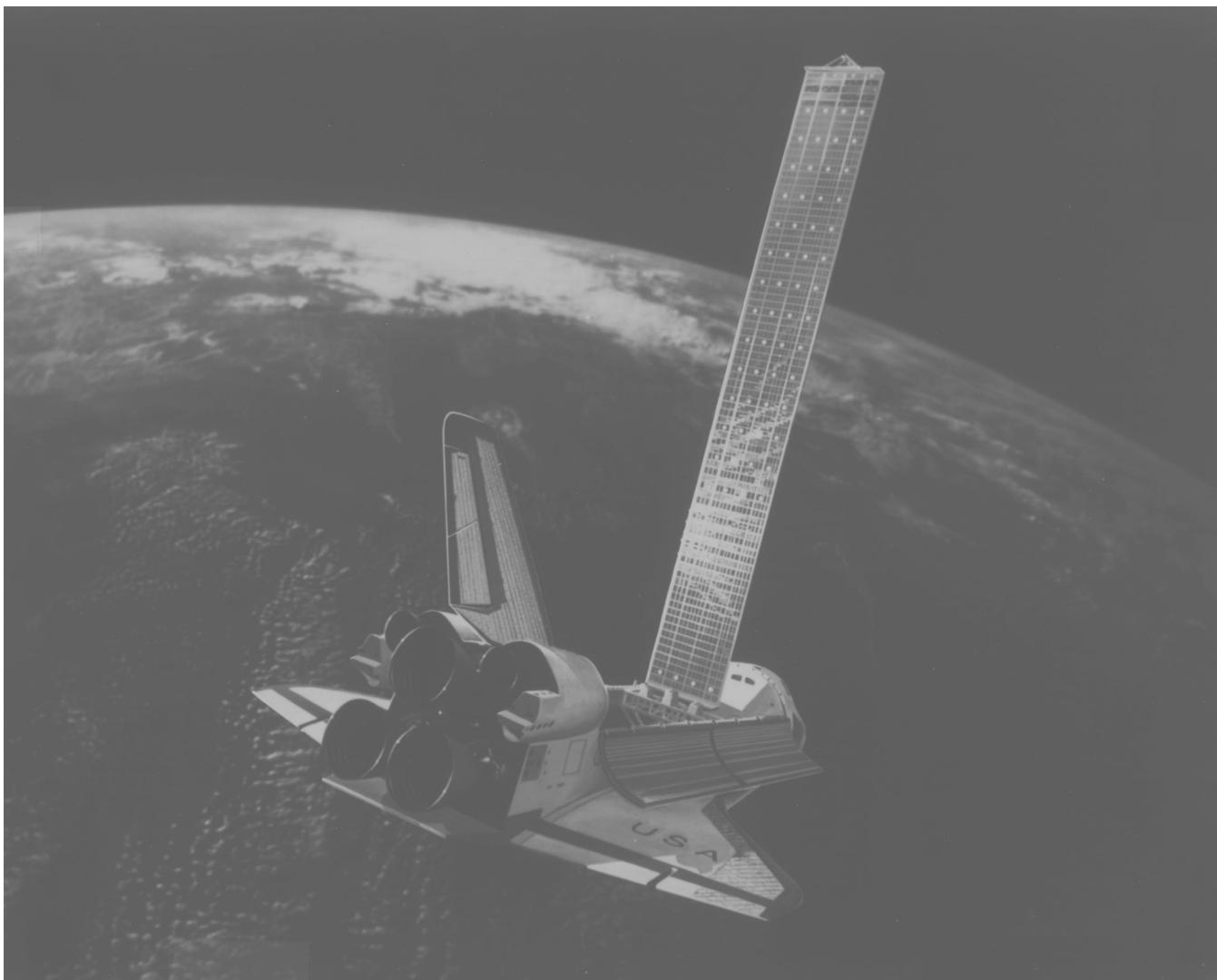


Figure 134. Solar array deployed.

Company mast that retracted into a canister (fig. 135). The solar array mast combination was attached to a Spacelab pallet in the shuttle cargo bay. The array/mast was deployed and retracted four times during the mission's 23 $\frac{1}{2}$ h of experiment time.

Dynamically testing a large space structure in orbit is highly desirable since such a large lightweight structure cannot be tested on ground in the gravity environment. Two approaches were used to acquire the dynamic data when the system was excited using the orbiter's vernier control reaction system (VCRS) jets. The two approaches were: (1) Photogrammetry using shuttle cameras to record the dynamic motion of the structure. Sixty-three white circles, approximately 16.5 cm (6.5 in) in diameter, were located at regular intervals on the front of the blanket. Another 63 were located on the back side of the blanket. These data, collected during and after the VCRS firings, were evaluated using special data reduction techniques to produce mode shapes and frequencies. LaRC developed the photogrammetry approach. (2) Dynamic augmentation experiment (DAE). The DAE is an adaptation of a multifield star tracker that was designed to determine the dynamic characteristics during the above discussed tests. A retroreflector field tracker (RFT) positioned near the base of the solar array wing used laser diodes to illuminate 23 targets on the array. A second microprocessor computed the dynamic displacement from the line of sight, feeding the results to an onboard recorder. The data were ground processed, determining mode shapes and frequencies.

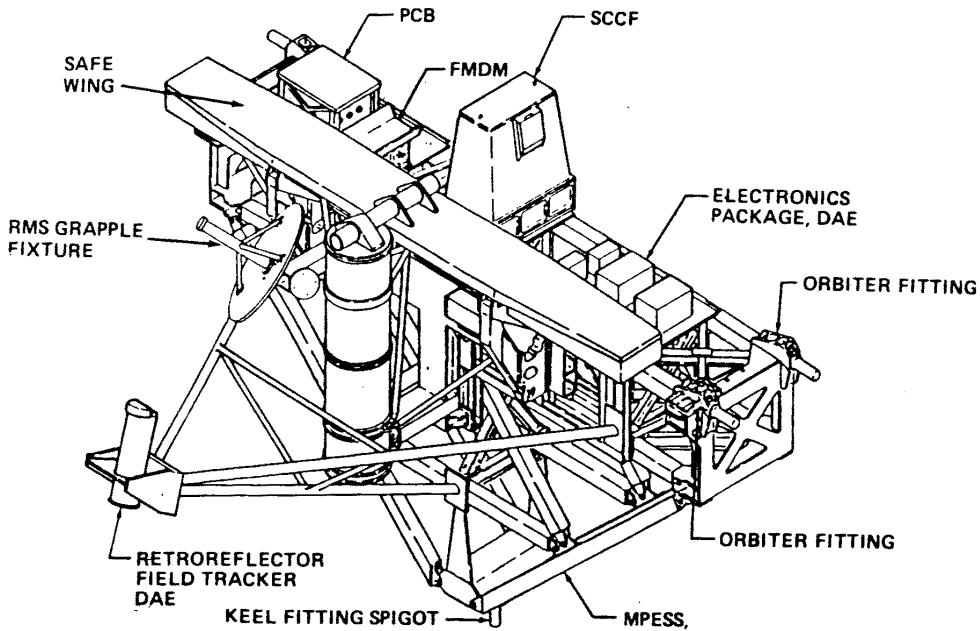


Figure 135. Solar array deployment mechanism.

The results of the dynamic experiment were very good. One anomaly occurred when the array went from day to night, causing curling of the array coupling pitch and yaw, and slightly shifting the frequencies. Therefore, the frequencies of the deployed array were consistently higher in darkness than daylight. Postflight analysis verified this effect. Overall, the predictions (after this modification) of frequency, mode shapes, and deflection amplitudes matched quite well.

Damping was consistently higher for in-plane motion than for out-of-plane motion. Lockheed performed tests showing that if the tension wire reel-out force was greater than the reel-in force by as little as 1 lb, then the entire kinetic energy would be dissipated in one-half cycle in the free-motion period. Testing found that the actual reel-out versus reel-in was about 0.65 lb.

The SAFE/DAE demonstrated that lightweight solar arrays, with reel-out/reel-in capability works very well. Power generation predictions were also verified. A first step in on-orbit dynamic testing of large space structures was demonstrated using two separate approaches. Both worked adequately.^{70 71}

2. Tethers in Space

a. Characteristics. Tethers can be used for many applications in space, a few of which are:

- (1) Electrodynamics power generation (electrodynamics brake)
- (2) Electrodynamics thrust generation
- (3) Multiprobe for atmospheric studies
- (4) Rotating controlled-gravity laboratory (tethered platform)
- (5) Tethered satellite for cosmic dust collection
- (6) Tethered docking

(7) Science applications tethered platform

(8) Waste disposal.

The first tethered satellite, called TSS-1, was built and flown to demonstrate the capability to deploy, perform science, and recover a tethered system. Two additional small-mass tethered systems, the Small Expendable Deployer Systems (SEDS), have been launched successfully from unmanned launched payloads. (These SEDS missions will not be discussed further in this report.) TSS-1 had a major problem with the level wind mechanism and did not fully deploy (discussed below). As a result, it was reflown in March 1996.

The TSS-1 system was composed of a spherical-shaped satellite, containing numerous scientific instruments, antennas, the reaction jet control system with appropriate sensors, and the tether attach mechanism.⁷² The satellite weighed approximately 520 kg. The basic features are shown in figure 136. The attitude control system is composed of four rate-integrating gyros with feedback, horizon scanner (Earth scanner), accelerometers and magnetometers, and thrusters (reaction jets) in the positions needed for attitude control, science, and tether deployment. There are eight thrusters groups. Four nozzles are in-plane (pitch), four are yaw, two are out of plane (roll), and four are in-line. The tether tension during operations is maintained by gravity, centrifugal force, and in-plane thrust.

The tether used for deployment, retrieval, and as a conductor was 20-km (12-mi) long. It had a core of Nomex™, the copper conductor, covered by a 0.012-in thick insulation and wound with 10 strands of Kevlar™, then braided on the outside with Nomex™. It had a diameter of 2.54 mm (0.1 in) and a mass of approximately 8.2 kg/km.

The support and deployer mechanism are shown in figure 137, including the docking and support ring, while figures 138 and 139 show the system in the cargo bay with the boom partially deployed. Figure 140 shows the tether control mechanism, while figure 141 is a sketch of the tether level wind. The system was composed of the tether control deploy and take-up mechanism, the satellite support structure, the extendable/retractable boom, the canister, etc., which served three functions: (1) provided a means of transportation in the orbiter (launch and landing); (2) served as an operational base for the deployer and, through the pallet, as an interface with the orbiter; and (3) deployed and retrieved the satellite. In the mission scenario, the total mission time was approximately 30 h. Deployment took the first 5.3 h, which was to be followed by 10.5 h of science time at the 20-km (12-mi) tether length. The first leg of retrieval took 6.75 h, followed by 5.0 h of science. During this time, an orbiter yaw maneuver can be conducted to dampen skip rope dynamics, if present and of a large enough magnitude. Final retrieval was to have taken 1.9 h and contain a period of satellite attitude control to dampen the remaining skip rope (fig. 142).

b. Evolution/Creativity/Innovation. The basic idea of a tethered system itself is very innovative, especially in light of potential applications mentioned earlier, and others not mentioned. The basic physics of the problem of using tethered systems centers around the conservation of momentum of the orbit. In this case, the orbiter and the satellite are a system in a specific orbit at a certain energy level, which must be held constant. The tether keeps the two bodies, though separated, operating as one orbiting system with the energy level of the system before the tethered satellite was deployed. A deployment force (thrusters) moves the satellite out from where the two end masses were initially in the same orbit. The gravity gradient force is proportional to the separation distance. After a certain distance from the orbiter, the combination of gravity gradient and centrifugal forces are sufficient to overcome system losses (friction) and continue the separation. Because the two masses are constrained by the tether, the upper mass gains a higher kinetic energy per unit mass than the lower mass. The tether constraint causes the orbiter and satellite to have the same angular velocity. During deployment, the lower mass will drag the upper mass until libration occurs. This results in momentum transfer through the tether from the upper to lower mass as tension. The off-vehicle attitude is caused by the Coriolis term of the acceleration expression creating libration of the system as the tether length is changed (fig. 143). The Coriolis force is perpendicular to the Earth's radius for radial deployment/retrieval.

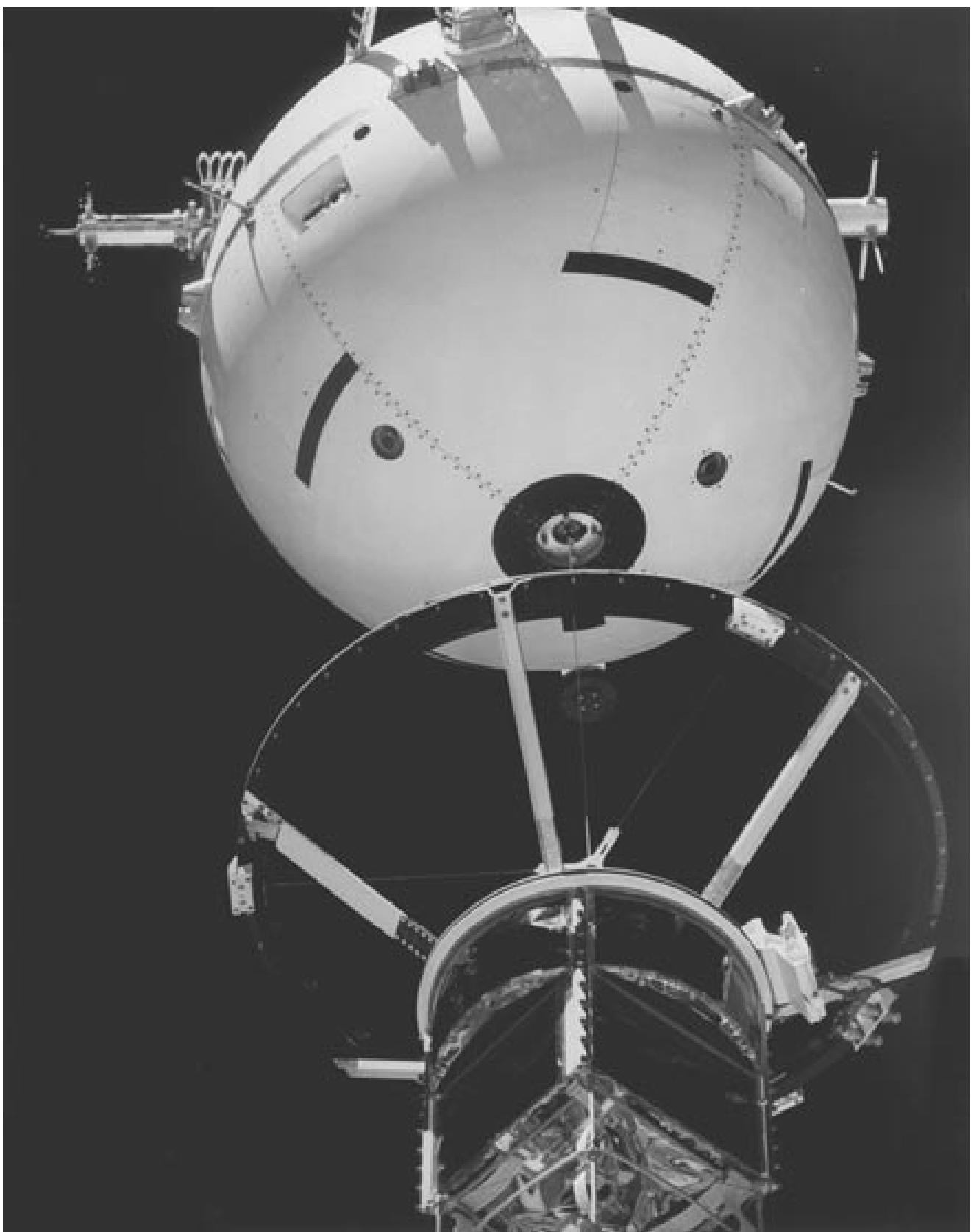


Figure 136. TSS-1.

Deployer-Satellite-EMP Pallet Interfaces

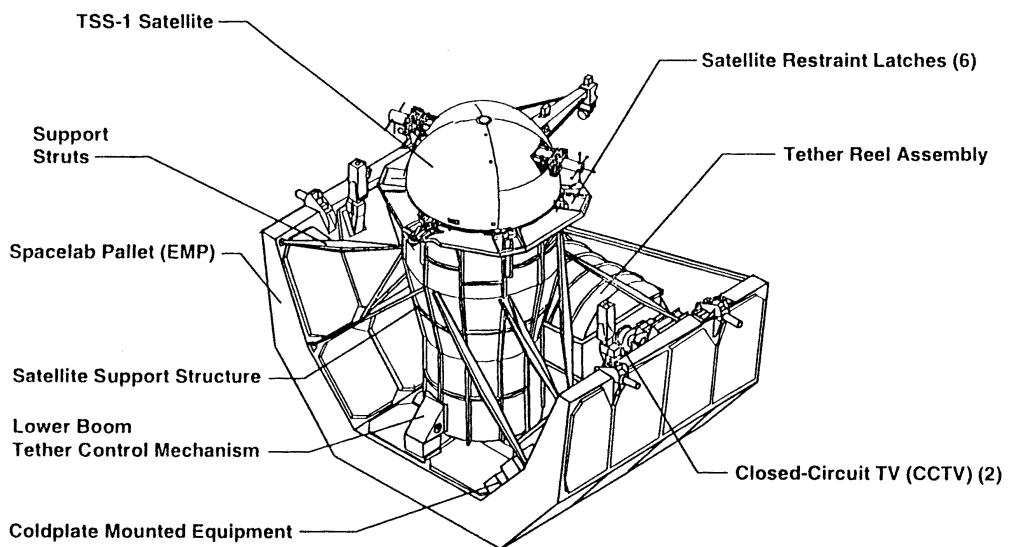


Figure 137. TSS-1 deployer mechanism.

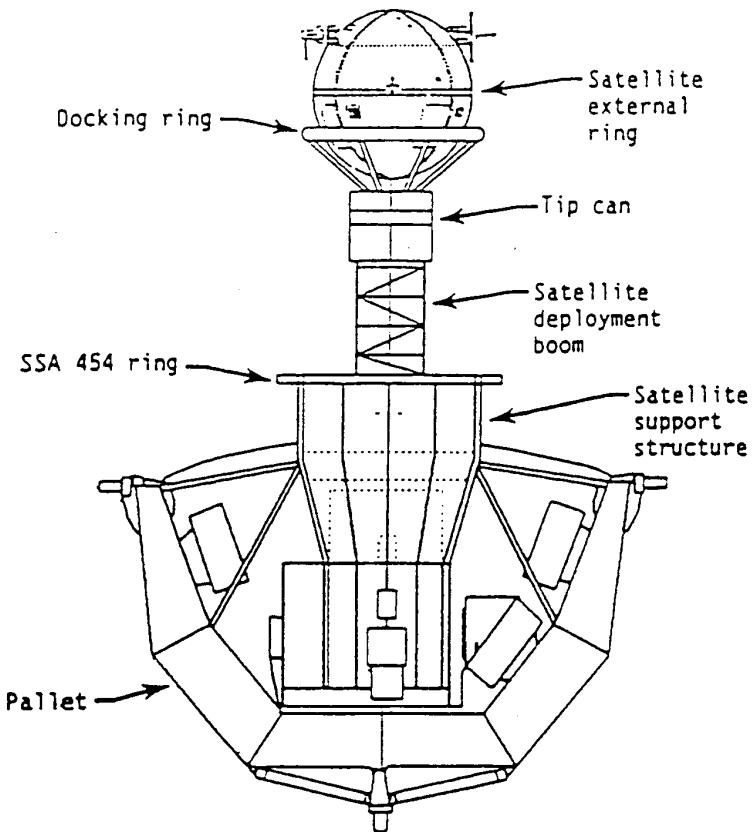


Figure 138. TSS-1 in cargo bay partially deployed.

TSS Satellite Deployment Scenario

Typical Electrodynamics Mission Scenario

- Orbiter Attains Approximately Circular 296-km (160-nmi) Orbit
- Unlatch Satellite and Deploy Outward Using the 12-m Deployment Boom
- Release Satellite and Control Upward Trajectory Using Tether Reel Motor to 20-km Deployment
- Control Satellite on Station
- Retrieve Satellite, Stop at 2.4 km, Complete Retrieval, Dock to Boom Tip, Retract Boom, and Latch Down Satellite

S-Band Communications & Ku-Band Tracking of Satellite

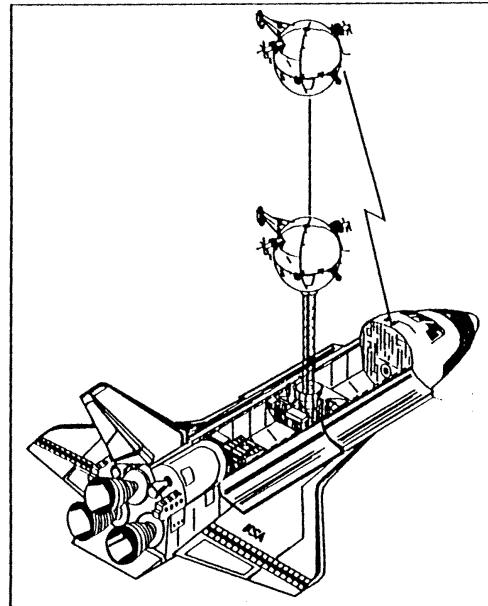


Figure 139. TSS-1 in cargo bay partially deployed.

Tether Control Mechanisms

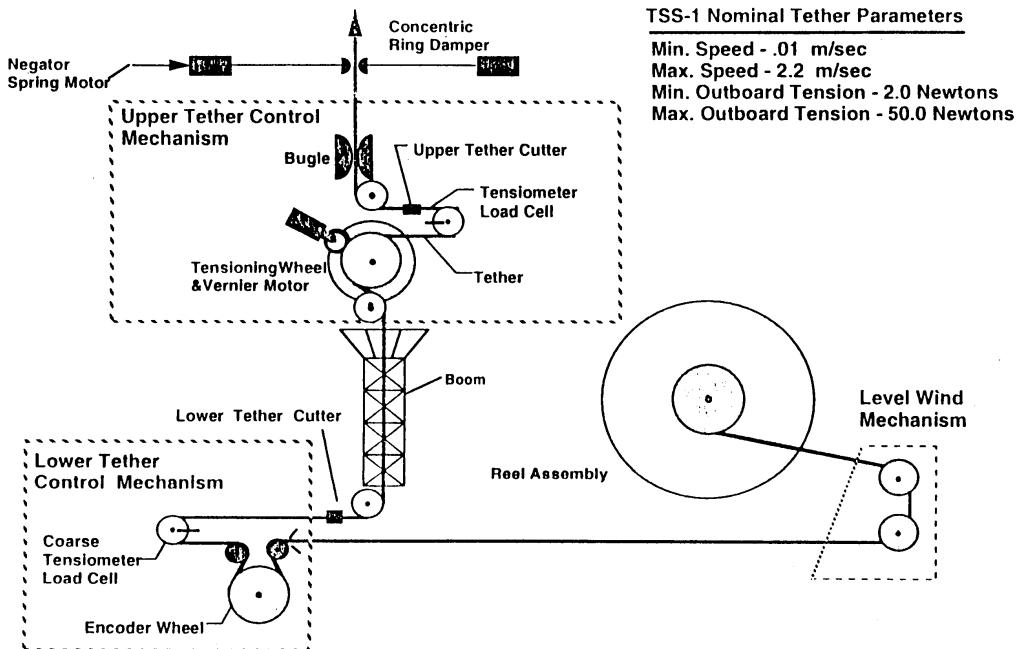


Figure 140. TSS-1 tether control mechanism.

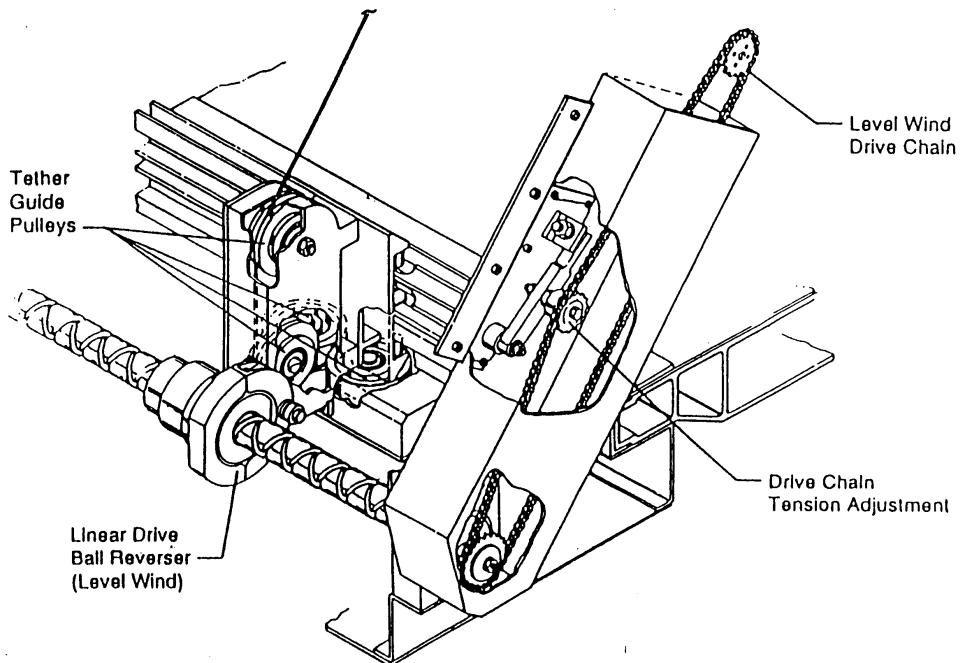


Figure 141. TSS-1 tether level wind.

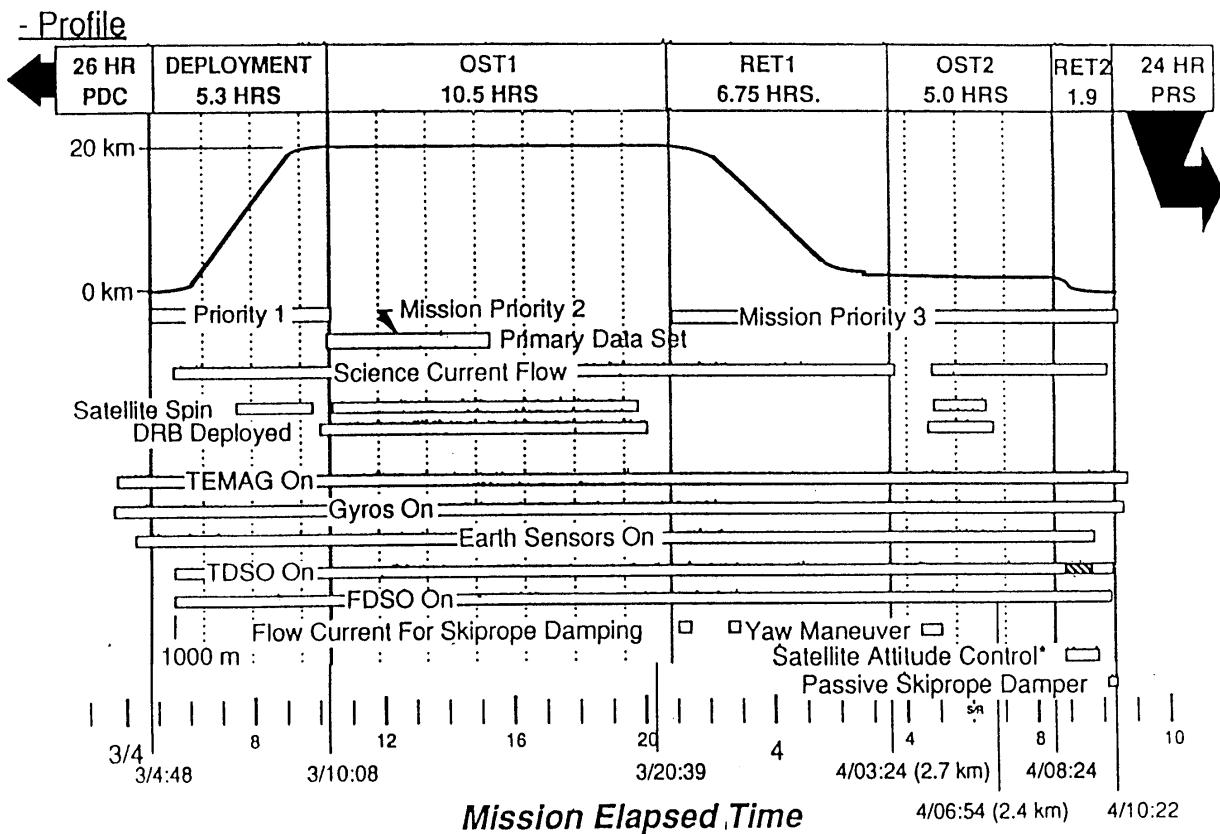
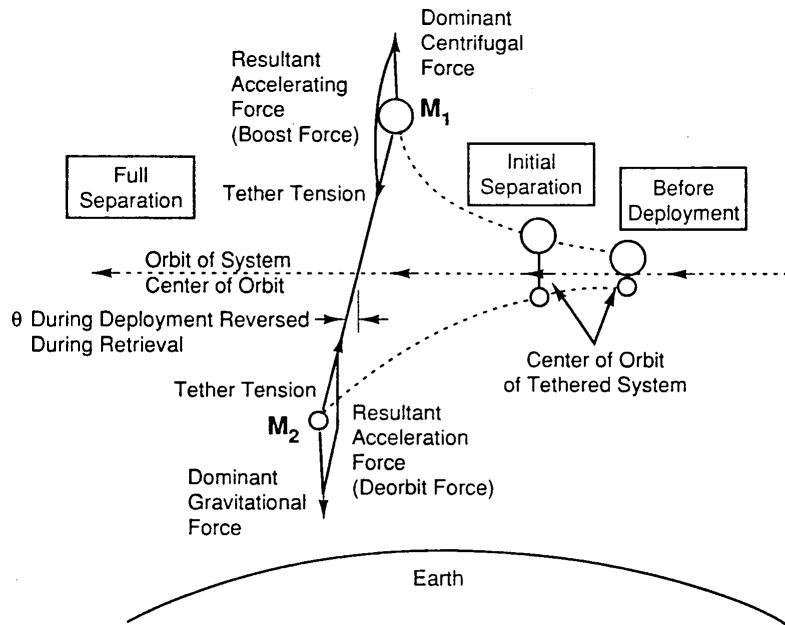


Figure 142. TSS-1 mission profile.



Tethered Deployment

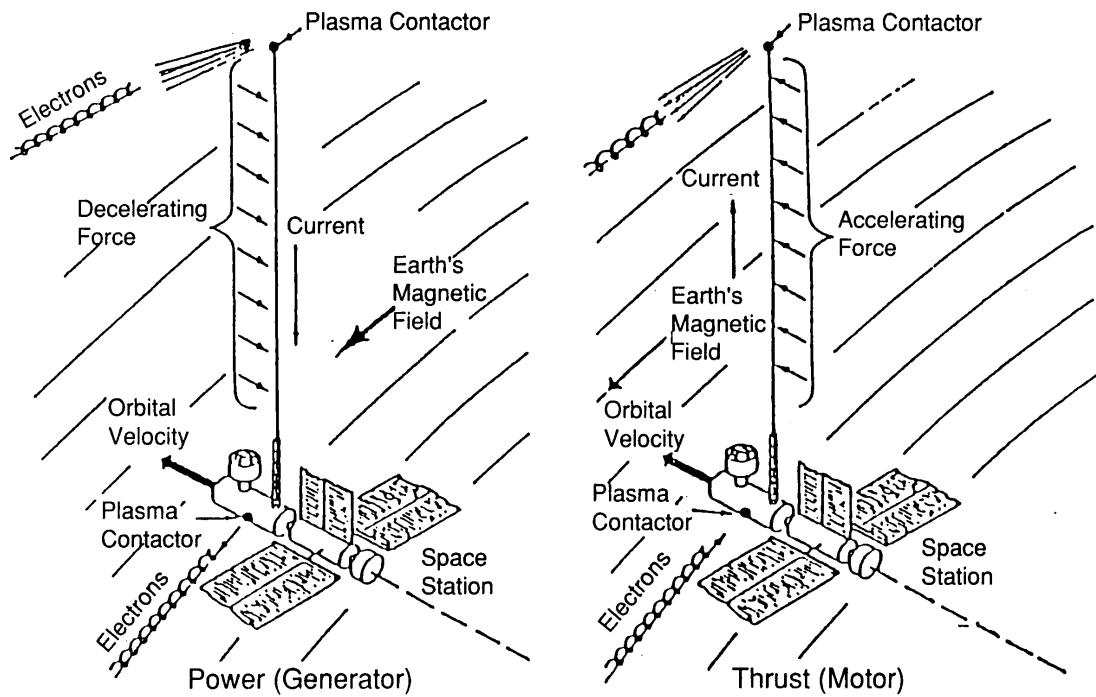
Figure 143. Tether deployment forces.

As the deployed system moves around the orbit with the new center of gravity located on the original orbit, it cuts through the low density atmosphere and the Earth's magnetic field. The aerodynamic drag deflects the tether, as do Earth's magnetic fields (fig. 144). Cutting through Earth's magnetic field causes a current flow, the direction of which can make it a generator or a thruster. Notice also that the two bodies are now in different orbits, thus cutting the tether has an orbit change effect. Obviously, many possibilities exist then for the use of tethers. It can be deployed earthward and used as a sensor in the atmosphere. Sent outward, it can do science. The reader can explore these and many more ideas in reference 73. There were many ingenious ideas used for TSS-1. Since most were used to solve problems of the mechanism or of the system interaction with the environment, they will be discussed in the next section on problems.

3. Problems

a. Dynamics. As discussed above, the orbiter-tether-satellite system interacts dynamically and with the environment, setting up several modes of dynamic oscillation. These modes are shown as if the orbiter was fixed, which is a fair assumption due to the relative differences in mass of the orbiter and satellite (fig. 145). The libration mode can be damped by changing the tether tension with the correct phasing with the motion. The pendulous mode can be damped with the satellite attitude control, if needed. The major problem is the tether string mode governed by the tether length and tension. This mode can be set up by the aerodynamic or electromagnetic forces, the greatest being the electromagnetic. As the tether is pulled in during deployment, due to energy conservation and the tether length changes, the amplitude and frequency increase. As the length shortens to around 600 m, the string mode and the satellite spin mode get in resonance, causing large satellite angular excursions. The results are either the inability to dock the satellite due to the misalignment or the loss of tether tension, hence, control. The basic problem is how to damp the tether skip rope so that there is not much energy available for resonating with the satellite spin motion. This skip rope management problem is

Electrodynamic Tether String Mode Excitation



Although as illustrated the tether forces are in-plane this is not the general case and excitation usually occurs in both planes.

Figure 144. Electrodynamic tether properties.

Phenomenon Definition

Vibration Modes

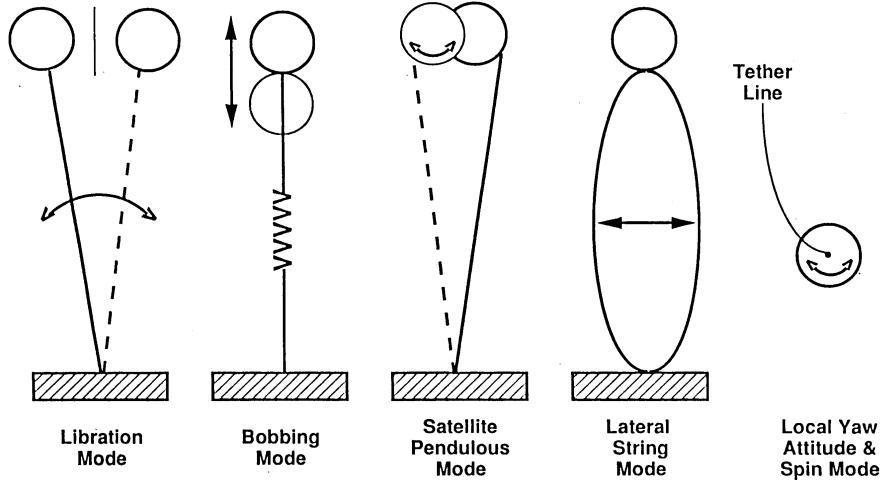


Figure 145. Dynamic modes of a tethered satellite.

complicated by the difficulty of sensing the amplitude, phase, and frequency of the skiprope dynamics. It can be inferred by the dynamics it introduces into the satellite, by visual observations from the orbiter, or by some other end effector sensor. It was decided to use the telemetered satellite control system data (rate gyros output) as the data source to infer the skiprope dynamics. Two ground operations data processing schemes were developed to determine the tether skiprope state. The first was essentially a Kalman filter that worked in the time domain. The second was a frequency domain scheme. Both produced accurate predictions and were used during the TSS-1 mission. The time domain filter was primary, with the frequency domain as backup. Numerous computer simulations were run with known skiprope conditions to verify the approaches. With this ability to determine the skiprope characteristics in real-time, the ability to deal with the skiprope problem was available.

The solution to the problem was then a multifaceted approach for containment of skiprope, producing successful satellite recovery, and the elimination of safety concerns involving the orbiter and crew. The basis of the approach was to determine how much skiprope amplitude could be handled from 600 m and provide a means of damping skiprope below this level prior to entering this period of satellite retrieval. The elements of this approach were: (1) observing the skiprope amplitude and phase using satellite information (rate gyro telemetry data) as discussed above, (2) orbiter yaw maneuver during the second period of science at the 2.4-km distance based on the observed data to reduce the skiprope amplitude below 20 m (the amplitude that could be handled during reel in from 2.4 km), (3) attitude control of the satellite during the resonance period discussed above, and (4) use of a passive tether damper to suppress the skiprope as the satellite gets close to the orbiter.

The passive damper was an innovation within itself. Figure 146 shows the damper installed in the docking ring. It consists of a yoke and ring through which the tether is passed from the reel below and three small negator motors located 120° apart on the docking ring attached with cables to the yoke/ring; thus, as the tether moves, the yoke/ring moves the cables and the negator motors provide damping to the skiprope. The design trick was how to get enough damping without getting so much that the yoke/ring forced a new end point for the tether, therefore, providing no damping. Through much testing, including testing in a thermal vacuum chamber, this design goal was achieved.

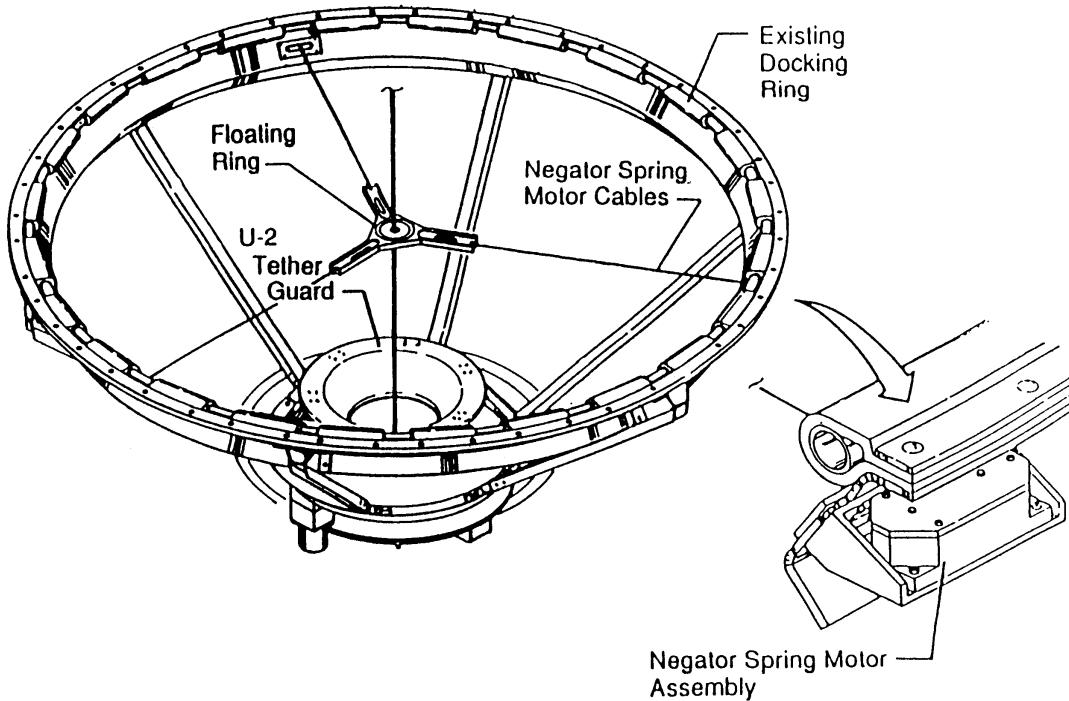


Figure 146. TSS-1 tethered satellite passive damper configuration.

Actual flight of TSS-1 showed conclusively that the multifaceted approach worked with little or no skip rope and successful satellite docking.

b. Tether Hang-Up. TSS-1 was not successful due to tether reel hang-up at 256 m of deployment. Many attempts were made to free the system and deploy the tether, but none were successful and the satellite was retrieved and returned to Earth for assessment and a later reflight.

Tether hang-up was caused by the installation of a larger bolt on the reel attachment to the pallet in order to eliminate a negative structural margin of safety for the shuttle lift-off loads. The larger, longer bolt protruded into the path of the level wind mechanism, catching it and locking it up in the deploy direction precluding further deployment. A simple change, not verified for its system effects, caused the loss of a potentially outstanding mission. The message is clear: changes must be verified for their effects.

Other problems occurred during development, but they centered around the typical problems of the development of mechanism and normal hardware and electrical fit problems. The development of mechanisms of deployment provided the greatest challenge. The deployable/retrieval truss technology was well developed by the time of TSS-1 and proved to be no real challenge.

L. National Launch System/Space Transportation Main Engine (NLS/STME)

Although the NLS/STME program was canceled, it was an early attempt to work a program using Government/industry product development teams and other changes in the contractor's relationship with the Government. The program was, in reality, a joint NASA and Air Force venture where NASA was the basic program manager for the engine. All this restructuring was taking place at the time of the peak of the Deming movement on quality in the United States and was influential in the approaches used for this program.

Initially, the program had a major concern over the NASA-leveled criteria and requirements. In order to deal with this issue, the first thrust was focused on the STME. The STME was to be developed by a contractor consortium composed of Rocketdyne, Pratt & Whitney, and Aerojet, with oversight by MSFC, using a chief engineer and a project or program chief. The contractor-led group was located in Huntsville, near MSFC, with the basic work accomplished at each of the contractor's plants. The contractors not only had the management of the total system at Huntsville, but had a manager at each plant. In order to efficiently design and build an engine in this manner, the engine was divided by components and assigned to the individual contractor's plants. Since the basic criteria and requirements came from Marshall's Science and Engineering Directorate, it was decided to form a team composed of the contractors' leads and the deputy directors of Marshall's Science and Engineering Directorate. The team was led by NASA. The team started the process by visiting each contractor's plant and discussing the program philosophy and the impacts of the current NASA criteria, etc., on program cost, etc. As the team progressed in this activity, it became clear that the requirements and criteria, to some extent, depended upon the management approach chosen by both Government and industry. The team spent a major effort in setting up this approach.

The basic philosophy chosen was the institution of joint Government/industry product design teams basically chosen for each engine component, plus an engine system and engine team. The teams were to be pure design and manufacturing teams responsible for the product, including cost and schedule. There was to be no distinction between the Government and contractor members, where each worked their various tasks and together decided key issues. A steering committee was set up to guide the program and resolve issues, and the program proceeded.

With this approach in place, the team could move into working requirements and criteria. Having Government specialists as working members of the PDT's allowed sustainable reductions in

documentation requirements (150 reports reduced to approximately 20). The criteria were worked separately using specialist teams of Government/contractor members. The guidelines given to the teams were that the criteria contain no “how to’s” and be restricted to critical mandatory criteria to ensure safety. The criteria were not to determine the design solution, freeing the design teams to do good engineering without undue constraints. Criteria were reduced by approximately 80 percent.

At the time of program cancellation, the teams were working and some development hardware was manufactured. Although this first approach was not perfect, it set a precedence that later programs have improved upon.

IV. CONCLUSIONS

Capturing what studying problems and experiences have revealed into a summary is very difficult; therefore, it is with some trepidation and risk that this necessary task is accomplished. Conclusions are in the eyes of the beholder. Each should feel free to draw their own conclusions from the text. In general terms, each separate set is valid and should be attempted. Woven into this fabric of “lessons learned” is the naturally evolving formation and interaction of the design, analysis, and science groups in a maturing technical organization. Central to the development of a product is the creative design engineer, whose primary function is consumed with the work ideas and who is focused on the advanced product, and its performance and reliability in all the interacting disciplines and the robust system.

Most of the time, the product performs well as designed, and sometimes it works adequately for the wrong assumptions and phenomena, usually under less sensitive or demanding environments. However, where the appropriate emerging environments trigger the unsuspected phenomena that lead to failure (*Challenger* SRM O-ring, SRB ignition overpressure, SRB aft skirt failure, etc.), experts are drawn from the scientific staff and institutions to complement the design core analysts groups to resolve the problems and establish another lesson learned.

What this implies is that a different mind set and skills are required for the different jobs in the process, from science to design to problem solving. Universities have moved away from this understanding and now depend too much on the computer/analysis without putting the students into the real world in order to develop the intuition required for design and problem solution. Students need to have experience with hardware failures. Ferguson, in “Engineering and the Mind’s Eye,” an MIT publication, has an excellent discussion on this viewpoint.

The author challenges each reader to compile their own valid set, corollaries, and derivatives.

1. Engineering has evolved from an overall system design performed by a chief designer in general at a drafting table to a design of specialization using large computers and work stations. This is accomplished through compartmentalization, first into subsystems, elements, and components. Then, each of these is separated out by disciplines. As each task by discipline is accomplished, it is integrated into the component, then the element, the sub-system, and finally the system (fig. 147). Looking at this evolved process, several lessons are obvious:
 - a. The integrity of the product is in the system, since nothing exists in isolation and only performs in the system as a system. “Now whenever a change is made, by the passage of energy, and a result is left, it takes place in a group of things,” David Pye.
 - b. The lack of a system focus is the main cause of loss of product integrity (problems) and is not the result of missing technology.

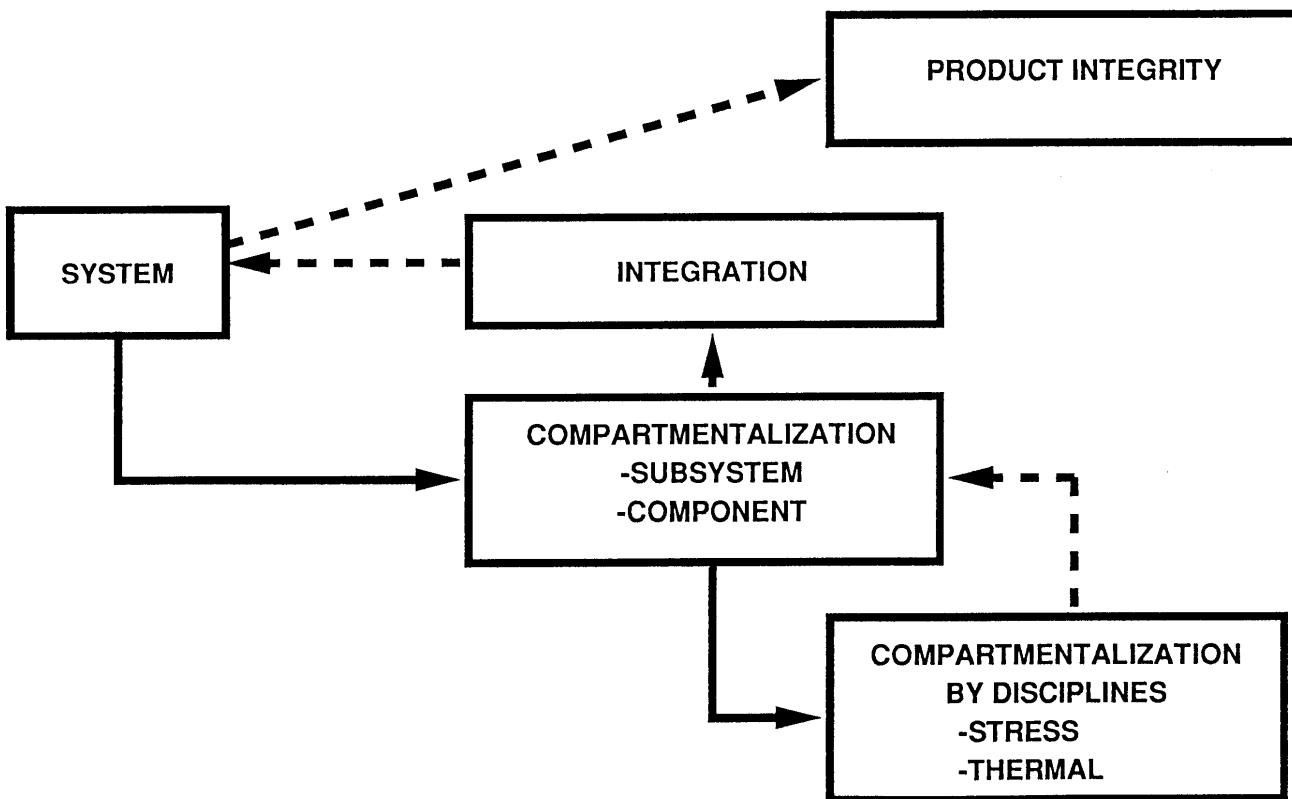


Figure 147. Design process.

- c. The need is for the indepth specialist, with a system focus/viewpoint, to parallel with a sound approach to systems integration that starts with analysis and test and ends with hardware. The paradox faced is: how do you maintain the specialization yet be a generalist and have a system focus?
 - d. The large volume of data/information generated requires the collapsing of the data into general physical symbols so that understanding is ensured. Corollary computer simulations must have the ability to go from wide angle (general) to telescopic (details) when required. Example: general stress field versus stress concentration at a weld or flaw.
 - e. Metrics for the system design must be established that include cost, performance, operations, manufacturability, and reliability.
 - f. Design is a balancing act between conflicting requirements. You must take some of what you do not want to achieve approximately what you want—always a trade that involves risk, complexity, cost, performance, and verification.
 - g. Senge's fifth discipline⁶ emphasizes the learning organization and the importance of systems, two very important parts of any successful organization and product development. In other words, nothing operates “on an island”; it reacts and causes reactions, all of which must be understood and balanced. We design systems based on our understanding of specialties and how they interact together.
2. Insurance for survival of a product, program, or organization depends on its ability to develop a three-pronged interrelationship focus consisting of people (skills), tools (includes facilities), and technologies. It is, therefore, a learning organization that balances the three

aspects, where problem history analysis becomes the fulcrum of the process. Taking away or minimizing any one of the three prongs creates instability, ensuring the long-term failure. Taking away the fulcrum takes away the basis for any insurance.

3. We must be constantly open to the challenges of the future, i.e., to changes and new technologies that bring about change. These challenges are not only the bevy of technical mountains to climb, but must include processes, management, etc. Our openness to these challenges implies that we must identify profitable changes and aggressively accommodate them.
4. Successful engineering, organizations, and design are based on the quality of the people involved—personal competency. Every person individually and every organization collectively must be continuously growing (a learning organization). The message is two fold: (a) search for opportunities for growth and (b) treat every experience, good or bad, as a challenge for growth. Every situation contains those potentials. Look for them. Personal entrepreneurship is, therefore, the hallmark of nobility for successful design and organizational growth. It involves empowerment, trust, constant training, and a learning environment that develops the total person.
5. Criteria or legal requirements must be simple, concise, and direct, providing order to the engineering, but not overpowering to where they stifle creativity and remove responsibility (Dupree). Criteria cannot replace good engineering, they should be quiet aids.
6. Models are just that, models, and are only as good as the assumptions used.
 - a. All analyses are simulations that are not complete (limited), which attempt to predict trends and what will happen. Models are not, therefore, exact representations of physicals.
 - b. A test (components, subsystems, systems, scale models, etc.) only partially replicates real-life situations. It is biased by the limited insight and provisions of the test engineer. You generally get what you are looking for. Improper boundary conditions can invalidate a costly test.
 - c. How you put the analysis and test together in a cohesive unit to get a validated design is the major problem engineers face.
 - d. Computer/codes are used as a cookbook and not as an extension of the human mind—a major error. Cookbook approaches save a lot of time and energy, but can and do miss the phenomenon. They are only aids to complement the human mind.
7. What you do up front to understand the system determines the product's performance and quality. Poor concept selection can never be righted with outstanding design engineering (Pugh). Spend the money up front and get the right concept, then good design engineering is effective.
8. To ensure that design is practiced effectively and efficiently, it is necessary to utilize techniques that enable the team to operate the design activity. According to Pugh, the designer's tool kit must include:
 - a. Independent analysis techniques, modeling, etc., applicable to any project or discipline
 - b. Dependent techniques and technologies/knowledge to include:

stress

thermal

dynamics

fluids

control and avionics.

9. Leadership is the key to the future of any project/products' success. The same is true of organizations.
 - a. The metric of leadership is the individual and collective growth of its people
 - b. The leader is a servant (Greenleaf, Covey, Dupree, Senge, Peck)
 - c. The people create the leader and give him the power
 - d. Leadership is doing one action at a time
 - e. Leadership interests, what he spends his time on, and the rewards passed out determine what he gets.
10. Teaming is an effective approach to ensure project/program success.
 - a. The team can perform little better than the skills of its people
 - b. "The tendency of a team is to perform at the lowest level of its members. Leadership and synergy can raise its performance above the level of the best team members," Senge. There is no one best teaming approach. The team approach selection must match the tasks.
11. The concept selection and design process is key to a quality product:
 - Progressive, convergent, increasing in depth concept selection
 - Sensitivity and trades to identify key parameters
 - System focus
 - Interdisciplinary analysis/design.
12. Requirements are the mantel that determine the product's characteristics, good or bad.
 - Must be challenged with minimum constraints
 - Minimum change
 - Minimum required to maintain order
 - Leave room for creative design (do not dictate design).
13. Models/analysis/simulations are virtual reality. Reality is in the hardware/system testing. Read it; the hardware contains the answer.
14. Lessons from the past contain the keys to the future, if used appropriately. Used indiscriminately as points of departure for extrapolation can lead to major problems.

15. It is prudent to design for fatigue and fracture control, especially for reusable systems:
 - Two different design points
 - Materials characterization at right temperatures necessary
 - NDE must be quantified
 - Eliminate dynamic tuning as a fatigue mitigation technique
 - Proof testing has merit if properly used.
16. All designs must have a system focus. Discipline focused skills must be sharpened and honed, but never in isolation:
 - Technical communication
 - Teaming
 - Interface control documents (ICD's).
17. Design for instrumentation during development, then test to off-nominal conditions and then to failure.
18. You must constantly be aware of, and guard against, dynamic tuning between structural, fluid, mechanical, electrical, and acoustical systems.
19. Rotordynamics has been a major design issue. Attack it up front and design for it:
 - Know environments
 - Model in detail
 - Test structures dynamically
 - Plan for damping augmentation
 - Pursue various options for rotor support systems.
20. Cost, operations, manufacturability, and supportability must be a fundamental part of the design equation. Metrics must be developed to support:
 - Cost and operational efficiency requires a multipronged approach
 - (1) Calculated risk-taking/management that obviously includes a detailed assessment of consequences must be a part of this risk-taking
 - (2) Product improvement that reduces cost and improves operations; reduction of the number of parts, simplicity, and robustness are parts of this equation
 - (3) Development of criteria and requirements that neither over-specify nor dictate the design solution.

All this must be accomplished without violating basic physics and while using good engineering practices.

21. There is no small change. Effects of changes usually have large magnification factors.
22. Failures will occur during design, development, and operations. Through the use of concurrent engineering teams composed of a cadre of all the potential specialists involved, including manufacturing and processing, and using a formal fault tree approach (fishbone works just as well), the probe causes and solution can efficiently be found. All organizations should be trained early in both teaming and fault tree analysis.
23. Regardless of the technical adequacy/integrity of the answer, it always has to be sold politically; therefore, it must become part of the design culture either by changing the culture or the recommendation. In other words, the proposal must always be politically viable. Engineers, therefore, must also be salesmen.
24. Quality must be built in, not inspected in. Inspections are only for insurance; therefore, quality must be a mainstay in the design team's discipline makeup.
25. Understanding and using statistics/probabilities approaches are fundamental for conducting studies, evaluating data, manufacturing hardware, understanding and quantifying sensitivities, and performing operations.
26. Maintaining electronic data bases of analysis, tests, and flights are necessary and should be planned for. Electronic transfer of data and networking is necessary in today's multilocation workforce. Communication is aided greatly by these approaches.
27. Design of experiments is an effective tool for both analysis and test planning and should be in each engineer's tool kit.

It would not be appropriate to conclude this report without making some statement as to the future challenges that face space exploration and what some of the present technology focus implies. The biggest challenge facing space exploration is, "How do you effectively consider cost along with performance and reliability as design metrics?" The biggest problem is obtaining the cost metrics so that appropriate trades can be made and a true optimum system designed. This is complicated by the fact that part of the cost metrics indicates higher performance and fewer launch stages, which greatly complicates the equation since higher performance indicates greater sensitivities to uncertainties that include, but are not limited to, manufacturing, environments, operations, and evolving requirements. For example, a single-stage-to-orbit (SSTO) launch vehicle must balance three different efficiencies to narrow margins if it is to meet both performance and cost goals. To a large extent, meeting these is a paradox in that to get one will result in a costly change in another. The three keys are: (1) a highly efficient propulsion system that not only includes Isp, but includes thrust and thrust-to-weight; (2) structural or dry mass efficiency (this is not only the basic structure, but includes TPS, components, avionics, etc.); and (3) to manage the losses (managing the margins) that includes technology readiness degradations, environments surprises, etc. Past launch vehicles had many impacts due to not properly managing the losses, resulting in load increases, inefficient trajectories, etc. Managing the losses effectively is a major challenge. This means that sensitivities must be well understood, traded, and managed. For example, high acceleration during the near-Earth phase of flight is efficient for managing gravity losses, but creates unacceptable loads for the structure and for the humans. True optimum trajectories create large aerodynamic loads, which are not acceptable. Load relief control reduces these loads, but introduces path deviations, thus performance loses. Then when the vehicle moves back toward the optimum after leaving the high-density part of the atmosphere, large angles of attack are introduced, increasing thermal environments. Leaving the Earth's atmosphere, Isp is important for propulsion system efficiency. Clearly then, understanding environments, sensitivities, and trades from a system viewpoint is mandatory in order to meet the challenges of the future. The same types of things are true for satellites, upper stages, etc.; however, the tent poles vary and are considered different parameters.

The hypersonic aerodynamic vehicle is getting a lot of emphasis and is a clear challenge of the future. The challenge basically follows the areas discussed in the proceeding paragraphs.

Delivering a cost-effective, minimum operating, intense space station is still in front of us although much work has been accomplished. Much fine tuning and effective procedures still must be developed.

Payloads, satellites, science, upper stages, etc., face the same set of challenges—reduced cost while maintaining performance. This is very difficult for all systems. The challenge is to break the cultural loop, reducing criteria, etc. For example, satellite missions could be structured so that more, but much smaller, systems could achieve the same goals without the big costly one-of-a-kind satellite, where a failure is a tragedy. The challenge is not only to attack how we can build and operate the system cheaper, but also how we plan for achieving science.

Finally, the prime challenge is to get back to the basic that accepts failure as a stepping stone to knowledge and not a failure. Engineering today cannot tolerate failure, which is leading to a protection (low risk) design that in the end will stifle space exploration. Space exploration involves risks, risks that must be calculated and minimized but not eliminated. Probably this is the hardest task engineering faces in the future.

APPENDIX A

The purpose of this appendix is to provide, by projects, a listing of the various problems encountered and understood by the author (not the total set, since they were in disciplines not part of my experience). The problems are summarized in matrix form, giving problem areas, problem statement, problem cause, and problem resolution. In some cases, an outline of the problem matrix precedes it. This brief summary should provide engineers with a starting point for design problem avoidance. It should be recognized, however, as was stated previously, that every design is a design of compromises and a design for failure, hopefully at a predetermined time and condition. Thus, problems cannot be avoided—they are in the nature of design. Over design, as well as under design, can kill. Design is a balancing act.

In addition to the problem matrices, specific lessons learned are discussed and, in some cases, particularly for the SSME, additional discussions of specific problems not discussed earlier in this paper are presented. It is hoped that this added discussion will provide some more indepth insight into the influence that high-performance requirements have on product sensitivities and problem occurrences. As the author has pointed out in other publications, the sensitivity of the system varies nonlinearly with the performance requirements.¹⁻³ The sensitivity is not only to environment unknowns, but includes requirement changes, manufacturing tolerances and mishaps, and hardware aging. The SSME is a great example of an outstanding machine that has high performance. The same is true of the total shuttle vehicle and also for the HST. The key lesson to grasp is the total design process, so that during design and development, shortcuts are not taken that lead to repeating past problems.

I. REDSTONE/JUPITER

The author's scope of activities was much narrower during the Redstone and Jupiter programs; therefore, the matrix of problems is greatly restricted. However, the impacts of these problems on our thinking and lessons learned were probably the greatest of any program. The problems experienced taught some very powerful lessons, including insight into several system interactions.

The structural control jitter that occurred on Redstone during control system checkout, while the vehicle was horizontal in a cradle, illustrates the subtleties of structural dynamics and control coupling. The spacing of the potentiometer was such that the stepwise control impulse tuned with the first bending mode. Dynamic coupling is a major concern in designing space systems. The fix was simple; break the coupling loop either with a filter or use a continuous rate gyro pickup (sensor). Propellant sloshing taught the same lesson in vehicle systems interactions. Sloshing also illustrated the need to have good simple models of the dynamic characteristics.

Redstone and Jupiter reinforced the need to develop technologies, models, and data pertinent to the project under development. No project can work all the potential technologies, etc.; therefore, informed prioritization of tasks is required for program efficiency.

The matrix (table A-1) lists the key problems from control coupling, to aerodynamics, and to thermal protection. The thermal problems were plume heating and the payload reentry, both requiring protection of components and structure from the heating environments. Extensive effort has been spent in the years since Redstone and Jupiter to further define both reentry and plume heating environments and protection systems. The development of the effects of plumes on launch vehicles is a very complex technology, from both an analysis and a testing standpoint. Scaling plumes in test, as well as measurements during flights, is very difficult. Plumes not only produce base heating, but affect vehicle drag, shift aerodynamic distributions, etc. Clearly, individual disciplines such as propulsion system design create system problems due to systems interactions, as discussed above. All successful projects have a good systems focus. Designers and engineers should not only be specialists, but must have a system focus.

Table A-1. Redstone/Jupiter.

Element	Problem	Problem Cause	Problem Solution
Redstone Control	Closed loop vibration of structure/control systems	Rate gyro potentiometer feedback as wiper arm jumped between wires from structural dynamic excitation, drove vanes/rudders, further exciting vehicle structural dynamics	Redesigned gyro pickup to provide a continuous signal
Vehicle	Oscillating shock off the nose cone setup acoustical excitation	Angles of nose cone, etc., coupled with flow field	Designed structure to withstand oscillating load
Aerodynamics	Aerodynamic loads during ascent through jet stream created large structural stress fields	Wind-induced angle-of-attack produced large aerodynamic loads on vehicle structure	Load relief control logic using Edcliff vane angle-of-attack meter to generate an angle-of-attack control signal allowed vehicle rigid-load relief (evolved for Jupiter to pressure sensors then to accelerometers)
Guidance/Control	Performance loss due to wind, etc., induced trajectory drift from optimum	Vehicle attitude control responses to winds, etc., caused deviations from optimal flight path	Developed drift minimum control laws that balanced between attitude control and path deviations.
Jupiter Vehicle	Instability of control system during max “q” caused loss of vehicle	Propellant slosh dynamics coupled with the control system causing unstable feedback and forced by the trajectory tilt step command	Introduction of propellant damping into the propellant tank and base lining of a continuous rate gyro signal pickup
	Loss of control during max “q” due to loss of control signal, loss of vehicle (first flight)	Plume heating/recirculation burned the control signal wires to the actuators, causing loss of control signal	Design and installation of protective heat shield over vehicle base with movable port to accommodate engine gimbal
	Aerodynamic unstable vehicle control without use of fin (aerodynamic compensation)	Control authority requirements were excessive using jet vanes or injection methods	Designed, developed, and implemented an engine gimbal ball to take out the thrust loads and gimbal the engine 7° in each axis for control; roll control accomplished using reaction jet control system
	Payload reentry thermal environment excessive	Aerodynamic heating during reentry	Designed, developed, and implemented ablative covering of the structure, eliminating the problem

II. SATURN/SATURN APOLLO/SATURN SKYLAB

These programs were the foundations of space exploration, ending with the “Moon walk” and the *Skylab* space station. The problem list (table A-2) again is restricted due to the author’s experience level on these projects, but is very significant in terms of lessons learned due not only to the fact it was on the frontier, but also the magnitude and scope of the endeavor. Landing a man on the Moon and returning him safely to Earth was a major achievement.

Saturn clearly emphasized the importance of dynamics, dynamics and control interaction, and systems focus in the design, development, and operations of complex launch systems. Most of the tools and techniques of dynamics have their roots in the Apollo program. The importance of fracture control, particularly for propellant tanks, was realized, with the first use of proof testing tanks starting on Saturn. Buckling of large propellant tanks was brought to the forefront, teaching the need to design for, and understanding potential failure modes, particularly those that have a medium-to-high risk of occurring. Saturn indicated that it was prudent to design for robustness in order to have flexibility during development and operations, as was discussed previously. Although in today’s world of austerity, it probably is not feasible now, Saturn clearly showed the advantage of evolutionary design using prototypes or flight test vehicles to develop and verify critical subsystems, etc., as was done using Saturn I and IB. Saturn was an excellent example of taxing the current program to provide facilities and technologies for future systems. The plume heating technologies, dynamic and static structural test facilities, hot fire propulsive test facilities, finite element structure modeling, modal coupling, etc., are examples.

The matrix (table A-3) contains some of the problems that occurred during the Saturn program.

III. SKYLAB SPACE STATION

The *Skylab* space station was the first long-term American demonstration of humans living and working in space. The human perspective was not only utilized to conduct science, but was the key to saving the spacecraft due to the problems experienced during ascent launch. This was exemplified in several ways by the several repair activities that occurred during the first two missions. The first was releasing the hung-up (undeployed) solar array (the other solar array was lost during ascent) and deploying a Sun shade (parasol) out the small workshop port. Releasing the stuck solar array required several tries during EVA’s, finally cutting the strap that constrained the array with bolt cutters. These activities, along with orientation changes, etc., put the station back in working order and allowed a full duration mission of the ATM and other sciences. On the second mission, a larger more effective Sun shade was carried up and installed using astronauts EVA further enhancing the operational capability of *Skylab*. During the mission, a few other problems occurred, all of which had more than adequate workarounds.

The *Skylab* missions taught many lessons, the most important being “that the ability to recover from major problems is the secret to successful projects.” Human presence in space greatly enhances the ability to recover from problems that occur. The other major lesson clearly was: do not create a gap in the Nation’s launch capability. Since the shuttle had not flown and the Saturn program had been canceled, the orbit decaying *Skylab* station could not be reboosted, although it was still functioning and could have served as an interim station. The robustness built into *Skylab* allowed the potential, even though *Skylab* had been inactive many years.

The matrix (table A-3) lists some of the key problems *Skylab* experienced during development and operations.

Table A-2. Saturn.

Element	Problem	Problem Cause	Problem Solution
Vehicle	Wind sounding balloon was unstable during ascent, creating erroneous data	Balloon was smooth-skinned, creating a type of vortex shedding, inducing unwanted balloon oscillations	Redesigned the balloon to have spikes patterned after golf ball dimples to eliminate the unwanted oscillations
	SI-C pogo	Vehicle longitudinal bending modes coupled with the hydroelastic tank and acoustical propellant duct modes, creating oscillating thrust that amplified the structural oscillations (unstable)	Designed and installed an accumulator on the feed line, detuning the system; no occurrences of SI-C pogo since
	S-II pogo	Same as SI-C pogo; filling accumulator during S-II engine start gave a short-term tuning of structural and propulsion system	Problem was shown to not be damaging; no fix instituted
	Ground winds caused a vehicle oscillation while attached to the pad, creating large base bending moments	Vortex shedding of cantilevered vehicle attached to launch pad	Designed and installed damper near top of vehicle between vehicle and service tower
	SI-C oscillating plume		Lived with problem; no observed impacts
	S-II oscillating shock		No impacts
	SIV-B panel flutter	Stiffness/frequency of panel was too low, causing structural panel mode to couple with aerodynamics	Stiffened panel, raising frequency
	SIV-B propellant utilization (PU) guidance and control coupling	Propellant sloshing gave a dynamic oscillation of propellant level, creating guidance and control signals that feedback, increasing the propellant oscillation	Designed a filter to filter out the slosh dynamics, giving steady-state fuel level signal; due to other problems, PU was eliminated
	Excessive max “q” loads due to wind gust and steady-state winds	Rigid-body accelerometers load relief control amplified gust induced vehicle structural dynamics response; rigid-body wind angle-of-attack coupled with attitude control, creating large aerodynamic loads	<ul style="list-style-type: none"> • Eliminated rigid-body load relief • Instituted monthly mean wind-biased trajectory • Instituted control to augment structural damping in first two lateral bending modes
	Excessive instrument unit deflection at the rate gyro location created control/structural inaction instability	The load path from the service module came in at an angle through the fairings, creating local deflection at rate gyro location, increasing structural gains, introducing instability	Changed location of rate gyro where local deflections were not present
	Hydroelastic (shell mode propellant dynamic coupling)	Inability to model coupling between fluid and structure	Developed finite element model of fluid and structure and verified with dynamic test
	Dynamic test data contaminated by vehicle test suspension system mode	Dynamic modes of suspension system coupled with vehicle lateral bending	Tied suspension cables at the center (2 by 4) to test stand to shift cable frequency
	Simulating free-free conditions for launch vehicle dynamic test	Vehicle constrained at base does not simulate free-free flight conditions	<ul style="list-style-type: none"> • Suspended vehicle using elastic cables • Suspended vehicle base on an oil bearing (free-free lateral modes only)

Table A-2. Saturn (continued).

Element	Problem	Problem Cause	Problem Solution
	Scale model dynamic test mode contamination	Manufacturing tolerances not scaled as vehicle geometry created nonlinear movement in tolerance space	Scaled manufacturing tolerances to same scale at the vehicle geometry
	Complex modes of clustered tanks difficult to model	Individual propellant tanks of cluster dynamically coupled, creating complex load paths and dynamic models	Generated special models that produced individual modes of tank elements with appropriate boundary conditions and then coupled them to system using Lagrangian equations and verified by dynamic test; this was a forerunner of present day model coupling techniques
	SIV-B <i>Skylab</i> fairing failure during ascent damaging <i>Skylab</i> solar arrays	Improper venting of solar array compartment during ascent caused delta pressure buildup that failed fairing and solar array	<ul style="list-style-type: none"> • On-orbit installation of new <i>Skylab</i> thermal shield • Astronauts space walk to free second solar array
	Increased vehicle loads during ascent for Saturn V <i>Skylab</i> configuration	Aerodynamic shape change due to elimination of command/service module, creating big change in aerodynamic distributions, creating large moment	Cleared for launch using monthly mean wind biasing and probabilistic instead of deterministic loads analysis
Engines	Propellant feedline gimbal bellows had a flow instability	Fluid flow across bellows created flow instability (type of vortex shedding)	Installed liners inside the bellows, eliminating unsteady flow
SIV-B Stage	SIV-B shell mode coupling with control	The deflection/frequency of shell mode created dynamic control structural coupling that was feedback, creating instability	Designed and installed a control signal filter, eliminating unwanted shell mode dynamics
	Large fluid motions created at engine shutdown on orbit	Energy conservation from high g to low g greatly amplified propellant dynamics, reducing propellant ullage pressure and potential engine restart problems	Installed large nylon baffles
	Zero-g sloshing instability	Coupling of sloshing dynamics with control system	Tailored control system logic and installed slosh baffles
	No slosh damping in lox tank	Baffle orientation was such that propellant flowed along baffle instead of against it; propellant surface is perpendicular to vehicle acceleration vector (acts as a free-falling object)	Reoriented slosh baffle
	Inadequate slosh model	Slosh model did not contain rotary inertia effect of propellant sloshing	A rotary inertia term to slosh model
	Slosh dynamics characterization beyond computer capability when vehicle dynamics and control were added	Solution of hydrodynamics equations consumed computer capability	Developed mechanical analogy (mass spring damper) to hydrodynamic solution by curve fitting model parameters; verified with dynamic test

Table A-3. Lessons learned.

Element	Problem	Problem Cause	Problem Solution
<i>Skylab</i>			
OWS	Loss of meteoroid shield	Aero protuberance and improper venting of m/shield	Improved systems analysis
OWS	EBW failed functional test	Conformal coating flow caused circuit interruption	Changed to maintain close, tight fit to prevent flow
OWS, ATM	304 CRES corrosion caused CDF end fitting to split	Intergranular corrosion caused by carbide precipitation of the grain boundaries and exposure to moisture	Metal reinforcement added for <i>Skylab</i> ; process changed for future hardware
ATM	False stars for star trackers	Explosive components with metallic name plates and I.D. tapes caused false stars for the star trackers	Material change
OWS	Viton static seal leak	Cold flow plus sterilizing process accelerated the problem	Changed to silicon seals; considered nonoperational environments for materials
OWS	Ice bridge from liquid dump	Liquid contacted another surface before the liquid changed to ice	Do not let liquid contact another surface before phase into ice
MDA	Moisture leaks in payload shroud	Overpressurization purge is inadequate to prevent rain leaks	Payload shrouds must be water tight
OWS	Paint debond	Wear and flaking	Use adhesive-backed materials or permanent finishes
Airlock	Coolant leak	Mechanical connections	Weld or braze connections
Airlock	Reduced heat exchanger flow	Dust/debris accumulation	Incorporate replaceable filters
OWS, airlock	Sticking valves	Particulate contamination	Avoid small clearances, filter placement, use ultrasonic cleaning
<i>Spacelab</i>			
SL racks	Rack STR margins	Load increases	Include good margins
SL avionics air system	Resource demand	Very limited resource	Allow for flow degradation due to filter blockage or schedule time for cleaning
SL module	New hardware stowage	Fit check too late in KSC flow	Fit check with flight hardware prior to flight flow sequence
SL condensate dump	Orbiter wanted to dump with SL system	Not possible since correct Q.D.'s were not stowed	Add correct Q.D.'s to SL or orbiter stowage
SL IML-2	CO ₂ level requirement from principal investigators (P.I.'s) not defined early	Late definition of low CO ₂ requirement	Early definition of requirement important
SL-1	Window cover thermal constraints violated	Crew failed to install cover before solar exposure	Develop better timeline for requalifying window
SL remote acquisition unit (RAU)	Data skips with increased freon temperature	Design deficiency and limit quality testing	Hardware modification
Space Station			
Modules	Condensation	M/D attachments and penetrations	Guard heaters

Table A-4. Space shuttle systems.

AERO	Problem	Problem Cause	Problem Resolution
Aerodynamics	Wing load increased over design; discovered on first shuttle flights	Inappropriate plume simulation in wind tunnel testing led to inaccurate aero distribution prediction	Beef-up wind leading edge Change max "q" trajectory from -2° to -6° Accepted 5,000 performance hit and launch probability reduction Wind constraints Day-of-launch I-load update
	Lift-off acoustic	Payload near the propulsion noise source	Water injection into propulsion system flames
Ignition transient	SRB ignition thrust created overpressure wave that dynamically drive the vehicle to 1 to 2 g's response during lift-off on STS-1	Expanding SRB thrust compressed air in flame trench/bucket, creating overpressure wave	Water in injection Water troughs for blockage (based on 6.4 percent propulsive model hot-fire test) Characterized SRB pressure effects on stiffness, also internal pressure
Loads	High sensitivity of structural loads during lift-off to model and environment uncertainties	Dynamics of four bodies plus payload are such that the elements tune dynamically with low damping, which creates high gains and large sensitivities to frequency content of environments	Dynamic test of elements and total configuration 1/8, 1/4, and full scale 300 mode dynamic lift-off analysis for loads accuracy 2-sigma worse-on-worse parameter variations for design loads 26 discrete design load cases Time delay of SRB ignition to min stored energy condition Staggered SSME start and abort shutdown 100 percent SSME lift-off SRB case elongation characterized
	Element-to-element interface loads sensitive to aerodynamics and thrust vector variation	Aerodynamics and propulsion and control forces created at two points between each element Small variations in environments are reacted through long moment arms	Body-to-aero variations included in design load Structural beef-up in ET
Dynamic	External lox tank buckling	Propellant weight without internal pressure	Beef-up structure through increased skin thickness Fill tank against pressure
	External H ₂ tank oil canning during model excitation	Dynamic test was conducted at room temperature; therefore, SRB to ET struts were canted approximately 7°, introducing roll moment into aft main frame	No fix required because flight will be at cryo temperature; therefore, struts will be perpendicular to tank and main frame
	SRB/ET/orbiter roll modes	The unsymmetrical coupling of the elements through the interfaces creates a rolling motion of the element and pivoting of the interfaces	Detailed dynamic test 1/8, 1/4, full scale Characterization of propellant dynamics, pressure effects, etc. Detailed finite element models of elements modal coupled together
	ET lox tank hydroelastic damping very low at high fill levels	Coupling between fluid dynamics and structural shell modes of tank	Detailed 3-D finite element hydroelastic model generated to mature modal characteristics
	1/4 scaling impacts on manufacturing	Testing in 1-g environments using standard manufacturing problems, creates exaggerated effects of joint slope, etc.	Redefined all manufacturing tolerances for 1/4 scale to 1/4 of standard tolerances
	1/4 scale SRB attach fixture distorted dynamic response	Tolerances of the aft SRB strut to ground fix was not rigid to ground, changing test boundary conditions	Redesigned test fixture to ensure rigidity

Table A-4. Space shuttle systems (continued).

AERO	Problem	Problem Cause	Problem Resolution
Interdisciplinary analysis/interaction	Load relief control impacts performance, thermal environments, flutter boundaries, and control authority	Load relief drifts the vehicle off the optimal trajectory; corrections introduce large heating environments and increases control authority requirements	Tailoring between control, loads, performance loss, and heating environments Gimbal of SRM nozzles for control authority
	Performance loss due to “q” constraints	Meeting loads and flutter “q” constraints requires combination of SSME throttling and trajectory lofting Throttling approximately 25 lb per 1 lb/ft ² of “q” Lofting approximately 275 lb per 1 lb/ft ² of “q”	Partial reduction of performance through adaptive SSME throttling Day-of-launch I-load update
	SRB holdown bolt hangup creating potential lift-off clearance problems and extended hole in the aft skirt	Uneven firing of pyrotechnics of frangible nut caused bolt to wobble and hang up	Tested characterized bolt pyros Improve pyro firing electronics
	Day-of-launch constraints reduce launch probability, causing launch postponement for four shuttle launches	Winds aloft create dynamic pressures and loads greater than vehicle capability	Adaptive SSME throttling Day-of-launch I-load update based on launch minus 2-h wind profile (problem not totally resolved; still a probability of launch hold)
	SRB reentry imposed separation conditions, impacts performance	SRB separation conditions “q” and α creates nonoptimal trajectory	Optimize requirements between SRB reentry and vehicle ascent trajectory Attrition criteria change Parachute size increased
STS-51L, Challenger	Vehicle destroyed during flight due to failed SRM causing loss of vehicle control	O-ring not sealing assembly segment joint due to low launch temperature and distorted case for reuse	Joint was redesigned using three O-rings and a y-clamp tank interface Flight suspended to redesign and verify joint
Orbiter tile	Debris damages orbiter thermal protection tiles, creating potential safety problems and cost refurbishment	ET insulation deadband Runway debris Pad debris Holdown bolt pyro debris	Changed ET bonding and inspection criteria Redesigned holdown bolt container Prelanding pad and runway cleaning
Control authority	Inadequate control authority during ascent using elevons and SSME's only	Aerodynamic/gust response Thrust misalignment Thrust mismatch	Redesigned SRM nozzle to gimbal $\pm 7.5^\circ$ after PDR

IV. SPACE SHUTTLE SYSTEMS

If the space shuttle taught one lesson, it was that the system effects predominated the vehicle characteristics and the problems experienced. Therefore, system requirements and integration are key elements for aerospace systems quality and, thus, success. It clearly taught a second lesson: “that the higher the performance requirements, the higher the sensitivities to system interactions, environments, uncertainties, manufacturing tolerances, and operational complexity.” The matrix of problems, coupled with the previous discussion, should give the reader a hint at these two profound lessons. The space shuttle made very clear that design is a balancing act between conflicting requirements. The balancing of these conflicting requirements requires interdisciplinary analysis and testing in order to properly quantify the key parameter sensitivities and the trade factors.

The use of working groups and panels in a pseudo-concurrent engineering teaming fashion is an efficient means for handling both the discipline technical issues and the systems issues and trades.

Managing shuttle was a major challenge since it involved several major contractors and several NASA field centers. The working groups and teams contributed greatly in meeting the management challenge. Another clear lesson from shuttle is that many development problems can be alleviated with operational procedures, etc.; however, they in general carry a high price tag.

The problem matrix (table A-4) lists some of the key problems encountered and solved. Many have been discussed previously in this paper or in other reports and papers. The missed aerodynamic distribution created problems that must still be dealt with operationally today. The solution to the overpressure problem on STS-1 was major. The sensitivity at lift-off to the release dynamics has impacted not only the vehicle structure, but also the payloads it flies. Future vehicles should attempt to design a release at lift-off that does not store high levels of energy that have to be released dynamically at lift-off. The multibody asymmetrical configuration clearly contributed to this problem.

A. Space Shuttle Main Engine

The SSME, being a reusable high-performance liquid propulsion engine, has taught many valuable lessons. These lessons have been both general in nature as well as very specific technically. The first major lesson is "high-performance requirements drive sensitivities, thus for reusable systems, problems in fatigue and fracture. Fatigue and fracture then must be a major part of the design and operation of these high-performance systems. Welds in these systems should be designed to have lower stress than the parent material, reducing fracture and fatigue concerns. Welds should be eliminated where possible. Stress concentrations should be reduced or eliminated. The design should accommodate NDE as well as repairs. This is particularly true for welds. Materials properties should be well characterized for the operational environments, particularly for fatigue and fracture. Design development hardware for instrumentation so that environments and dynamics, etc., can be verified. High energy concentrations such as turbomachinery are very susceptible to instability problems such as whirl and must have design augmentation to control potential problems. Dynamic tuning between structure, flow, and acoustics is a major source of problems not only in turbomachinery, but also in all high q flow areas and combustion devices. Many of these problems cannot be predicted; therefore, one must be prepared for problem occurrence and have established problem investigation tools such as fault trees available and understood, otherwise problems can grind programs to a standstill. In summary, high-performance liquid engines are very complex, environmentally, dynamically, thermally, etc., requiring indepth technical penetration and superior systems integration to ensure a quality product. Care must be exercised from design, manufacturing, and operations view points. Hot firing of the system is required to understand, develop, and verify these systems.

1. Lox Pump Bearing Stress Corrosion. Stress corrosion is a problem that has plagued designers from the start of technology. The corrosive material starts a crack and accelerates its growth to failure. Since it is a known problem, great care is taken to manufacture and assemble hardware in clean rooms, etc. Drying and cleaning processes are also implemented to reduce the potential of a problem occurring. Even with these special efforts, small processing changes can result in unpredicted failures. Such was the case on the high-pressure lox pump that flew on STS-27.

The SSME high-pressure lox pump is a high-energy (level and density) system that spins up to 30,000 r/min. The shaft has impellers on one end to pump the cryogenic lox. The other end has a dual stage turbine that provides the energy. The shaft that contains these elements is mounted to the housing with ball bearings. The bearings are fitted on the shaft such that there is a constant stress on the inner raceway. This is accomplished by cooling the shaft and then inserting the bearings. The presence of any contaminant with water will start the process. In other words, the triangular relationship between the magnitude of the stress field, water, and contaminant leads to the failure.

The cracked race was found with borescope inspection after the STS-27 flight (fig. A-1). Materials evaluation showed the cause to be stress corrosion. Chlorine was found on the part. The stress field is known within certain bounds. Lab tests have shown that the stress level present was sufficient to crack



HPOTP Turbine End Bearings

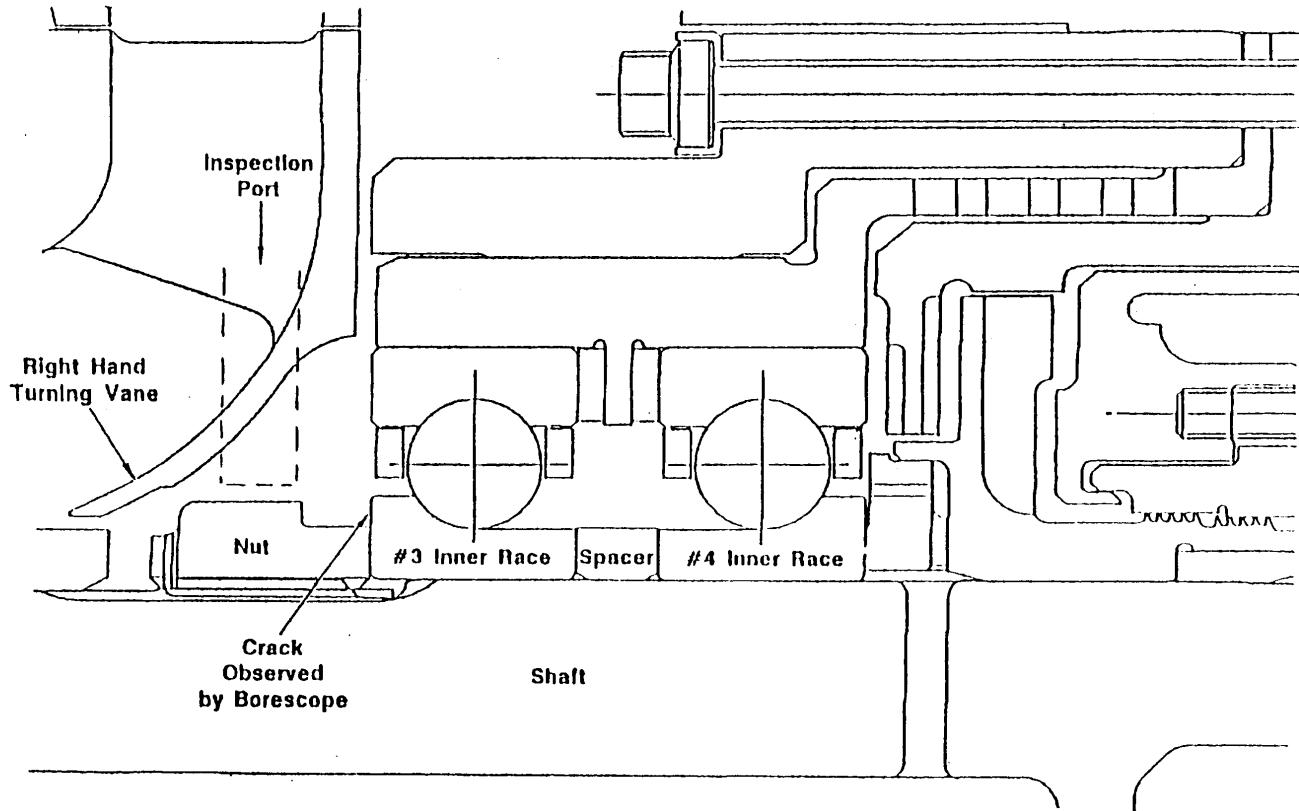


Figure A-1. Bearing failure/stress corrosion.

the race in the time available. The frequency (external accelerometer) response data show that the crack was not present in the pump green run, but was present during the flight. When a failure like this occurs, a detailed investigation must be conducted on all manufacturing, transportation procedures, etc., to isolate a cause.

What had happened was that Rocketdyne, in order to have more capability to support shuttle launches and development testing, built a new pump manufacturing facility for flight pumps. Some changes in the manufacturing process and drying procedures were instituted that were different from those used in the development pump room. The process in both facilities includes super cooling the shaft to get the shrink fits required. On these super-cooled parts, water can be condensed out of the air. This necessitates the drying procedure to eliminate the problem. For some reason, the drying procedure did not work properly, causing the cracked race. Changes have been made in the procedure, with no further problems occurring.

2. Lox Pump Bearing Wear. The original high-pressure lox pump has had a problem achieving the required lifetime before refurbishment. One of the causes for early refurbishment was excessive bearing wear. The specific cause of the bearing wear has not been determined at this time, though most of the parameters that contributed to wear are known. The results of ground testing have shown various degrees of wear and indications of wear (cage harmonics in vibration and strain data), including pump

failure during engine hot firings. The last failure that occurred was engine 02/2 test firing 904-044. The following excerpt from the failure investigation report describes the incident.

"On June 23, 1989, static test 904-044 of space shuttle main engine (SSME), unit number 0212, was terminated by the engine controller when the main combustion chamber (MCC) pressure redline limit was exceeded.

"The purpose of the 1,338-s test was to evaluate the HPOTP block I bearing modifications, HPOTP weld 3 instrumentation and internal instrumentation. The planned test conditions included 215 s at 109-percent thrust, 550 s at 104-percent thrust, 90 s at 100-percent thrust, and 320 s at 96-percent thrust. Then two throttling profiles to 65-percent thrust were planned. Overall engine performance on 904-044 was normal until 1,267 s when a slight downward shift in MCC P_c occurred as a result of downward shifts in HPOTP and HPFTP discharge pressures. Both preburner oxidizer valves opened to rebalance the engine. At 1,270.67 s, a sudden, massive loss of lox flow from the HPOTP caused the MCC P_c to decrease such that the low P_c redline limit was exceeded and the test terminated.

"Posttest inspection and engine teardown revealed mechanical failure of the HPOTP turbine section, rupture of the OPB bowl, failure of the MOV actuator and failure of the PCA. Disassembly of the PBP bearing package showed severe degradation of No. 2 bearing with cage destruction and excessive ball and race wear. The left-hand inducer blades had heavy erosion damage and slag formation on the suction surfaces.

"The most probable cause of the mishap was failure of the PBP No. 2 bearing. Excessive wear of the balls resulted in mechanical damage to the armalon cage approximately 320 s prior to the mishap. The wear of the 440 C bearing and the breakup of the cage continued until several of the balls jammed in the outer race and skidded on the inner race. The frictional heating resulted in molten debris from the balls and races being ejected into the coolant. The high temperature particles were oxidized and forced into the left-hand side of the inducer. The subsequent erosion and promoted combustion on the suction side of the inducer by the high temperature particles created a large gas volume that unloaded the HPOTP. The speed of the HPOTP increased until the first stage turbine disc ruptured. The failure of the disc resulted in the failures of the OPB bowl, the MOV, and the PCA.

"The HPOTP U/N 2216R1 on engine 0212 was a block I development unit. The mode included a thin-bladed inducer, 15-vane inlet, PBP bearing back pressure seal, reduced tolerance on PBP damping seal diameter, ion implanted bearings, elongated cage pockets, and increased coolant flow to the turbine end bearing. The time on U/N 2216R1 PBP No. 2 bearing and its failure mode were consistent with prior bearing failure. Based on the data and hardware evidence, none of the development features being tested were significant contributors to the mishap."

Evaluation of the dynamic data from external accelerometers, and strains mounted on the pump housing and internal strain gauges on the isolator and cartridge, showed a distinct event at 1,240 s, indicating some shift, apparently the cage cracking. At this time, the backpressure around the No. 1 and No. 2 bearings started rising, probably due to bearing residuals (cage and wear), stopping up the orifice until pump failure. On prior tests, at approximately 3,000-s total run time, the cage harmonics started showing on the internal strain gauges. The external strain gauges showed the harmonics around 40,000 s. The 2X cage harmonic was present from the test start and became more and more organized.

Cage harmonics exist because of uneven ball sizes. Theoretically, this is easily shown. The various ball size combinations produce different ball sizes. Though the cage harmonic signature characterizes relative ball sizes, it is also a good indicator of mean ball wear. The presence of cage harmonics in a pump green run test flight is a rejection criterion for flight worthiness. If one is interested in the details of this failure, secure the mishap report.

The excessive ball wear is a major program cost that results from two sources: (1) pump rejection after green run and (2) the limited number of flights during which it is feasible to run a bearing.

Ideally, one should get at least five flights before teardown and refurbishment. Currently, this is not possible. Changes to improve the bearing life are in process. The turbine bearings wear can be adequately accessed after each flight by making a shaft micro travel measurement. The pump bearings (No. 1 and No. 2) are the main diagnostic problem.

3. Engine and POP's 2106 Incident. The combustion system/process has been the source of many problems in liquid propulsion systems. The source of the problems is the big energy source and the extreme environments that, when coupled with small changes, create problems. For example on July 1, 1987, test 902-428 of engine 2106 experienced a premature cut-off after 204.12 s of a 700-s test. The test ended when the high pressure fuel turbopump (HPFTP) discharge temperature dropped below the disqualification limit. Inspection revealed moderate erosion of the oxidizer preburner (OPB) faceplate and a burn through of the high pressure oxidizer turbopump (HPOTP) inlet housing. Leak checking of the OPB found a large leak due to braze cracking around a baffle. Metallurgical inspection of the crack surface showed 10 arrest lines that seemed to match with the 10 proof cycles seen by the hardware. Review of the test history found that this unit had experienced three large lox dome detonations (or POP's) exceeding 8,580 g's peak-to-peak, with the last one happening on test 902-427. Based upon these observations, it was determined that a poor quality braze joint was work hardened by proof cycles. The POP during test 902-427 then loaded the braze joint to failure. POP's are caused during start and shutdown when a mixture of oxygen and hydrogen suddenly ignites (fig. A-2). POP's can be controlled with start and shutdown sequencing, but not eliminated; therefore, the method of brazing has been improved and the shutdown sequence has been changed to reduce the chance of having damaging shutdown POP's. Operationally, POP's are monitored on all engines, thus all cut-off POP's exceeding a threshold limit implement a faceplate deformation inspection and leak check. If damage has occurred, repair or replacement must be made.

Reference: E2106 Incident Investigation final report.

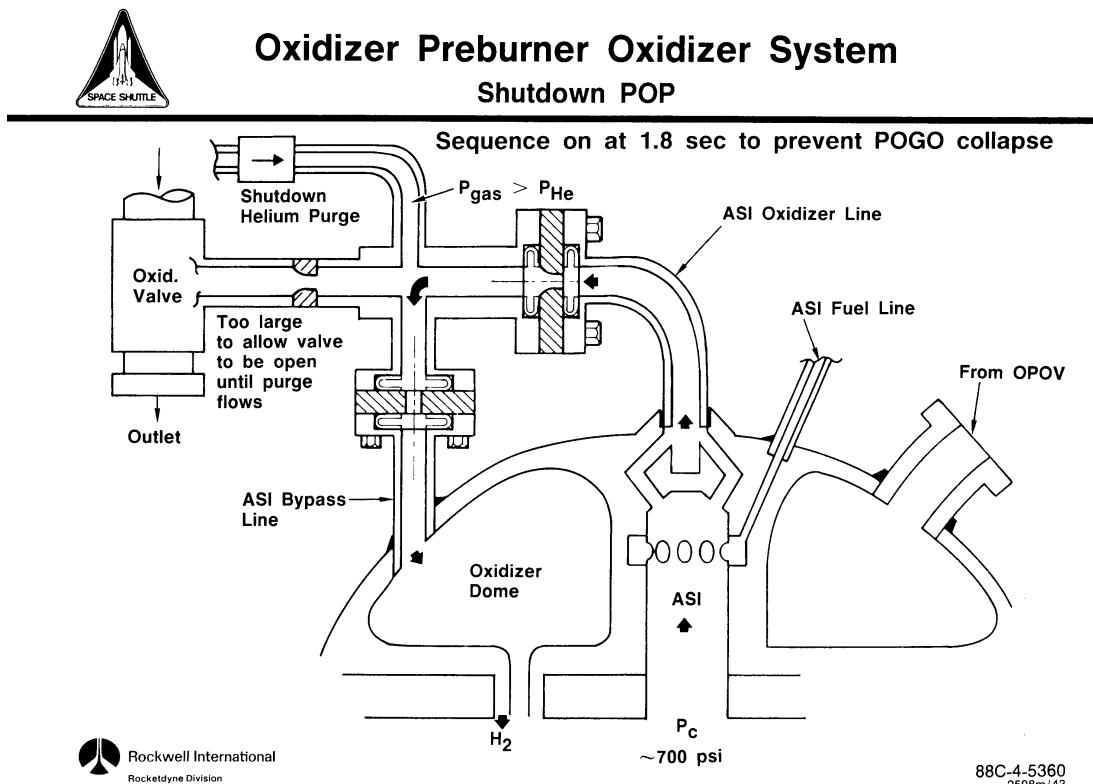


Figure A-2. Shutdown POP.

4. HPFTP Kaiser Nut Failure. The inlet into the turbine portion of the HPFTP experiences hot gas temperatures of up to 1,444 °F. To protect the turbine bearing supports from this hot gas and guide vanes, a flow shield is installed, which is held in place by a nut. When the flow shield is viewed from the side, it resembles a World War I German helmet. Therefore, the flow shield and nut are referred to as the “Kaiser hat” and “Kaiser nut.”

With the introduction of the “phase I full power level (FPL) engine,” three incidents of the Kaiser nut backing off have occurred. The first occurrence was on HPFTP 2008. The nut backed off and the flow deposited it into the first stage nozzle area, with the lock passing through the turbine. Then on HPFTP 9207, the nut and lock came off and the nut passed through the turbine. This time, several first- and second-stage blades were damaged. Finally, on HPFTP 0110, the nut and lock again came off and were found in the first stage nozzle area.

Instrumentation showed that the area of the nut was approximately 500 °F hotter than in the FPL engines. Analysis found that the nuts were failing in phase I engines due to the increased temperature. Various nut redesigns, locking features, and ceramic coatings were investigated. Failure of engine 2013 was caused on the first use of a redesigned nut, which allowed a leak path for hot gas to reach the turbine end bearings.

Ultimately, the problem was fixed with, in addition to a redesigned nut with no leak path, a different engine start and preburner sequence. This sequence, certified in 1982, significantly lowered the temperature flowing into the turbine inlet and onto the Kaiser nut.

5. HPOTP Capacitance Failure Incident. A premature engine cut-off occurred at 41.80 s during test 902-120. A fire was observed in the HPOTP area. The failure investigation identified the fire origin in the area of a capacitor device. The initial mode of failure was thought to be yielding and/or structural failure of the capacitance package, which caused rubbing in a lox environment. In the early phase of SSME development, this special capacitor device was installed to measure the bearing loads and the rotation speed of the HPOTP. Since this device was initially intended to collect data for a better understanding of the bearing package area, its further use was discontinued. The message is clear; intrusive instrumentation in lox environments must be designed with great care. Technology is still needed for measuring the motions of rotating turbomachinery parts.

6. Engine 0213, G-15 Crack. Joints between parts of a propulsion system, especially the engine, are a constant concern and source of many problems. The joint between the nozzle and main combustion chamber (MCC) of the SSME is an example. Upon disassembly of SSME 0213 the MCC and the SSME nozzle, joint G15 was exposed, showing the G15 seal to be buckled and cracked due to overheating. This occurred after 34 tests and 15,114 s of hot-fire time (fig. A-3). Also, three of the bolt holes on the nozzle were found to be cracked. However, a G15 joint leak-check done after the hot fire did not identify a leak. The MCC-to-nozzle combination on this engine had the maximum effective tube protrusion, which caused the nozzle coolant tube crown to erode and the flow restriction inhibitor (FRI) to deteriorate. The tube erosion, together with hot fire time, contributed to tube leaks (fig. A-4). Although tube erosion, leaks, and FRI deterioration were noted during the 34 tests and bluing and/or cracking of the seal was expected, the extent of the bluing and cracking was a surprise. Aerothermal analysis found that the high protrusion and cluster of vented or leaking tubes created an artificial (hydraulic) protrusion that directed core gas into the cavity. Discoloration of the G15 sealing surface indicated exposure to hot gas of approximately 2,500 °F. A similar occurrence was noted on engine 0209, which had 0.077 in of effective protrusion and three leaking tubes.

Based upon ground test experience with varying degrees of protrusion and/or gap, a life limit was placed on the hardware to assure a margin against the seal cracking. All life limits maintain a factor of two for each combination of protrusion and/or gap tested on the ground. Additionally, all flight engine G15 seal areas are inspected for the integrity of the FRI. If any FRI degradation exceeds the empirical data base, the nozzle is demated and the sealing surface is inspected. Currently, with the

improvement of nozzle coolant tube stacking, a need for thicker, denser FRI material is being investigated to further decrease the effect of the hot gas leak in this area (fig. A-5).



ENGINE 0213 G-15 CRACK

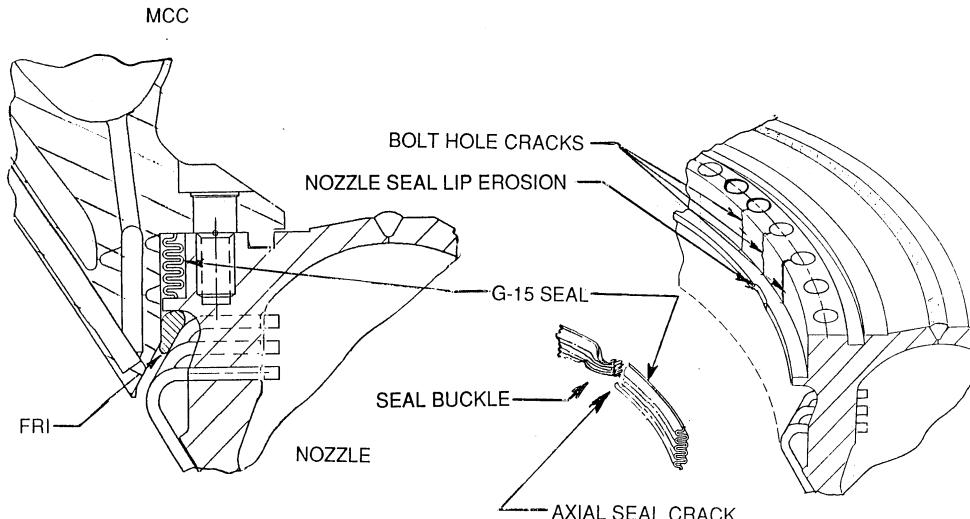


Figure A-3. G-15 configuration.



ENGINE 0213 G-15 CRACK

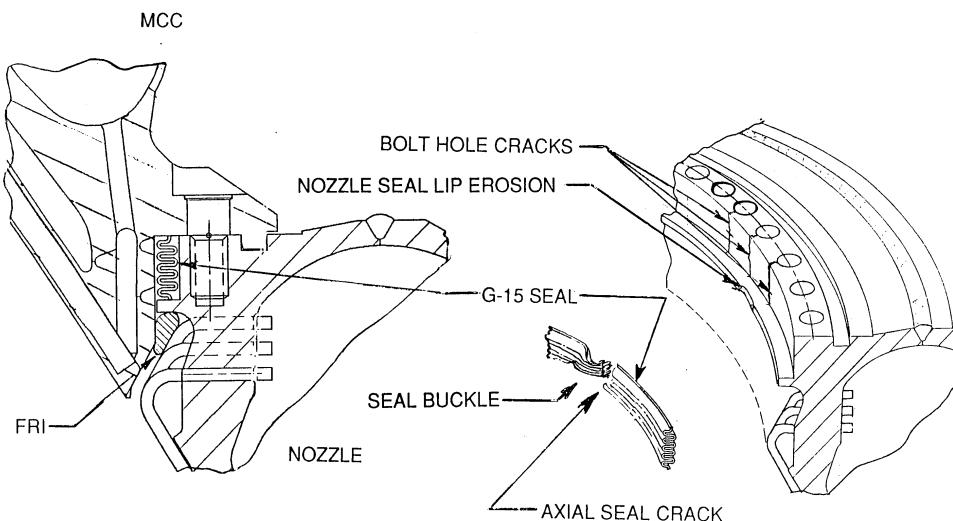


Figure A-4. G-15 nozzle erosion.

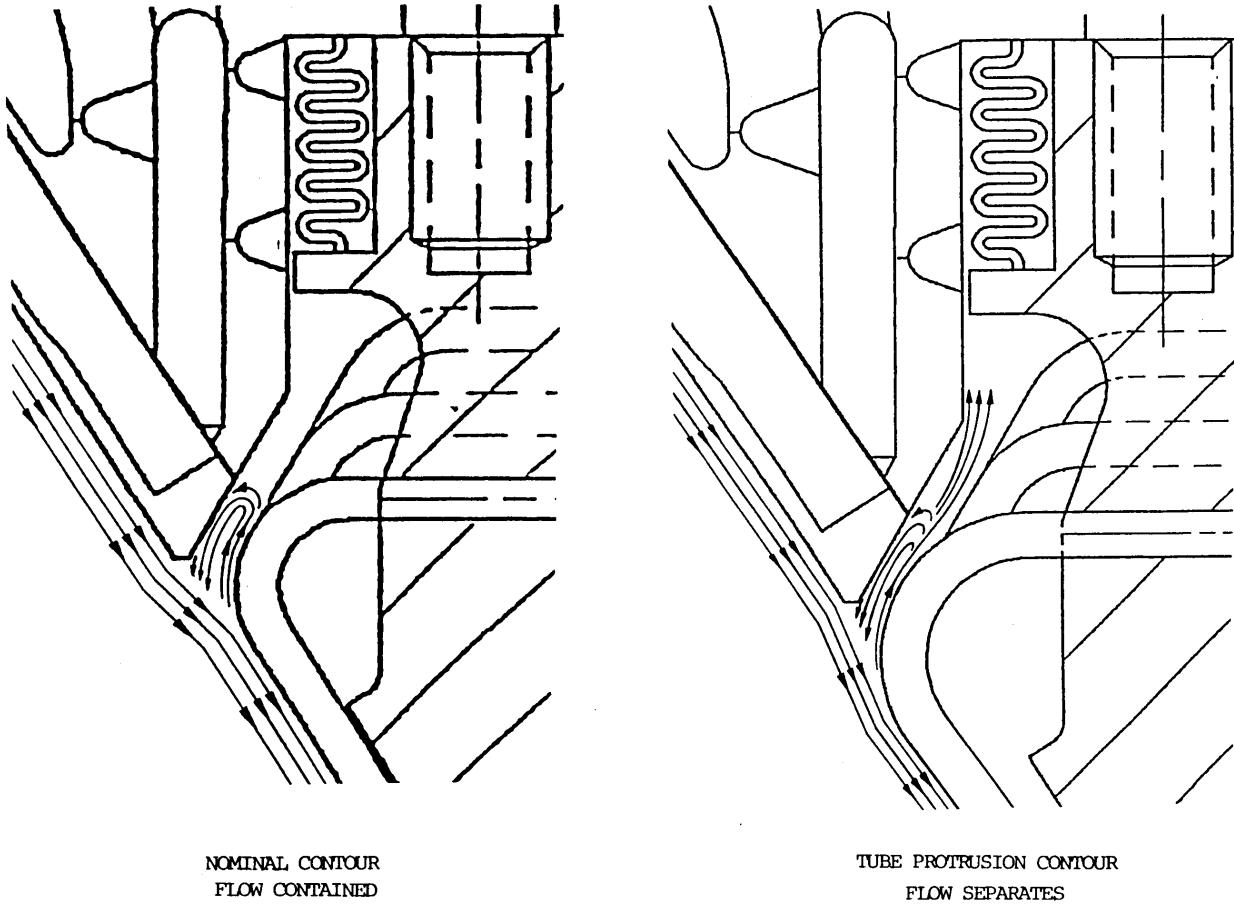


Figure A-5. Recirculation heating gauged by tube protrusion.

7. Fuel Preburner (FPB) Lox Post Failure. As was discussed earlier, combustion devices are a concern. The lox posts that carry lox to the combustion chamber are typical of a class of problems that occur (figs. A-6, A-7, and A-8). Two incidents of fuel preburner lox post failures have occurred in the history of the SSME test program. Failure investigation of engines 0009 and 0204 found that the damage to the FPB was caused by similar failure mechanisms. Metallurgical examination of the lox posts from both of these engines discovered high cycle fatigue (HCF) cracks located at the braze fillet radius regions. Also, all engine 0009 lox posts exhibited cold worked material (austenitic to martensitic transformation) at the fracture location. Analysis and tests confirmed high main stage mechanical vibration and dynamic flow loads. Based on this, it was determined that the through cracks probably developed at the fillet radius during main stage, thus mixing lox and fuel. However, it was noted that although mechanical vibration was the main load source, it was insufficient to cause the damage. Probably, offset fuel sleeve holes caused an additional flow-induced alternating stress. This, when combined with the mechanical vibration stress, exceeded the material endurance strength.

Based on these observations, an eddy current inspection technique was developed to determine the extent of the cold work at the fillet radius area due to fatigue. Metallurgical data determined that an eddy current reading of 5 V or more is sufficient to aggravate the fatigue failure. A method to reduce the fatigue load on the lox posts where eddy current exceeded or met a 5-V value was developed. This technique requires the placement of three support pins, equally spaced between the inner and outer sleeve annulus, thus reducing the cantilever length and dynamic loads. All current flight preburner lox posts are inspected at a predetermined interval and pins are installed if required. Also, an inspection is done to verify that all pins are in place or a fatigue damage algorithm is used to limit hardware use.

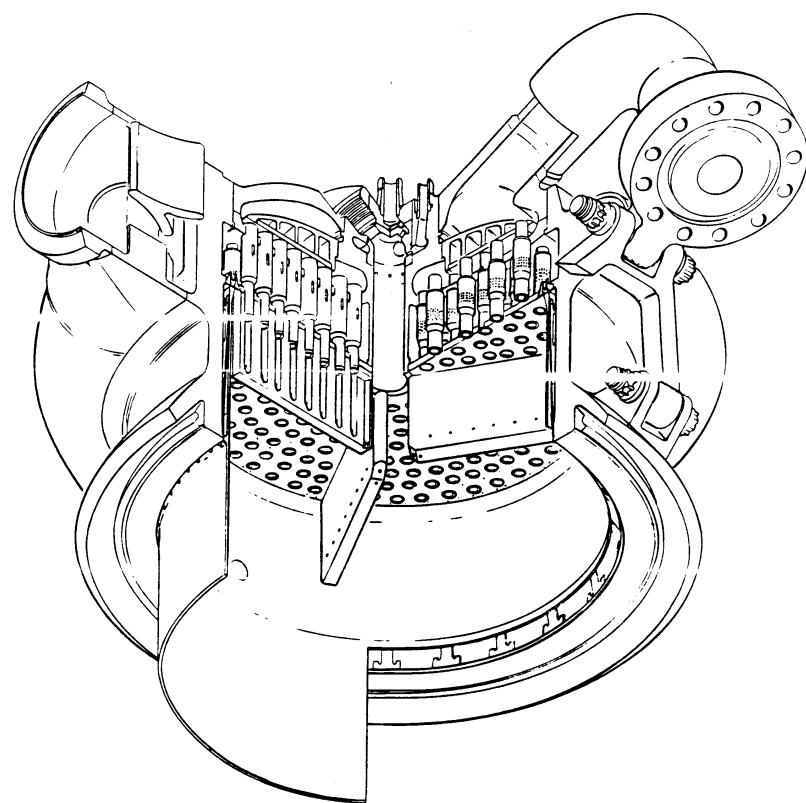


Figure A-6. Fuel preburner.

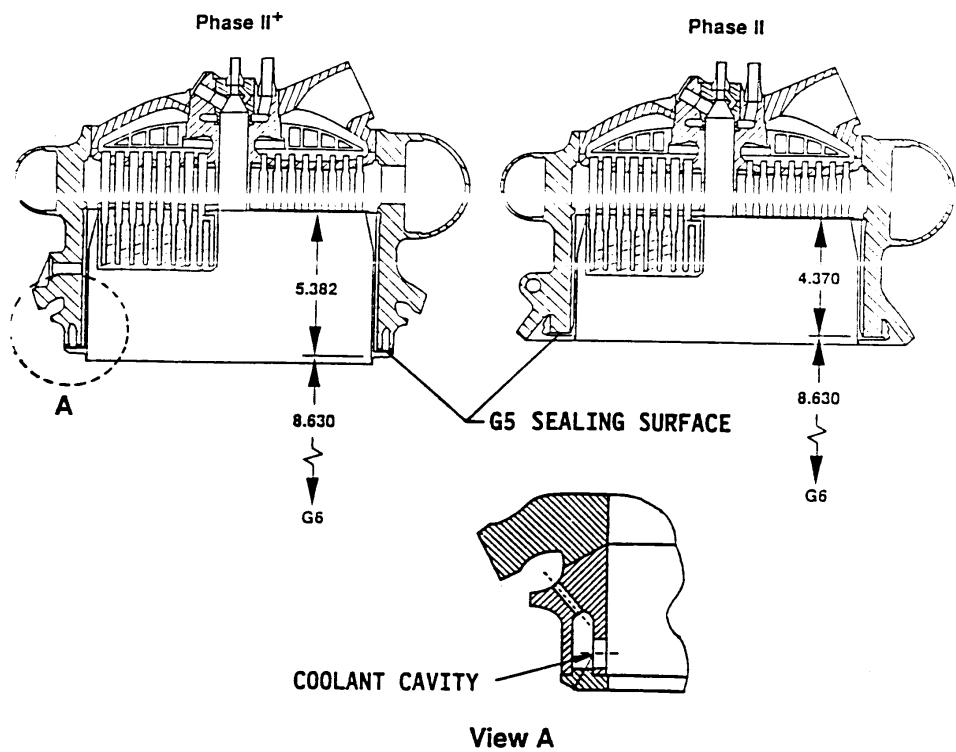


Figure A-7. Fuel preburner changes.

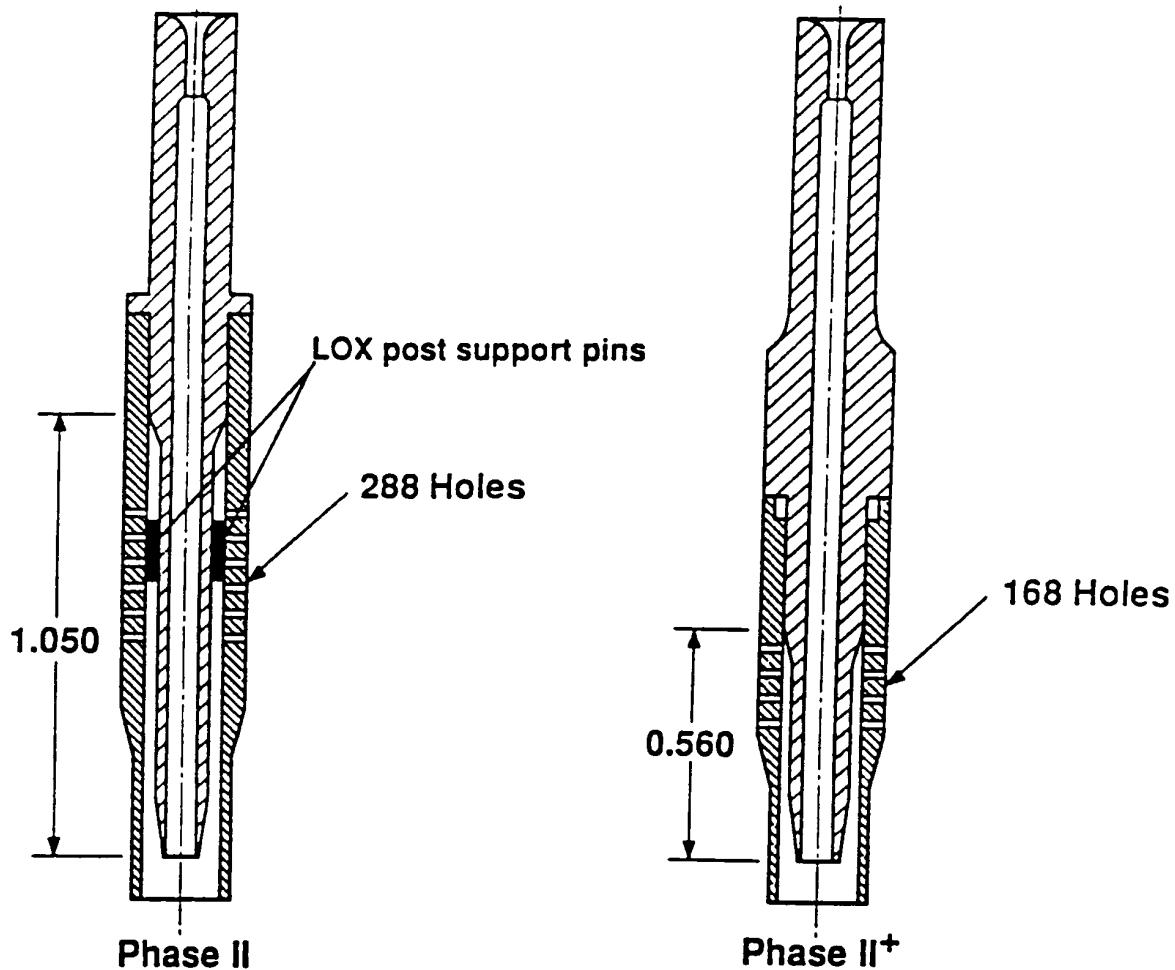


Figure A-8. Fuel preburner element.

In combination, these controls have stopped reoccurrence of this failure. In addition, a design change was made for the more recent two-duct powerhead configuration. This design change increased the thickness of the inner sleeve and reduced the cantilever length of the lox post sleeve.

8. Test 901-222-Engine 0007, Heat Exchanger Fire Incident. The heat exchanger is used on the SSME to convert lox to gaseous oxygen (gox). This system is a series of tubes surrounding the oxygen pump preburner, pump turbine region. Problems have occurred in the heat exchanger. For example, the first test of SSME engine 0007, test 901-222, was terminated at 4.34 s of a planned 50-s test. Engine cut-off was due to the exceedance of the heat exchanger outlet pressure minimum redline. Also, an external fire was observed in the area of the oxidizer preburner. Extensive damage occurred to the combustion system, heat exchanger coil, hot gas manifolds, main injector, and associated ducts (figs. A-9, A-10, and A-11).

A failure investigation team was formed to investigate the cause of the incident and to identify corrective actions to reduce future failures. It was concluded that the incident was caused by a leak in the heat exchanger coil near the inlet and discharge area of the heat exchanger. Oxygen from the coil leak mixed together with the fuel-rich preburner combustion gas and was ignited by the gas temperature. This failure could have been caused by one of two possible modes: (1) undetected damage to the inlet tube during reaming of the inlet for removal of the weld drop-through and/or (2) arcing damage associated with a heat exchanger welding operation at the coil support bracket.



HEAT EXCHANGER COIL WELDS

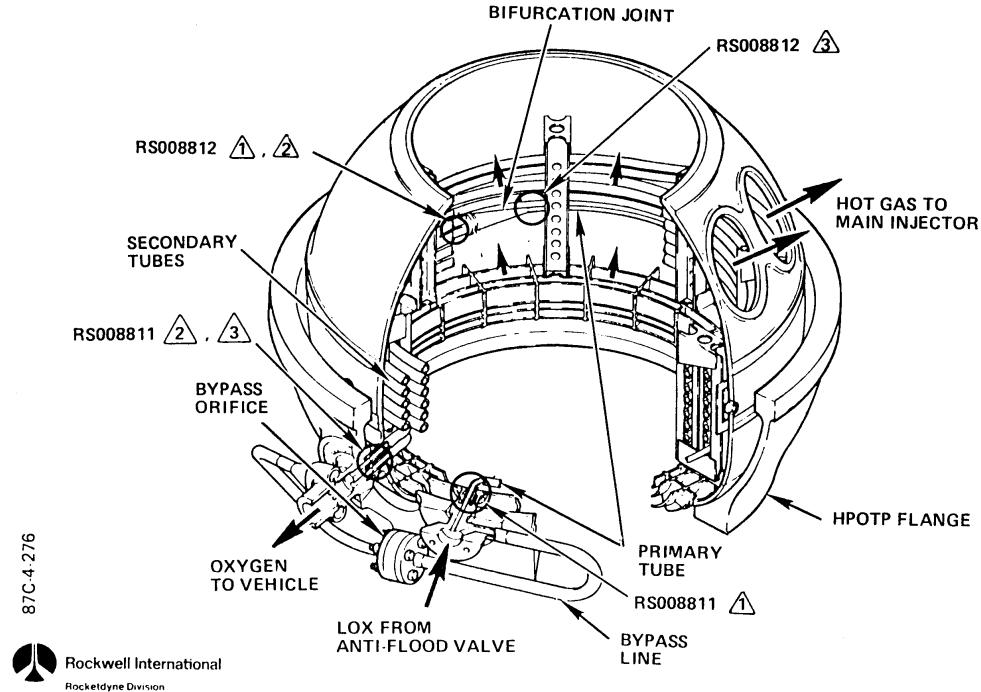


Figure A-9. Heat exchanger.

RS008806 BIFURCATED ASSEMBLY

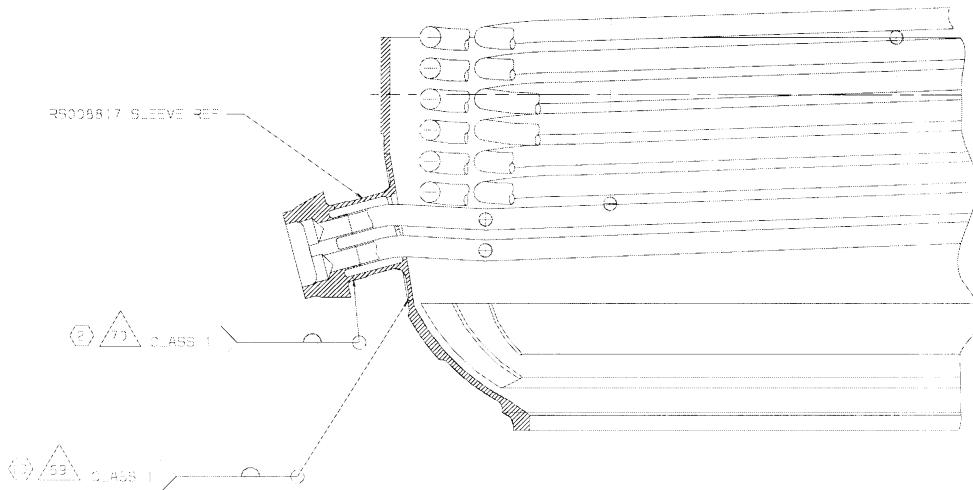


Figure A-10. Heat exchanger details.



SSME HEAT EXCHANGER TUBE ASSEMBLY

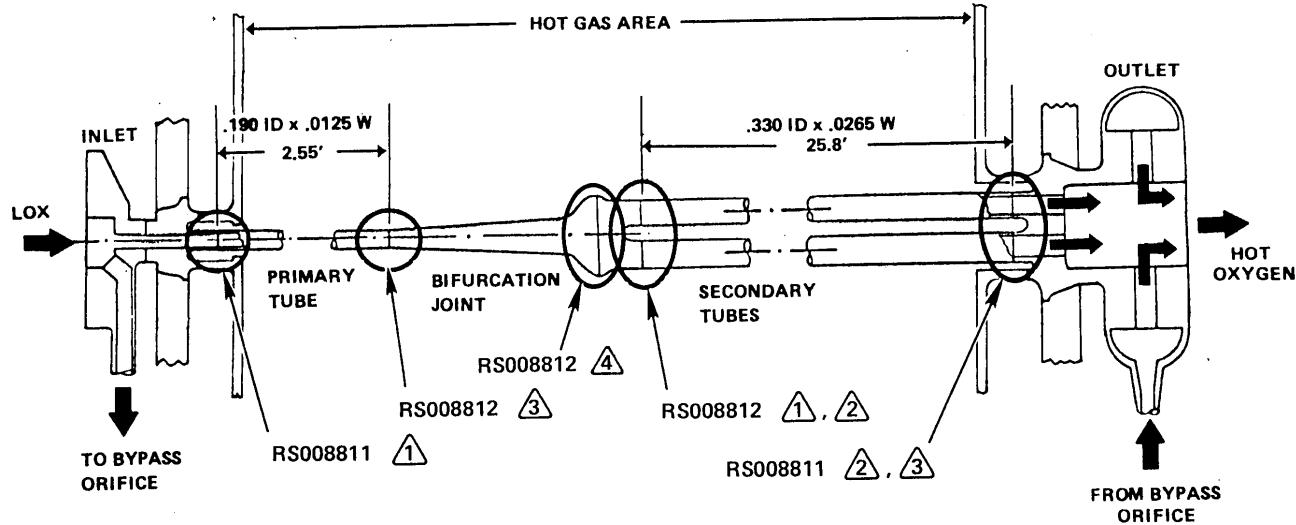


Figure A-11. Heat exchanger bifurcation joint.

Corrective actions have been implemented for all future builds. Some of them are: (1) leak-test before each hot-fire test, (2) proof and leak check after each HPOTP installation, (3) visual inspection of the heat exchanger area for evidence of damage, arcing, or weld discoloration, (4) modify the manufacturing process to provide an electrical ground to the bracket being welded, (5) positive stop and pilot for reaming and boroscope inspection after reaming, (6) increased coil proof test pressure to provide a proof factor of 1.26 instead of 1.2, (7) improved leak detection requirement, and (8) purging coil using a high-velocity nitrogen purge to clean out any contamination or debris in the tubing or outlet area immediately before the installation of the inlet and discharge lines. In addition, all completed units were inspected to ensure that a condition similar to the failed unit did not exist. Together, these actions have improved safe engine operation.

Tables A-5 through A-9 list the problems related to the SSME.

Table A-5. Outline of SSME problems.

1. Turbomachinery

a. HPOTP turbine blades/disks

- (1) First-stage turbine blade cracks
 - Radial shank
 - Radial platform
 - Radial firtree
 - Tip seal rail
 - Center plate rail
- (2) First-stage turbine blade high-cycle fatigue transverse shank cracks.
- (3) Second-stage turbine blade cracks
 - Radial airfoil
 - Radial shank
- (4) First-stage disk interstage seal pilot rib fillet radii cracking
- (5) Turbine disk curvic teeth damage
- (6) First- and second-stage disk firtree cracks

b. HPFTP turbine blades/disks

- (1) First-stage turbine blade failures
 - November 1977
 - December 1977
 - September 1981
- (2) First- and second-stage radial airfoil cracks
- (3) First-stage transverse leading-edge airfoil cracks. Transgranular fracture.

c. HPFTP whirl

- (1) Rotating assembly instability with three-stage pump interstage seal. Subsynchronous whirl vibration.
- (2) Phase II HPFTP subsynchronous vibration. Subsynchronous frequency ranges from 47 to 52 percent, tracks pump speed

d. HPOTP whirl

- (1) Phase I rotor is susceptible to second rotor mode whirl.
- (2) Rotor instability

e. HPFTP bi-stable pumps

- (1) Periodic MCC pressure oscillations due to control system interaction with flow variations from HPOTP preburner pump

f. HPOTP bearings

- (1) Turbine bearing inner race cracking
- (2) Delamination of No. 3 bearing cage
- (3) Bearing No. 4 wearing more than bearing No. 3

h. Other HPOTP cracks

Table A-6. Turbomachinery problems.

a. HPOTP turbine blades/disks

Component	No.	Problem	Problem Cause	Problem Solutions
HPOTP	1.a.1	First-stage turbine blade cracks • Radial shank • Radial platform • Radial airfoil • Radial firtree • Tip seal rail • Center plate rail	Low-cycle fatigue caused by thermal shock Propagation is likely caused by sustained load hydrogen embrittlement High-cycle fatigue/chipping caused by damping contact	DAR 2272: Life limited to 5,000 s and 21 starts; supported by statistical analysis (single flight reliability) and assessment of cracking as benign
	1.a.2	First-stage turbine blade high-cycle fatigue transverse shank cracks	Previous one-piece damper design did not provide enough damping against vibration.	First-stage turbine blade redesign was two-piece damper. New design improved damping, reduced dynamic stresses, and corrected transverse shank cracking. Test history >600 tests and >250,000 s without cracking
	1.a.3	Second-stage turbine blade cracks • Radial airfoil • Radial shank	Low-cycle fatigue, thermomechanical	Life limited by DAR to 5,500 s and 22 starts; supported by statistics (SFR), >550,000 s without transverse cracks
	1.a.4	First-stage disk interstage seal pilot rib fillet radii cracking	Failure mode is low-cycle fatigue assisted by internal hydrogen embrittlement predominantly driven by shutdown thermal shock, large strains caused by small radius	Problem solved by redesigning with larger interstage pilot rib fillet radii, rework has passed certification, lift limited to 14 starts before rework
	1.a.5	Turbine disk curvic teeth damage	Removal of excess gold from the curvic teeth results in gouges (fabrication induced)	Life limited by DAR 2268 until reworked to remove gouges and restore to CEI life
	1.a.6	First- and second-stage disk firtree cracks	Low-cycle fatigue caused by thermal shock assisted by hydrogen embrittlement due to missing gold	Life limited by DAR 2023 and 2024

b. HPFTP turbine blades/disks

Component	No.	Problem	Problem Cause	Problem Solutions
HPFTP Sept. 1981 Dec. 1977 Nov. 1977	1.b.1	First-stage turbine blade failure (Incident No. 22) (Incident No. 5) (Incident No. 4)	Localized high-temperature streaking from the fuel preburner caused disbonding of the first-stage turbine tip seals; excessive rubbing of the blades caused blade failure, rotor speed decay and a resulting HPFTP inlet volute rupture and lox-rich shutdown	<ul style="list-style-type: none"> • Elimination of the tip seal nickel plating (first stage) • Improvement in turbine inlet temperature profile • Tighter control on tip seal clearance at assembly • Improved retention of Kel-F seals (failure increased temperature/speed) • Decreased turbine temperature redline • Eliminated nonstandard plugging at lox posts in preburner injector
Sept. 1978	1.b.2	First- and second-stage radial airfoil cracks	Caused by thermal fatigue	Eliminated by application of blade coating. Decreased start transient temperature spikes
April 1979	1.b.3	First-stage transverse leading edge airfoil cracks; transgranular fracture (8 pumps with cracks)	Crack propagation is HCF fatigue; caused by superposition of high-cycle excitation during test on a high mean stress caused by start thermal spike	Problem reduced by modified engine start with reduction of start thermal spike (7/82); 22X inspection intervals were specified (4/79)
Feb. 1977	1.b.4	First-stage transverse trailing edge cracks	High-cycle fatigue caused by damper/blade bonding	Eliminated by elimination of plating on damper

Table A-6. Turbomachinery problems (continued).

b. HPFTP turbine blades/disks (continued)

Component	No.	Problem	Problem Cause	Problem Solutions
May 1978 to present	1.b.5	First- and second-stage platform cracks; intergranular and interdendritic cracks	Caused by thermal fatigue	Decreased start transient temperature spikes; required boroscope inspections on first-stage blades
Nov. 1978 to present	1.b.6	First- and second-stage radial shank intergranular cracks	Caused by thermal fatigue	Decreased start transient temperature spikes; coating on first-stage blade shanks
Dec. 1978 to Feb. 1991	1.b.7	First-stage transverse upstream shank cracks; pressure and suction side corners (15 pumps)	Low-cycle fatigue thermally induced	Coating on first-stage blade shanks seems to have eliminated cracking
June 1981	1.b.8	First-stage transverse downstream shank cracks (11 turbopumps)	Thermal cycles initiate cracks because of local surface melting surface carbide, and corner stress riser; growth occurs due to high-cycle fatigue	Phase II modes: recontoured and coating; no cracking with recontoured blades
Nov. 1978	1.b.9	First-stage transverse firtree cracks; cracks at bottom firtree lobe (12 turbopumps)	Transgranular fracture due to overload and assisted by hydrogen embrittlement	Shotpeen and radoised firtree corners reduce acceptable microporosity size limits, mass finish, and optimized fabrication sequencing; reverse broach first-stage disk
May 1977	1.b.10	Second-stage platform cracks; crystallographic fracture; pieces of platform lost; one crack turned and became transverse	Fracture caused by stage one HCF; initiation on underside of platform at fretting caused by damper	Boroscope inspect through turnaround duct; replace old style damper with new lighter precision damper
May 1979	1.b.11	Second-stage transverse downstream shank cracks; deepest crack 0.20 in	Initiation at geometric stress riser; propagation due to stage 1 HCF	Phase II modes; recontoured, shot-peened, and coated
July 1978	1.b.12	Second-stage downstream firtree cracks (39 turbopumps)	Crystallographic fracture assisted by hydrogen environment; initiation occurs at carbides at surface	Stress relief and bead blast; shot-peen and radiused firtree corners, improved surface finish on firtree downstream face mass finish, optimized fabrication sequencing, reduce acceptable microporosity size requirements of recontoured shank
Feb. 1978	1.b.13	First-stage damper pocket cusp cracks found in 15 turbopumps	Thermal transient	None – margin tested, risk minimal
July 1978	1.b.14	First-stage turbine blade lobe cracking resulting from blade biasing	A difference in the direction of broaching between the first- and second-stage disk	Revise the direction of broaching of the first-stage disk to be the same as the second-stage disk
HPFTP	1.b.15	Rotating unbalance of turbine disk (phase I HPFTP)	The first-stage blades were coated with zirconium; zirconium has poor adhesion characteristics; loss of coating caused rotating unbalance	Changed to Nicraly coating only on blades
May 1990	1.b.16	First- and second-stage turbine disks curvic teeth; after gold plating, excess gold was frequently found on the load flats of the curvic teeth; upon excess gold removal, damage occurred to the fillet region of the curvic teeth on several disks	Excess gold occurred due to insufficient masking; the damage occurred due to improper techniques for removal	A rework procedure was established to repair the damage to the curvic teeth root radii; the reworked configurations were incorporated as new configurations for the first- and second-stage disks
July 1991	1.b.17	E0215 failure; failure was due to a second-stage turbine blade fracture (incident No. 34)	Blade fracture was due to a large internal flaw assisted by hydrogen embrittlement	Limited blade life; established a computer tomography inspection criteria to attempt to flush pores

Table A-6. Turbomachinery problems (continued).

b. HPFTP turbine blades/disks (continued)

Component	No.	Problem	Problem Cause	Problem Solutions
Oct. 1991	1.b.18	First-stage blade stop tab fracture; rub mark 360° on fishmouth seal below lip; loss of blade stop creates a 2.33 grm unbalance, which equates to a 0.2 grm increase in synchronous vibration	Failure mode identified as hydrogen-assisted high-cycle fatigue; localized assembly stresses in blade stop radius in combination with operating environment	Fragment is of insufficient mass to cause downstream damage; revised assembly procedure to preclude blade stop damage during installation
April 1987	1.b.19	First-stage downstream pressure side corner lobe cracks	Mechanically induced by grinding to remove excess gold	Shotpeen and radiused firtree corners; elimination of gold on downstream face
Feb. 1989	1.b.20	First-stage radial downstream firtree cracks	Intergranular cracking	Microporosity size limits; mass finish removes rough edges

c. HPFTP whirl

Component	No.	Problem	Problem Cause	Problem Solutions
HPFTP	1.c.1	Rotating assembly instability with three-stage pump interstage seal; subsynchronous whirl vibration		Incorporation of straight, smooth sleeves and seals improved the rotordynamics stability; damping seals replaced smooth seals to increase the rotodynamic stability margin
May 1988	1.c.2	Phase II HPFTP subsynchronous vibration; subsynchronous frequency ranges from 47 to 52 percent, tracks pump speed on ramps, displays some wandering during a test, has split into dual peaks, and has modulated with pump speed; also, occurs less often with straight/smooth seals, and more often with lower side load	The initiation of subsynchronous vibration is dependent on the turbine sideload; as shown by the presence of subsynchronous vibration on the FMOF TAD configured HPFTP's; there is no definite correlation between subsynchronous and rubbing	Recommended that the incorporation of damping seals be included in the phase II+ design for the two-duct HGM powerhead; not considered a problem for phase II

d. HPOTP whirl

Component	No.	Problem	Problem Cause	Problem Solutions
HPOTP	1.d.1	Phase I rotor experienced first rotor mode whirl at >50-percent synchronous speed	The straight turbine interstage seal design enhanced the first rotor mode whirl; the scalloped interstage seal ring enhanced the gas preswirl in the inlet of the interstage seal	Redesign of the turbine interstage seal to a tapered shape plus a swirl brake upstream enhanced the rotor's stability by removing the driving forces The scallops of the interstage and seal ring were eliminated, stopping the driver of the fluid preswirl and adding to the rotor stability
	1.d.2	Phase I rotor experienced second rotor mode whirl at 90-percent synchronous speed	The labyrinth seal design increased sensitivity to second rotor mode whirl	New preburner pump damping seal design reduces response amplitude and provides additional second-mode stability margin; subsynchronous whirl no longer exists

Table A-6. Turbomachinery problems (continued).

e. HPOTP bi-stable pumps

Component	No.	Problem	Problem Cause	Problem Solutions
HPOTP	1.e.1	Periodic MCC pressure oscillations (thrust variation exceeds CEI allowable) due to control system (OPOV position) interaction with flow variations from HPOTP preburner pump	Flow stall at lower power level in preburner volute discharge vanes	<ul style="list-style-type: none"> • Limit orbiter to 67 percent of minimum power level • Redesign volute vane, vane angle, and contour change • Screen out one of approximately every 20 volutes exhibiting bi-stability in green run tests

f. HPOTP bearings

Component	No.	Problem	Problem Cause	Problem Solutions
HPOTP	1.f.1	Turbine bearing inner race cracking	Stress corrosion cracking (SCC) initiated from corrosion pit; key ingredients for SCC are hoop stress and trapped moisture	<ul style="list-style-type: none"> • Roughen spacer to between 32 and 64 surface finish to enhance moisture removal • Delete heat gun use to minimize moisture collection • Spacer not bottomed prior to dry cycle to enhance spacer/bearing gap drying • Multiple chills without dry cycle not permitted • Preheat vacuum dry oven to 185 to 195 °F • Initiate vacuum cycle when critical hardware temperature is within 5° of oven temperature • Vacuum cycle extended from 2 to 3 h to enhance moisture removal
	1.f.2	Delamination of No. 3 bearing cage	Four-lobe nut design tended to cause pumping of lox against the No. 3 bearing cage	Turbine bearing nut changed from four full-length lobes to two shortened lobes; problem corrected
	1.f.3	Bearing No. 4 wearing more than bearing No. 3	Uneven load on the bearings and/or unequal cooling and/or unloading	Turbine bearing cartridge enlarged; problem of uneven wear improved but not solved
	1.f.4	Bearing No. 2 wearing more than bearing No. 1	Uneven load on the bearings	Pump bearing isolator enlarged; even loading of bearings was not accomplished, therefore, change did not help uneven bearing wear; unincorporated solutions; hydrostatic bearing, silicon nitride balls
	1.f.5	Uneven wear of ball bearings between bearings No. 1 and 2, and between bearings No. 3 and 4	Uneven load on the bearings	Preburner pump bearing isolator and turbine bearing cartridge have had their deadband increased; deadband is the space between the outer race and the isolator and cartridge; the greater the space, the more movement before the load is transferred to the isolator and cartridge; in general, the turbine end bearings No. 3 and No. 4 improved; little change in the uneven wear between the No. 1 and No. 2 bearings

Table A-6. Turbomachinery problems (continued).

f. HPOTP bearings (continued)

Component	No.	Problem	Problem Cause	Problem Solutions
HPOTP	1.f.6	PBP ball bearings (No. 1 and No. 2) experienced spalling	In-housing rotor balancing with inadequate lubrication/cooling	Thermal growth of ball bearings caused balls to expand to the point of detrimental rubbing with the race; the inner race diametral bearing clearance was increased in conjunction with elongated cage pockets; eliminated in-housing rotor balancing; PBP bearing springs between bearings No. 1 and No. 2 were softened to reduce the axial load on the bearings (bearing preload); the bearings no longer spall, but bearing wear has increased
Sept. 1977	1.f.7	Bearing number 3 failure (incident No. 3)	Excessive coolant flow forces in the preburner pump end bearings unloaded the No. 1 bearing, allowing high radial motion; the excessive pump end radial motion increased the turbine end overhung rotor deflections; unequal load sharing of the turbine end bearings (No. 3 with most of the load) combined with the high synchronous rotor deflections degraded the turbine bearings, allowed rubbing and ignition	<ul style="list-style-type: none"> • Decreased the bearing coolant forces-pump and turbine • Added turbine end bearing coolant jet ring • Increased bearing preload • Redesigned isolator and cartridge • Improved balance
June 1989	1.f.8	Preburner pump bearing failure (incident No. 33)	HPOTP PBP excessive ball wear and failure of high time bearings; bearing heat generation escalates cage degradation and breakup; cage particles block coolant exit, causing pressure buildup and asymmetric coolant flow; high-velocity particles impact and ignite on the inducer, large gas volume in inducer and impeller area allows rapid speed increase; turbine overspeed causes first-stage disk failure	The block I modification of this turbopump was unique; the design modification was intended to reduce bearing wear and is not a flight design

g. HPFTP bearings

Component	No.	Problem	Problem Cause	Problem Solutions
HPFTP July 1989	1.g.1	Bearing cage cracks	Six cage fractures to date; cracks are discovered during normal recycle; test history indicates cracks related to total time	Experience with fractured cages is benign; demonstrated factor of 2 in excess of 2,500 s; life limited bearing to 2,000 s; cage was thickened on 10 K pump
May 1991	1.g.2	Missing chrome on turbine bearing carrier at the No. 4 bearing location	Heavy spinning of No. 4 bearing outer race caused by ball/race friction, cage/race friction, and bearing loads	10 K pumps with the FEP bearings provide a reduced driving force for outer race spinning; no dynamic signature related to wear/loss of chrome; no anomalous bearing ball and raceway conditions resulted from loss of chrome or preload spring wear

Table A-6. Turbomachinery problems (continued).

h. Other HPOTP cracks

Component	No.	Problem	Problem Cause	Problem Solutions
HPOTP	1.h.1	Cracks in the second-stage nozzle, which extend beyond the acceptable region of the inner shroud relief grooves	Axial cracks are attributed to low-cycle thermal fatigue damage produced by thermal strains during engine startup and shutdown transients; aggravated by casting microporosity/shrinkage	The cracks are controlled by visual inspection at each disassembly; all cracks outside the relief groove area are evaluated by MR; life limited by DAR; hipping (hot isostatic pressing) of casting to reduce microshrinkage
	1.h.2	Cracking cup washers	Previous cup washer material specification was on the sheet metal material from which the washers were formed; cold working of material resulted in high hardness and low ductility, predisposing washers to cracking	ECP written to define the material properties on the shaped form
	1.h.3	First-stage turbine disk coolant jet ring S tubes cracking	High-cycle fatigue	S tube material changed from 347 CRES to A286 stainless steel; A286 has increased fatigue resistance, solving the problem
	1.h.4	Hydrogen leak allowed by 0.25-in crack in housing coolant plug weld No. 11	Class II plug weld is susceptible to shrinkage cracks	All HPOTP's are leak tested prior to flight
	1.h.5	Balance piston cavity pressure port welds	Lack of fusion or incomplete penetration in class II fillet welds, Nos. 22 and 24, of the balance piston; existing NDI could not detect root side or imbedded critical size HCF flaws	Use as is; these welds are tolerant to large defects; soap bubble and mass spectrometer inspection done after every acceptance test
	1.h.6	Housing seal groove fillet cracking	Low-cycle fatigue	Problem determined to be acceptable via fracture mechanics; flaw will not propagate past a shallow residual stress depth
	1.h.7	Diffuser vane cracks	Main impeller blade wake dynamic loading plus high mean stress caused high-cycle fatigue cracks at leading edge of vanes	Improved loading path by cutting back the leading edge of the diffuser vanes 0.10 in; reconfigured grooves adjacent to vane passageway; shotpeen leading edge area of vane after proof test
	1.h.8	Main housing drain "T" fitting leaks	Stringers in fitting material (CRES)	Changed fitting material from CRES to Inconel 718™; leak check CRES fittings postflight
	1.h.9	Main housing microfissures	Casting process caused surface microfissures; core reaction	Life limited to 11,024 s by DAR; improved process to eliminate core reaction
	1.h.10	Jet ring cup washer high-cycle fatigue cracks	Flow environment caused high (40 kHz) frequency high-cycle fatigue cracks	Life limited by DAR; redesign TBD

Table A-6. Turbomachinery problems (continued).

i. Other HPFTP cracks

Component	No.	Problem	Problem Cause	Problem Solutions
HPFTP May 1979 to present	1.i.1	First-/second-stage turbine nozzle cracks; cracks are on leading edges of vanes and inner and outer shrouds; cracks initiate at low and high time	Low-cycle fatigue caused by thermal transient start	Modified start sequence to reduce thermal transients; cracks are shallow and monitored by borescope and dye penetrant inspections
March 1985	1.i.2	Bellows shield cracking	Failures not function of starts, time, or power level	Designed a short tip bellows shield; vastly reduced occurrence of cracking
July 1985	1.i.3	First-/second-stage impeller indentations; 30-mil deep cracks predominantly in axial direction; cracks observed in all three indentations in or around galled areas; 15 indentation depth, two sets of indentations, three indentations per set	Indentations created by pounding type contact and rubbing between impeller inner diameter and the tie bolt lobes; relative contact that caused localized heating accompanied by high-strain levels to form microstructurally changed material susceptible to crack initiation and propagation	Redesigned impellers
July 1989	1.i.4	First-stage tip seal/housing lug fractures	High mean stresses in both housing lugs and tip seal lip; stress is induced by housing distortion; there is sufficiently high dynamic stresses for high-cycle fatigue failure	Redesign first-stage tip seal lip at three locations to remove effects of housing distortion; existing hardware was to be reworked; the redesign eliminated static load on housing lugs, reduced mean stress in tip seal lip, distributed pressure load to ends of tip seal, increased alternating stress capability by a factor of 4, and smoothed scalloped shape of tip seal
August 1989	1.i.5	Second-stage diffuser pilot lip crack; circumferential cracks up to 7-in long; interstage seal wear typical on units with cracked diffusers	All eight cases due to out-of-print diffuser or housing; diametral interference of 0.0187 to 0.0297 in	Document housing and diffuser diameters prior to assembly; increased groove radius; a pilot lip fragment would be contained during hot fire testing; prior flight experience with cracked diffuser was benign
September 1989	1.i.6	Center body casting to pump end housing forging weld 59 indications; x-ray indications show up as shadowlines and radial indications; concern for joint integrity because of x-ray indications and, shadowline masking of defects	Defects result from welding difficulties	Analytical results and inspection requirements assure flight safety; welding speed slowed from 30 to 20 in per minute for new housings
	1.i.7	Turbine support welds 16 and 17 flaws	Penetrant indications are frequently observed at these welds after normal machining or thermal processes, this problem was designated a producibility item	Fracture mechanics analysis was conducted to develop a criteria for acceptable flaw sizes; low-stress machining was employed; weld surfaces are polished to remove flaws prior to etch and pent; the preparation for weld 17 was modified to improve accessibility when making the weld
September 1990	1.i.8	Titanium inlet sheet metal weld offsets; initial HCF analysis showed many offset conditions to be unacceptable	Weld offsets are caused by difficulty in the welding process	More detailed analysis was started to eliminate conservatisms; for this purpose, more hot-fire strain gauge data were obtained, laboratory testing to determine bending/membrane splits was conducted, and a fine mesh finite element model was used for calculating mean stresses

Table A-6. Turbomachinery problems (continued).

i. Other HPFTP cracks (continued)

Component	No.	Problem	Problem Cause	Problem Solutions
HPFTP March 1985	1.i.9	First-stage impeller front shroud crack; high-cycle fatigue crack initiated at impeller shroud O.D.; initiation occurs at site having adverse surface conditions (corner break and surface finish)	Impeller diametral mode resonance with first-stage diffuser vanes; most probable impeller response mode identified by modal tests and structural dynamics analysis; mode characterized at 12 mod 3 mode; this occurs in the 32,000 to 32,600 r/min range; and it requires interaction with diffuser vanes	Initially, the operation at the most probable response mode was limited to 160 s in the 32,000 to 32,600 r/min range; improved control of diffuser vane dimensions; also, improved surface finish and shroud corner break control; increased shroud corner radius allowed for increased DAR limits
February 1991	1.i.10	Housing HCF life; discharge volute welds 223 and 228 HCF predicted life based upon limited hot-fire strain gauge data and worst-case drawing geometry is less than the CEI	Several housings held with weld discrepancies during fabrication cycle; similar weld mismatches have been observed in old and new housings	Flight based on single flight reliability; analysis underway to determine mean and alternating stress to calculate life; 3-D FEM submodel parametric model to account for offset and radius of volute
August 1980 to present	1.i.11	Thrust ball cracks	Cracks found on the ball wear tracks indicate high metal temperatures and thermal gradients; rubbing between ball and insert causes localized heating during contact	By analysis, high-surface tensile stresses predicted; cracking relieves residual and/or thermal stresses; moly lube added to thrust ball to reduce friction; thrust ball inspected after every test; cracked balls are replaced
December 1991	1.i.12	Second-stage KEL-F support pilot lip crack; visual inspection detected crack in radius at pilot/support interface	Two occurrences of support pilot fracture in program; first fracture at large pilot and second at small pilot; high-cycle fatigue propagation; first fracture due to out-of-print condition; second fracture still undetermined	Changed requirements for repairing KEL-F supports; defects deeper than 0.20 in and cracks are rejected; monitored by DAR inspection limit
January 1992	1.i.13	HPFTP housing inner ring cracking type A (through) cracks through the bolt holes common to all units	Low-cycle fatigue caused by punch loads from discharge struts and aggravated by hot gas leaking into the coolant circuit	DAR 1199 requires a complete bolt hole inspection at each turbine disassembly and at 20 starts and even 10 starts thereafter; the interval drops to 5 starts if crack conditions are severe enough
October 1981 to present	1.i.14	HPFTP turbine support sheet metal fragments; dislodged fragment of inlet sheet metal detected during posttest inspection; second occurrence on same unit	Crack initiation and propagation by high-cycle fatigue	All HPFTP's are inspected after every hot-fire test for any out-of-specification cracks; new inspection requirements; DAR limits sheet metal to 4,250 s
March 1992	1.i.15	Weld 56 turnaround duct weld crack	Crack initiation and propagation by high-cycle fatigue	Added infield inspection and DAR that life limited the weld to 13,600 s
July 1992	1.i.16	HPFTP main housing bellow crack; investigation of source of Haynes 188 in engine revealed cracks and missing fragments from main housing bellow	Only one occurrence observed on current configuration; bellows manufactured with thin wall	New units correctly machined
June 1990	1.i.17	LPFTP second-stage rotor crack in outer shroud; cracked rotor is fleet leader for time and fifth fleet leader for starts; unit No. 82106	High-cycle fatigue failure	DAR 2431 limits life to 15,000 s

Table A-6. Turbomachinery problems (continued).

j. HPOTP Other

Component	No.	Problem	Problem Cause	Problem Solutions
HPOTP July 1978	1.j.1	Capacitance probe failure (incident No. 9)	Contact of the capacitance probe plates with the pump shaft speed nut; the resulting frictional heating and abrasion of metallic particles in a lox environment caused ignition	Removed capacitance probe
March 1977	1.j.2	Primary lox seal failure (incident No. 1)	Ignition in seal package area; most likely the result of primary seal bellows failure	<ul style="list-style-type: none"> • Redesign seal package • Increased barrier pressure lower redline from 75 lb/in² gauge to 170 lb/in² absolute • Instituted HPOTP secondary seal cavity pressure redline of 100 lb/in² absolute maximum
	1.j.3	Turbine to preburner interface leakage failure effects	HPOTP eccentric ring damage during pump installation	New single-piece design implemented on all pumps to reduce chance of crushing the ring

k. HPFTP Other

Component	No.	Problem	Problem Cause	Problem Solutions
HPFTP	1.k.1	Axial and radial rubbing of the first-stage turbine blade tip seal	Problem not accounted for in original design	Reconfigured to eliminate axial and radial rubbing
	1.k.2	Axial rubbing between the second-stage disk and turbine interstage seal	Insufficient clearances between turbine interstage seal/second-stage fishmouth seal and second-stage disk/forward blade platform	Increased the clearances between the turbine interstage seal/second-stage fishmouth seal and second-stage disk/forward blade platform
	1.k.3	Thin sheet metal bulging of inner and outer shrouds	Insufficient original design	Increased the sheet melt thickness to eliminate bulging
	1.k.4	Pressure losses in the turnaround duct portion of the FMOF HPFTP turbine	Phase I HPT did not eliminate pressure losses; phase II had high turbine temperatures and transverse delta pressure	Reverted back to FMOF inner and outer shroud of turnaround duct, but with thicker sheet metal in diffuser section, to reduce high turbine temperatures and transverse delta pressure
	1.k.5	High turbine temperatures and reduced pump efficiency	Metal impeller labyrinth seal	Changed from metal impeller layby seal to Kel-F seal; the Kel-F seals contributed to reducing the turbine temperatures and improving pump efficiency
February 1984	1.k.6	Coolant liner buckling failure; engine 0108 (incident No. 28)	Coolant liner cavity overpressure due to excessive hydrogen flow past the lift-off seal into the liner caused coolant liner buckling and subsequent collapse of the turnaround duct	Redesign to eliminate excessive seal leakage; redesign to eliminate interstage seal axial rubbing; coolant liner pressure redline incorporated; acceptance criteria developed for coolant liner temperature and pressure
October 1981	1.k.7	Turnaround duct failure (incident No. 23)	High-cycle fatigue cracking of weld 56 due to lack of weld penetration	Weld joint redesigned to provide better weld penetration
August 1989	1.k.8	KEL-F seal internal/external Kel-F melting, T-lock damage, support fretting, antirotation pin damage, nut lock damage, and loss of preload on tapir nut	Relative motion between details; internal melting (because of its possibility of generating large pieces) is most critical; internal melting is believed to be due to vibrations below 12 kHz	Seal redesign; improved bottoming procedure has eliminated support fretting and antirotation pin damage; increase torque on retaining nuts

Table A-6. Turbomachinery problems (continued).

k. HPFTP Other (continued)

Component	No.	Problem	Problem Cause	Problem Solutions
December 1991	1.k.9	12 kHz vibration; most return to flight units exhibited this phenomenon	It is believed that the source of 12 kHz may be from the main housing volute liner vent hole diameters (01/93)	Flying based on margin demonstrated by unit 2027; continued margin testing on units 2127 and 2227; develop a method of quantifying damage and flight acceptance criteria
HPFTP April 1982	1.k.10	Kaiser hat nut failure (incident No. 25)	The Kaiser hat nut redesign provided a hot gas leak path to the Kaiser hat, resulting thermal stresses led to fatigue failure of the hat; the hat failure allowed hot gas flow into the turbine bearing coolant, which failed the bearing; this was the first test of a new Kaiser hat nut redesign	<ul style="list-style-type: none"> • Redesign Kaiser hat nut to eliminate leak path • Decrease ASI jet impingement temperature on nut
HPFTP November 1989	1.k.11	LPFTP housing galvanic corrosion	Galvanic corrosion under copper due to fluids entrapped during plating; attributed to moisture cryopumped during ground test; moisture provides electrolyte for galvanic action between dissimilar metals; significant corrosion is averted if flight occurs within 1 year of the initial ground test	Inspection of LPFTP volute based on hot fire time and time between potential moisture introduction and first flight; vacuum dry instituted after ground test to remove moisture
March 1990	1.k.12	LPFTP 330 Hz phenomena; LPFTP 4006 exhibited a strong 330-Hz phenomena that reached a maximum amplitude of approximately 6 g; several units have had amplitudes in excess of 5 g	Believed to be caused by incipient alternate blade cavitation	Considered as a benign condition because of ground test history beginning with first fuel vent and continuing through two units greater than CEI life, flight history beginning with STS-1, development tests with special instrumentation, and pump disassembly results

1. Lox Alternate Turbopump

Component	No.	Problem	Problem Cause	Problem Solutions
Lox Pump	1.l.1	Pump inlet housing vane leading edge cracking	Subsurface porosity breaking through during operation	Improved casting technique
	1.l.2	Bellows shield buckling	Delta pressure across shield, insufficient venting	Vent holes added
	1.l.3	Turbine turnaround duct cracking on vanes in fillet	High-hoop strain caused by thermal gradient at shutdown	<ul style="list-style-type: none"> • Remove vanes • Slot inner flow wall
	1.l.4	Turbine inlet housing dome cracks	Higher than expected temperature on dome (2,200 °F), high thermal strain	Redesign, dome is now separate piece from inlet housing (hubcap design); change dome material from Inconel 718™ to Haynes 230™
	1.l.5	Turbine inlet housing cracks in core breakout region of struts	High stress concentration caused by tight fillet radius and rough casting surface	Casting redesign, added material so hollow core does not break through, holes drilled instead
	1.l.6	Inducer fatigue crack near fillet between blade and hub; failure on unit 7-1G	Blade rub on O.D., due to tight clearance and high vibration	<ul style="list-style-type: none"> • Eliminated vibration problem • Open OD. clearance from 10 to 29 mils

Table A-6. Turbomachinery problems (continued).

i. Lox Alternate Turbopump

Component	No.	Problem	Problem Cause	Problem Solutions
Lox Pump (cont.)	1.I.7	Bellows failure through crack initiated at bolt attachment	Stress concentration from bolt locking key was oriented in worse location; slot was undersized (not to print) high stress induced during key installation due to interference	<ul style="list-style-type: none"> • Slots moved 90° • Slot size is now controlled • Reduced fit between bellows and turbine outer vane support
	1.I.8	SSME turning vane fatigue cracks when ATD HPOTP utilized	ATD turbine discharge directs more hot gas on vane than current SSME HPOTP; flow-induced vibrations cause cracking	In work (two options) <ul style="list-style-type: none"> • Redesign ATD turnaround duct geometry to direct flow similar to current SSME pump • Strengthen turning vane, increase vane thickness, or change material
	1.I.9	Pump end ball bearing life below prediction, heavy wear on inner race, cage, and balls	Loss of lubrication	<ul style="list-style-type: none"> • Changed ball material to silicon nitride • The bearing health was improved by inner race cooling and outer race guided cage
	1.I.10	Step increases in synchronous vibration levels	Instability of the inducer and shroud clearance aggravated by cavitation loads	Increased inducer tip clearance (29 mil); improved damping seal

m. Fuel Alternate Turbopump

Component	No.	Problem	Problem Cause	Problem Solutions
Fuel Pump	1.m.1	Lift-off seal cracking during initial pressure test	High bending stress in first convolute	<ul style="list-style-type: none"> • Change taper in convolution region • Material change, went from Berylco to MP35N
	1.m.2	Pump discharge housing cracks on trailing edge of splitter vanes	3-percent strain on trailing edge, displacement-driven problem; original P&W finite element model had loads applied in wrong direction	<ul style="list-style-type: none"> • Increase housing wall thickness • Add stiffener ribs • Cut back trailing edge • Add dual splitter
	1.m.3	Diffuser cracking around orifice hole	High stress concentration due to sharp edge	<ul style="list-style-type: none"> • Increase radius • Improve casting technique
	1.m.4	Turbine inlet cracks on inner shroud trailing edge	High strain during startup transient due to thermals	<ul style="list-style-type: none"> • Slotted shroud, eliminated hoop continuity • Changed interference fits
	1.m.5	Turbine housing/discharge housing bolt cracking	High temperature on bolt head due to recirculation of hot gas	<ul style="list-style-type: none"> • Add additional cooling to the cavity • Reduce recirculation with flow blocker

Table A-7. Rotordynamic problems.

Element/Subsystem	Problem	Problem Cause	Problem Solutions
Rocketdyne HPFTP	Subsynchronous vibration	Rotordynamic instability-rotor supported on smooth seals will exhibit subsynchronous self-excited whirl when the speed approaches twice the first natural frequency	Damping seals reduce the destabilizing cross-coupled forces and allow stable operation beyond twice the first natural frequency
Rocketdyne HPOTP	Subsynchronous vibration	Rotordynamic instability-rotor whirled at 90 percent of shaft speed due to destabilizing forces generated by slipping spline	Splines were piloted to prevent slippage and damping seals were incorporated to supply stiffness and damping
Rocketdyne HPFTP	A 175-Hz subsynchronous whirl limited pump speed to 22,000 r/min	Rotordynamic instability	Bearing carriers were stiffened after coulomb friction devices proved unsuccessful; further stiffening was achieved by replacing two major labyrinth seals with two smooth seals between the three pump impellers; this raised the whirl frequency from 175 to 300 Hz without being eliminated
ATD HPOTP	High-synchronous vibration	Hydrodynamics excitation resonance	Redesigned inducer
ATD HPFTP	Alpha vibration	Hydro possibly	Not solved
Simplex pump	Critical speed in the operating range (analytical)		Lengthened bearing span, lightened inducer, reduced overhang

Table A-8. Combustion devices.

2. Combustion Devices

a. Main Combustion Chamber

Component	No.	Problem	Problem Cause	Problem Solutions
MCC	2.a.1	Delamination of EDNi/Narloy-Z bondline; typically located on MCC aft end	Attributed to two different operations in the plating process: (1) High copper levels in the cathodic activation process (2) Mercury contamination during EDNi process	<ul style="list-style-type: none"> Improved process to remove copper and other metallics for the bath; established requirements that allow only 25 percent of the previous allowed metallic contaminants Mercury was entering the tank through the tooling used to rotate the part; all the mercury has been removed from the tooling and replaced by copper brushes
	2.a.2	EDNi material properties do not meet specifications; yield strength is below the specification minimum	Inadequate control and definition of processing variables; lack of proper process monitoring techniques, changes in raw material, and general contamination	A comprehensive set of CPI and Taguchi experiments were conducted to define the major process control factors; limits have been established for these factors and are being tightly controlled; an automated process control system has been installed; a class I ECP was processed to better define the process, including the accept/reject; final acceptance of EDNi properties is determined by a specimen from excess stock on aft end

Table A-8. Combustion devices (continued).

Component	No.	Problem	Problem Cause	Problem Solutions
MCC (cont.)	2.a.3	Inlet welds 4 and 5 do not meet CEI requirements when minimum wall thickness and maximum allowable drawing mismatch are considered	Conflicting tolerances; manufacturing can make part within specifications, but certain combinations of thickness and offset prevent NCF CEI life to be shown by analysis	An additional requirement was added to determine if units meet CEI life requirements or to determine the allowable life; this was done using analysis, pull test strain data, hot-fire strain data, or surface contour data; this problem points to the need to do sensitivity studies in the design phase
	2.a.4	Liner life analysis methodology does not predict the observed midchannel failure location	Current analysis methods are not capable of modeling the failure of a hot gas wall of a MCC liner	
	2.a.5	Pin hole leak in weld joint 12 detected during postflight leak check of STS-26	X ray was able to identify lack of fusion in weld joint 12	MCC unit 4001 was repaired by grind-out and weld fill; bondline was not impacted; large throat MCC configuration modifies parent and weld material of welds 10, 11, 12, 13, and 14
	2.a.6	Two-duct hot-gas manifold MCC erosion; overheating of hot-gas well causes pinholes, cracks, and dog-housing; extent of damage exceeds that experienced by higher time units	Many factors contributed to the accelerated erosion of MCC hot-gas wall; the primary cause has been attributed to an increased amount of hot-gas ingestion into the main injector coolant plenum; other contributing factors include lox post underbiasing and high resistance MCC's	Redesigned main injector flowshields have been incorporated into the phase II+ engines to decrease the amount of hot-gas ingestion; other changes included increasing the lox post biasing requirements and incorporating a revised CCV schedule to increase the MCC coolant flowrate at RPL and 104-percent power level; additionally, a new method of calculating the MCC flowrate was developed
	2.a.7	Burst diaphragm failures on engines 2015 and 2107	The presence of the silicone elastomer on the diaphragm cap is required to dampen the vibration during engine operation; diaphragms with improperly bonded elastomer are subject to failing at the coined groove intersections due to operational vibrations; the unbonded elastomer that was evident on these parts is suspected of reducing the damping effect that allows for vibration and premature failure of the diaphragm at the coined grooves	The elastomer application process has been changed to improve the bonding characteristics; because bond failures usually result from poor surface cleanliness, a solvent rinse has been added prior to applying the silicone primer; the primer cure has also been changed to room temperature rather than the previous forced-air circulating oven cure and the elastomer is vacuum processed prior to being injected into the burst diaphragm cap; improved bonding has been demonstrated in the lab when parts were processed with the modified application process
July 1980	2.a.8	MCC lee jet failure resulting in incorrect chamber pressure reading (incident No. 18)	The lee jet retainer ring was installed improperly; CH A PC failure resulted from a PC sense port purge lee jet becoming dislodged and exposing the sensor to the MCC coolant discharge pressure	<ul style="list-style-type: none"> Implemented shutdown for "major component failure" prior to SRB ignition during ground testing Redesigned lee jet retention device to prevent dislodgment Added HPOTP turbine discharge temperature lower redline
	2.a.9	Weld offsets	General problem with weld offsets due to difficulty in welding sections with thin walls	Improve welding process; design should be robust with respect to weld offsets

Table A-8. Combustion devices (continued).

Component	No.	Problem	Problem Cause	Problem Solutions
MCC (cont.) March 1985	2.a.10	Outlet manifold weld cracks (incident No. 30)	Undetected flaw in outlet manifold neck; an external leak developed; see document RSS-8595-39	Near-term solution was to reinspect MCC's to verify no CIFS, reduce the life by one half and to reinforce outlet with EdNi; the long-term solution was to go to a single piece forging

b. Nozzle

Component	No.	Problem	Problem Cause	Problem Solutions
Nozzle November 1979	2.b.1	Nozzle steerhorn failure (incident No. 15)	The fracture of the steerhorn was caused by the use of improper weld wire in the welds on either side of the T-joint; the joint failed at the time of maximum strain during shutdown	<ul style="list-style-type: none"> • Modify weld wire process • EdNi plate nozzles in the field with wrong weld wire • Initiate nozzle lab testing considering static and dynamic loads to verify math models; incorporate new steam loop design
May 1979	2.b.2	Nozzle steerhorn failure (incident No. 13)	Low-cycle fatigue associated with the 200 to 400 Hz oscillations in the start and shutdown transients; oscillations were unknown at the time of the failure	<ul style="list-style-type: none"> • Add strain gauge instrumentation to better define the start and cutoff oscillations • Redesign steerhorn duct with the capability to withstand newly identified loads; increased the wall thickness from 0.049 to 0.080 in; support the steerhorn at the No. 9 hatband and include an expansion loop in the downcomer
June 1982 to April 1984	2.b.3	Aft manifold thermal protection burned up on reentry (STS-4, 6, 8, 9, 11); no nozzle structural damage	During reentry, a shock attached to the nozzle blunt body (aft manifold lip, causing high stagnation temperatures resulting in TPS melting	TPS redesigned to eliminate blunt body at aft manifold
January 1991	2.b.4	Aft manifold slam holes cracked as a result of low-cycle fatigue; no effect on engine function or performance	Engines start and shutdown transient loads (due to flow separation) in combination with the slam hole Kt produced fatigue at the hole ligament	Fatigue arrestor bolts were designed and implemented to extend the life of the slam holes to CEI requirements; these bolts eliminate the Kt during the compression cycle of the loading
September 1988	2.b.5	G15 seal overheated and cracked (stress rupture) during STS-26	The seal heating is a result of hot gas flow being driven into the joint by nozzle coolant tube protrusion into the hot-gas flow, ovalness of the tube bundle and gaps between the MCC and the coolant tubes	<ul style="list-style-type: none"> • A flow recirculation inhibitor was designed and implemented to preclude hot gas from reaching the seal • Limits on tube protrusion and gap were established
May 1992	2.b.6	Nozzle fatigue cracks in stubout fillet radius	Fractography confirms transient fatigue mechanism as a result of transient loading and fillet radius stress concentration	Life limit nozzles to 18 transients, which is 50 percent of 36 transients (when the first crack initiation occurred)
	2.b.7	Nozzle stubout weld indications	Welding process insufficient to eliminate the existence of porosity in pores with tails	Life limit nozzles based on severity of porosity found
1982 to present	2.b.8	Coolant tube fatigue	High strains during start and sea-level shutdown transients	Criteria established for allowable leakage limits; permits class I and II leakage on hot and cold walls; class III and higher leaks are unacceptable and require repair

Table A-8. Combustion devices (continued).

b. Nozzle (continued)

Component	No.	Problem	Problem Cause	Problem Solutions
1984 to present	2.b.9	Coolant tube corrosion	Contamination seeds and sustains corrosion	Fabrication process improved to minimize residual contaminants; no leakage permitted on coldwall side; minor leakage (CL I) permitted on hotwall; unacceptable leaks are repaired
May 1993	2.b.10	Feedline elbow bracketed welds	Welds with incomplete penetration developed fatigue cracking during start and sea-level shutdown transients	Life limit established at 22 unrestrained transients with an inspection required at 11; welds reworked to extend life
July 1993	2.b.11	Stubout weld defect	Fatigue growth in a weld defect (lack of fusion) while under start and sea-level transients	Life limited by generic stubout DAR at 27 total transients with an inspection required at 18 unrestrained transients

c. Powerhead

Component	No.	Problem	Problem Cause	Problem Solutions
Powerhead	2.c.1	Fatigue of lox post creates leaks and destroys engine powerhead through lox hydrogen fire	Hot-gas H ₂ flow creates vortex shedding coupled with large static load	<ul style="list-style-type: none"> Initial solution; installation of flow shields on outside posts, two posts per shield Two-duct hot-gas manifold Super posts tested but not incorporated
	2.c.2	Crack initiation and propagation in powerhead G6 flange; potential hot-gas leakage across G6 seal	Low-cycle fatigue failure due to high strain caused by flange roll and high HPFTP turbine discharge temperatures; only two powerheads with >85 ms starts	<ul style="list-style-type: none"> TBD High start units in test will be inspected at approximately 80 starts
	2.c.3	Powerhead coolant duct welds have excessive mismatch and did not meet CEI life requirements	Baseline design deficiency; fabrication of welds difficult; wall thickness too thin	<ul style="list-style-type: none"> Utilize inner guide ring to minimize offset on later phase II ducts Increase the thickness locally in the area of the weld joint on phase II+ ducts
	2.c.4	Main injector oxidizer inlet manifold base ring welds do not meet CEI requirements; specifically welds 6, 7, 16, 17, 52, and 53	Analysis assumes all worst-case geometries, i.e., maximum offset, minimum wall thickness, material properties not well defined; design deficiency and fabrication difficulty	Actual geometries measured for weld joints 6, 7, 52, and 53 and materials test program done; weld joint 16 blended, 17 tracked by DAR; newer powerheads have forged tees so welds 16 and 17 meet CEI; robotic welding improved fabrication
	2.c.5	FPB liner tab failure on engine 2206	Manufacturing error; weld extended into radius, pump impact during installation	Considered unique
	2.c.6	Oxidizer preburner "POP's," which cause interpropellant plate deformation and potential braze joint damage	Comingling and combustion of hydrogen and oxygen in preburner lox dome due to aspiration effects during transients	Flatness check of faceplate is required if a "POP" occurs above a specified "g" level; changed start and shutdown sequence to reduce occurrence
March 1978 June 1978 July 1980	2.c.7	Main injector lox post thread/tip failures (same as item 2.c.1) (incident Nos. 6, 7, and 17)	High-cycle fatigue failure due to large fluctuating pressure loading and large static load	Replace CRES post tips with Haynes tips
July 1981 September 1981	2.c.8	Main injector lox post inertia weld failure (same as 2.c.2) (incident Nos. 20 and 21)	High-cycle fatigue failure due to large fluctuating pressure loading and large static load	Replace CRES posts with Haynes posts

Table A-8. Combustion devices (continued).

c. Powerhead (continued)

Component	No.	Problem	Problem Cause	Problem Solutions
	2.c.9	Main injector interpropellant plate cracking	High thermal gradient across the interpropellant plate and hydrogen charging of the hot-gas surface causes thermal cracking	Improved the heat shield retention system to eliminate heat shield damage and accepted the thermal cracked as a benign self-limiting condition
	2.c.10	Main injector lox post stub life	Fatigue life limited based on correlation to interpropellant plate cracking	Generic DAR based on single flight reliability and elliptical radius incorporated in 1987 to reduce stress concentration factor
July 1987	2.c.11	Engine 2106 OPB interpropellant baffle pin braze joint failure (incident No. 31)	Poor braze quality on units that were brazed in ME&T furnace; 10-percent braze on failed joint; ductile overload assisted by LCF	Verified braze quality on units brazed in manufacturing furnace to be a minimum of 50 percent braze; problem unique to units brazed in ME&T furnace; improved braze process control
	2.c.12	Cracking on I.D. of FPB diffuser	Failures of plug weld No. 3 allows diffuser to be internally pressurized	Cracked diffusers are acceptable as is; no risk to engine operation, weld No. 3 has been modified to improve weld penetration
	2.c.13	Preburner lox post transformation to martensite	Transformation can be caused by material lot variation and/or dynamic loading	Phase II, install support pins or deactivate post at higher eddy current readings; phase II+, posts redesigned and lower dynamic loads
	2.c.14	4,000 Hz vibration on the main injector lox inlet splitter vanes creates fatigue cracks	Vortex shedding of splitter due to small differences in vane trailing-edge geometry, only 20 percent of injectors show resonance at 4,000 Hz	<ul style="list-style-type: none"> • Sharpened trailing edge and increased open space between leading edge of the two splitter vanes • No 4,000 Hz resonance since fix incorporated • Gimbal bearing accels added to screen
	2.c.15	Main injector lox post outer diameter defects	Handling and fabrication damage	Analytically life limited and revision to tooling/process
	2.c.16	Heat exchanger leak detected in weld No. 4 bifurcation joint area	Inclusion in parent material	Changed material to vimvar to eliminate inclusions and redesigned to single-coil heat exchanger, thus eliminating welds
February 1984	2.c.17	Fuel preburner manifold boss crack (incident No. 29)	High stress concentration where boss is welded to manifold	Redesigned boss to be integrally machined for development engines; boss does not exist on flight engines
July 1980	2.c.18	Fuel preburner liner and chamber burn through (incident No. 16)	Injector faceplate deformation biased lox element, which caused liner erosion; low coolant pressure allowed hot gas to contact the body wall	Inspection is done to verify element concentricity; divergent liner was added to reduce hot gas circulation; metering ring was redesigned to increase coolant pressure; a thermal shield was added over the body wall, but later removed
August 1977	2.c.19	FPB housing burn through (incident No. 2)	Similar to incident No. 16	

Table A-8. Combustion devices (continued).

c. Powerhead (continued)

Component	No.	Problem	Problem Cause	Problem Solutions
February 1982	2.c.20	FPB fuel injector blockage (ice) (incident No. 24)	Water was introduced into the engine during an electrical discharge machining process performed on the test stand; subsequent drying purges ineffective; during engine start, cold hydrogen caused the residual water to freeze; resulting ice blocked several FPB fuel element inlets, causing a high oxygen to fuel ratio and high combustion temperatures	Redesigned boss to be integrally machined for development engines; boss does not exist on flight engines
January 1981	2.c.21	FPB injector lox post failure (incident No. 19)	Metallurgical examination revealed all posts to exhibit cold work at fillet radius where high-cycle fatigue crack was found; GH2 flowed into lox post via crack and ignited, which eroded off lost post tip; damage self limits during mainstage, but during shutdown, GH2 flows into lox dome through damaged element; recirculating GH2 ignited with lox residual in dome and caused remaining element damage	<ul style="list-style-type: none"> • Tighter fabrication tolerances • FPB lox element support pins • OPB eddy current inspections
	2.c.22	FPB lox post failure	High dynamic loading due to swirling flow caused by misdrilled holes	Tightened control on drilling operation, added eddy current inspection, and installed support pins; phase II+ element were redesigned to reduce stress
December 1978	2.c.23	Heat exchanger coil failure	Heat exchanger coil leak most likely caused by a welding arc burn or wall thinning due to reaming operation	<ul style="list-style-type: none"> • Changed welding procedure • Changed reaming procedure • Proof test after each HPOTP installation • Mass spectrometer leak test after each proof test and prior to each hot fire test • Visual examination of heat exchanger after every HPOTP removal
	2.c.24	Heat exchanger coil leak detected at weld No. 1	Lack of weld fusion	Revising welding and inspection procedure; designed single tube heat exchanger with no weld No. 1
	2.c.25	Lox inlet seam weld rework	Mistracking of EB weld of shells; unauthorized TIG tack usage on the root instead of face, contributed to embedded defects	Short term – shells installed – grind groove along length of EB weld root side and IVC dye penetrant and x ray; shells not installed – grind to remove defects, TIG weld fill, inspect long-term, one-piece shell, no EB weld
	2.c.26	Proof failure of powerhead duct	Duct not heat treated	Perform hardness tests on all ducts in the fleet; ensure all ducts in manufacturing obtain heat treat
	2.c.27	Liquid metal embrittlement (LME) in ASI tube brazing	Torch braze of sleeve and collar to ASI line as it passes through thrust cone caused tube to crack and braze material to fill crack if excessive heat was applied with tube restrained	Lox line borescope inspected after torch braze to verify no LME; fuel line eliminated braze of sleeve and collar and left a 0.040-in gap requirement between tube and thrust cone

Table A-8. Combustion devices (continued).

c. Powerhead (continued)

Component	No.	Problem	Problem Cause	Problem Solutions
	2.c.28	Duct cracking due to stress accelerated grain boundary oxidation (sagbo)	sagbo	Recommended to not heat pass 903, but if required to be done under ME&T supervision
	2.c.29	Secondary faceplate retainer failures	High-cycle fatigue failure due to high-flow loading	Short term – Deactivated and rodded high flow posts (8, 9, 78, and 79); long term – heavy wall secondary faceplate retainers
	2.c.30	FPB ASI line rupture	Contamination (metallic particle) lodged in ASI causing blockage; line overpressurized and burst	Graphite wrap to reinforce ASI line
	2.c.31	ASI contamination	Contamination blocked ASI fuel inlet, causing preburner ASI to go lox rich, eroding face plate inner rows	Fuel filter or redesigned fuel filter

Table A-9. Ducts and valves.

3a. Ducts

Component	Problem	Problem Cause	Problem Solutions
Ducts June 1989	Tripod linkage failure of flex joint C (incident No. 32)	Fatigue initiation at undersize fillet radii region	DAR 2296: limit use with a HPFTP with synchronous frequencies greater than 609 Hz; restriction of time accumulated at power level greater than 104 percent; restriction based on power level equivalency derived by direct correlation to the failures; verify fillet radii
June 1985	Titanium high-pressure duct flange cracking	H2 assisted sustained load cracking due to joint preload	Reduced joint preload; initiated a penetrant inspection at every 45-day interval; redesigned using new material (Inconel 718 TM); all titanium ducts have been removed from the flight program
June 1984	Low-pressure fuel duct buckling	Pressure built-up between duct and nickel plating caused by cryo pumping during propellant bleed cycle	Install helium bag; redesign foam system (bonded foam)
	Incoloy 903 TM corrosion (pitting of the flange sealing surface)	Incoloy 903 TM susceptible to corrosion	Flange groove relapped to remove corrosion; new material of plating on seal being tested at this time for possible future modification
October 1989	Flex duct overangulation	Engine yaw stiff-arm disengaged resulting in overangulation; human error	Engine visually inspected; remove and replace the articulating fuel bleed duct; complete FRT's while gimbaling the engine in the allowable envelope
January 1990	High-pressure fuel duct failure during cryogenic proof test	Ruptured at 8,000 lb/in ² on first pressure cycle to 8,360 lb/in ² , oxygen contamination of weld resulting in brittle fracture titanium weld	Palmdale facility modified/qualified for titanium welds (all flight ducts are now made of Incoloy 718 TM)
February 1992	Torn LPFT discharge duct flex joint liner	Liner was installed in a reversed direction of flow, due to reverse flex joint flanges	Liner flow direction verification implementation initiated at flex joint level, fabrication level, and engine assembly level; reinspect all fleet ducts

Table A-9. Ducts and valves (continued).

Component	Problem	Problem Cause	Problem Solutions
July 1991	Low-pressure oxidizer duct bellow ply failure	Fatigue crack propagate through inner ply; lox leaks into inner ply cavity during operation; lox expands during posttest warmup resulting in inner ply deformation and bulging	All bellows inspected posttest visually for bulging; life limit imposed per FLC 2595 (50 percent of the failed unit)
November 1990	Field site stretch fastener requirements	Requirements specified in the specification were too tight	Relaxed tolerance requirements based upon statistical analysis and revised the baseline analysis for the bolt joints; revise specifications per ECP 1197 to impose specific calibration requirements
October 1990	RS007030 preburner fuel duct failure during proof test	Duct was not heat treated; portable hardness tester results invalid on Incoloy 903™	All components in the particular heat load verified; limits of portable tester determined; reviewed heat treat practices and fabrication flowpath of hardware
	Rigid fuel bleed duct stress corrosion cracking 1985 first occurrence 1988 second occurrence	Stress corrosion cracking of CRES 21-6-9 material	All CRES 21-6-9 material ducts; application of zinc chromate primer and strontiumchromate overcoat as inhibitor (STS28 to present); reinspect corrosion protection integrity at 2, 4, and 6 years on one unit
September 1988	LPOTP discharge duct flex joint antibacklash tie pin (UCR A020205)	Tensile failure during fabrication due to a misuse of a tooling fixture	Revise flex joint and duct drawings to add notes to prevent compression of the flex joint and resulting tensile load on antibacklash tie; inspected next six flex joints via x ray to verify corrective action
July 1986	Elongation of hydraulic supply hose	Undersize braid wire; mishandling (UCR A004122)	Procurement inspection plan amended to measure and record the braid wire diameter used on each unit; measure the length of the hose before and after the proof test
August 1982	Discharge duct failure (incident No. 27)	Mounting of experimental ultrasonic flowmeter to high-pressure oxidizer duct led to HCF rupture of duct wall resulting in pump unloading, overspeed, and turbine disk failure in parallel with rubbing, internal fire, and pump housing rupture	<ul style="list-style-type: none"> Establish criteria for mountings on lines, ducts, and pressure vessels Require government approval for configuration changes from that which is certified

3b. Valves

Component	Problem	Problem Cause	Problem Solutions
Valves December 1978	MOV fire 901-225 (incident No. 12)	Fretting of inlet sleeve due to flow oscillation at 4x HPOTP shaft speed; fire initiated in the inlet sleeve to bellows flanged joint	Increase sleeve and flange thickness; sleeve bolted with 16 screws and locked with cup washers; replace laminated 302 SS shims with one Incoloy 718™ space; apply dry lube to selected interfaces
July 1979	MFV housing crack E2002 (incident No. 14)	High-cycle fatigue failure; most probable cause of crack initiation was stress corrosion cracking during heat treatment	Life limit imposed based upon an analysis correlated with failure and fillet radii in the failure region (interim fix); completely redesigned cam follower area; (increase fillet radii) and implementation of process change
	PCA accumulator cap	During proof test testing, the accumulator cap deflected and lost its sealing capability	Redesign the PCA accumulator cap to increase its thickness from 0.400 to 0.650 in and increase the length of the two attachment bolts

Table A-9. Ducts and valves (continued).

Component	Problem	Problem Cause	Problem Solutions
May 1982	OPOV seal failure (incident No. 26)	Shutdown sequence allows hot gas to propagate to ASI tee-block; seal eroded by hot-gas impingement	Modified the shutdown sequence for shutdown after 5.0 s; shutdown between 1.5 and 5.0 s requires ball seal replacement
	Pogo accumulator Z baffle cracks	Turbulent buffeting by main oxidizer flow caused high-cycle fatigue cracks, which initiated in top flange of baffle vanes	Required the vane edges to be chamfered; crack length of 0.15 in or less deemed acceptable by analysis
June 1978	Fuel inlet blocked by nitrogen (incident No. 8)	Solidified nitrogen collected in the screen in the main propellant duct and restricted hydrogen flow to the LPFP; the nitrogen was introduced into the tank during pretest chilldown operations due to an incomplete electrical interlock between the two tank vents, the forward engine chill valve, and the tank vent line, GN purge; and the incorrect positioning of the main propellant duct bleed valve and the forward engine chill valve	<ul style="list-style-type: none"> • Modification of LH tank vent line purge sequencing interlock • All run tank vent line purges converted to helium • Relocation of LH barge tank vent line purges to preclude backflow to the barge tank • Modification of test stand chilldown procedures and safe-guarding systems and procedures • Implementation of engine inlet pressure redlines

B. Solid Rocket Motor/Boosters (RSRM/RSRB)

Although, on the surface, it appears that the RSRM/RSRB is nothing but a large, simple booster, it turned out to be a very complex system with many development problems. The complexity was further enhanced by the requirement to recover and reuse the system 20 times. The magnitude and complexity of the problems listed (table A-10) point clearly to this observation. The harshness of the reentry, water impact, and recovery operations clearly added to the problems. The ablative gimbaled nozzle has been the source of many concerns and problems. The joints and seals are a constant concern and were the cause of the *Challenger* accident that destroyed the shuttle and killed the crew (Presidential Investigation Committee Report).

There were several lessons learned in the RSRM/RSRB program. Since the SRM/SRB dry weight only cost 1 lb of payload for each 10 lb of dry weight, it was a good trade to add conservatism in the design by using maximum/minimum loads, which then allowed load increases without redesign. The lesson: robustness early in a program provides flexibility to solve downstream problems.

The SRM clearly taught that an apparently simple system that contains high energy (pressure, temperature, thrust) is in reality very complex, requiring great care and attention to prevent failures. When additional functions such as providing control authority, more complexity and issues are added. From a system perspective, however, the best place to put control authority is where the potential authority resides (SRM thrust). In this case (space shuttle), major control disturbances also came from the SRM natural thrust misalignments. Taking this disturbance out at its source is good logic. SRM/shuttle reinforces the axiom that all design is a balancing act, a series of trades, giving up some things in order to get others you want or need.

The matrix (table A-10) provides a listing of many of the problems experienced in the project and should serve as a technical base for problem avoidance in the future.

Table A-10. SRM/SRB problems.

Component	Problem	Problem Cause	Solution
Aft skirt STA-2B – 1986 STA-3 – 1988 Bracket – 1994	Structural failure occurred during static test at loads below design requirement; large cost associated with NDE inspection of critical region	A weld joint in the primary load-carrying structure fails at strain levels far below the values predicted by materials testing	An external bracket has been designed to reduce the strain in the weld; the bracket was tested in the first quarter of 1994
Bracket – 1994	Failure of nominal configuration at higher than STS-3 load	Study in process	Pending
All aluminum components	Corrosion – Significant amounts of salt water corrosion are evident on much of the SRB hardware	The repeated exposure of aluminum to salt water is causing a deterioration of the SRB components	Cost-effective NDE and analysis methods are being considered
Forward skirt	Water impact and tow back damage	Forward skirts have been scrapped due to extreme water impact damage; the skirts leak water during tow back, damaging expensive electronics	Water impact damage is considered attrition; leakage problem has been reduced by improved door seals
SRB attach struts	Expensive NDE and proof testing	Conservative fracture mechanics analysis	Reduction in NDE requirements by improvements in analysis
ET attach ring	Expensive NDE and damage assessment	Water impact loading can cause significant damage to the ring	None
Systems tunnel	Assembly	Occasional problems in assembly, caused by buildups in the tolerance along the length of the tunnel	Engineering has been put in place to allow slotted holes and large washers to be used during assembly
Nose cap	Thrust post did not receive proper heat treat	Lack of vendor quality control	Thrust posts removed and replaced
SRB separation motors, 1979	Potential for plume impingement damage to orbiter tiles	Size of particles in propellant combustion products	Minimized propellant aluminum content, 2 percent; extensive test and analysis verification
SRB nose cap, 1988	Shedding of thermal protection material damaged orbiter tiles	Nose cap assembly age and TPS material properties	Developed and implemented new TPS material
Blast container system, 1988	Frangible holdown nut debris lost	Inadequate blast container design	Designed, built, tested system with plunger connected to holdown bolt by frangible link to close container
Holdown bolt hangup	Broaching of bolt in aft skirt hole during lift-off	Bolt rotation after frangible nut separation	Optimized frangible link size; minimize bolt velocity loss and constrain bolt to axial translation
TVC system, 1982	50 Hz oscillation in TVC system during ground test	Limited cycle oscillation was caused by feedback of nozzle modes through actuator differential pressure loop at the elevated hydraulic fluid temperature seen in ground tests	Established fluid temperature limits for LCC; analysis indicated any flight oscillations would be less than ground test experience; no function or structure degradation
Recovery system main parachute floats, 1982	FLOATS FOR TWO MAIN PARACHUTES ENTANGLED DURING DEPLOYMENT, DESTROYING ONE CHUTE ON STS-3	FLOATS BEING STORED ABOVE CHUTE PACKS AND FRUSTUM TILT DURING DEPLOYMENT	CHANGE FLOAT DESIGN SUCH THAT FLOATS ARE STOWED BELOW MAIN CHUTE PACKS AND ATTACHED TO RISERS
Recovery system g-switch, 1982	LOSS OF BOTH BOOSTERS ON STS-4 WHEN CHUTES WERE PREMATURELY RELEASED	G-SWITCH FOR RELEASING MAINS AT WATER IMPACT WAS TRIGGERED BY PYRO SHOCK AT FRUSTUM SEPARATION	ELECTRONICALLY BLOCK OUT G-SWITCH UNTIL AFTER FRUSTUM SEPARATION
Recovery system main parachutes, 1992	EXCESSIVE DEPLOYMENT DAMAGE	CHUTE CONTACT WITH STRUCTURE INSIDE FRUSTUM DURING DEPLOYMENT	INSTALL RIP-STOPS ON ALL MAIN CHUTES TO REDUCE DAMAGE PROPAGATION DUE TO DEPLOYMENT LOAD

Table A-10. SRM/SRB problems (continued).

Component	Problem	Problem Cause	Solution
Aft SRB components, 1981	Water impact damage	High-impact velocity	Changed main chute size from 115 to 136 ft and applied foam to aft skirt internal components
SRB aft skirt aft ring, 1982	Water impact damage	Intact thermal curtain	Reduced curtain strength and reverified functionality for ignition overpressure
Aft skirt thermal curtain, 1982	Structural failure during reentry	Aerodynamic flutter	Risk of reentry thermal damage acceptable weighed against increased water impact damage with stronger curtain
Rate gyros in forward skirt	Post separation damage	Potentially reentry/vortex shedding water impact, tow back environment	Perform gyro functional acceptance testing prior to reuse
Rate gyros in forward skirt	Gyros/mounting structure tuned with fourth SRB bending mode	Mounting structure flexibility	Reinforced mounting structure
SRB, 1979	Excessive reentry acoustics	Nozzle acoustics during reentry	Designed to separate nozzle aft exit cone at apogee
Aft skirt internal components, 1981	Thermal damage	Reentry heating/hydrazine fire	Delayed nozzle aft exit cone separation until after maximum reentry heating
Main parachutes	Lagging inflation failure to inflate	Inherent to large chutes deployed in clusters	Various improvements implemented; vent cap addition, packing changes
Reefing line cutter	Failed to fire	Pull angle too large	Acceptable risk; redundant cutter worked, low probability of high pull angle
Pilot chutes	Negative margins	Increased deployment loads due to higher SRB apogee	Waivers for higher load based on test, then the pilot chute was redesigned
Drogue deployment bag	Negative margins	Increased deployment loads due to higher SRB apogee	Waivers for higher load based on test, then the drogue bag was redesigned
Main parachute support structure	Negative margins at corner fittings and frustum shear beam	Increased deployment loads due to higher SRB apogee	Waivers due to nonsafety of flight issue, then enlargement of corner fitting bolts and holes
Main parachute deck fittings	Negative margins	Increased deployment loads due to higher SRB apogee	Redesigned fittings with larger pins and flanges
Thrust vector control system frames	Weld cracks	Lack of full weld penetration due to configuration design	Analysis indicates the weld is not required; frames are fully inspected prior to use
ASRB IEA box	Orbiter wind load	IEA box location	Value added by changing box location, reducing wing load, less than cost of new aero data bases; IEA location was not changed
SRB IEA box, 1982	Vibration criteria exceedance	IEA box hard mount	Designed, developed, built vibration isolation system
SRB APU, 1979	Vibration criteria exceedance	APU hard mount	Design vibration isolation system
SRB APU	Turbine blade root and tip cracking	Unknown	Statistics/analysis employed to determine critical crack size (0.09 in) and required time between inspections (16 hot firings, 2 flights)
ET RSS electronics panel	Vibration criteria exceedance	Panel hard mount	Designed vibration isolation system

Table A-10. SRM/SRB problems (continued).

Component	Problem	Problem Cause	Solution
SRM field joints, January 28, 1986	Leakage of combustion gases causing the 51L disaster	Seal response was inadequate for the joint displacement/rate at ignition	Redesigned the joint to decrease displacement/rate by 10X and heated seals to enhance response
SRM stiffener stubs	Cracks occur at stub holes, potential unsafe condition for reuse	Water impact	Repair damage by removing stress concentrations
RSRM igniter seals, 1988	High seal temperature required for seal resiliency, putty blow holes	Joint displacement magnitude, use of putty	Redesigned joint, reducing displacement and replacing putty with insulation j-seals
RSRM case segments, 1991	Forward case segments buckled	Water impact, slap down event	Water impact damage accepted as attrition
RSRM nozzle/case joint 1992	Gas path through polysulfide, blowby of wiper O-ring	Trapped air at nozzle assembly forced through uncured polysulfide; wiper O-ring/NBR contact not intended as seal	Extensive study indicated no change required; primary and secondary seals fully functional
RSRM aft dome Y joint, 1992	Heat treatment cracks missed by NDE	Pooling of mag/particle fluid; eddy current of painted metal	Changes in NDE procedure and analytical verification of proof test screening
RSRM, 1993	Burn roughness at 65 to 75 s burn time	Collection, entrainment, ejection of molten slag	Statistical quantification of 3-sigma mag, structural assessment, and process controls to prevent changes
RSRM forward case segment, 1993	Inclusion crack in interior membrane missed by NDE	Unknown	Safety of flight ensured by pressure proof test screening of cases
RSRM nozzle, 1994	Phenolic liner/metal debonds	Study in process	Partial solution – implementation of silane primer on metal parts
RSRM propellant, 1979	Analytical representation of stiffness	Inadequate computer methodology for modeling propellant and pressure	Developed empirical stiffness/pseudo-pressure effect using one-fourth-scale test data
RSRM propellant, 1993	Analytical representation of stiffness	Inadequate computer methodology for modeling propellant and pressure	Developed modification to NASTRAN differential stiffness code
RSRM nozzle	Excessive erosion, pocketing	Nonoptimum composite part ply angles	Changed ply angles for phenolic rings
RSRM combustion chamber	16 and 32 Hz press oscillations during motor burn	First two acoustic, organ pipe modes	Accumulated statistical data base, computed maximum 3-sigma magnitude, included effects in systems loads equations for structural assessment
SRM filament wound case	STS-2 failure under prelaunch loads	Excessive delams during manufacture	Process changed to adequately wet winding fibers with epoxy resin (program canceled, improvements implemented in Titan project)
SRM filament wound case	Excessive motor elongation at ignition	Bending deflection of steel joints and soft composite/steel connection at joints	Changed winding pattern so that composite case portions decreased in length at ignition
ASRM test segment	Segment failed at 10-percent proof load	Incorrect heat treat for massive bolted joint, NDE failure to find cracks	Recommended design specific heat treat profile be developed; removed NDE supplier from approved list
ASRM/RSRM flex bearing	ASRM torque greater than CEI specified requirement	Interpretation of requirement	Clarify division between ASRM and ASRB torque contributions

Table A-10. SRM/SRB problems (continued).

Component	Problem	Problem Cause	Solution
RSRM flex bearing	Delaminations	Uneven heat distribution during bearing cure process	Relocated heating elements on bearing hemispherical shims
Nozzle boot ring	Boot failure on DM-8	Unsuccessful boot ring redesign	Changed to a dual-ply angle design
Propellant	Cracks	Stress concentration at star grain/center post transition	Changed local propellant shape to eliminate stress concentration
ET/SRB strut	50-percent through crack	Lack of weld penetration	Improved welding
RSRM thrust	Pressure/thrust spike	Slag ejection	Statistical loads bound determined acceptable
RSRM nozzle	Joint 3 primary seal erosion	Gas paths through RTV backfill	Process change
RSRM liner, insulation	Environmental	Asbestos used in liner, insulation	Replacement material, Kevlar™ filled NBR, being developed/tested
RSRM component refurbishment	Environmental	Cleaning processes use ozone depleting compounds	Spray-in-air aqueous cleaning solution processes being developed/tested
RSRM bonds	Environmental	Bonding processes use ozone depleting compounds	Non-ODC replacement adhesives are being developed/tested
RSRM bonds	Increased findings of low density areas	Improved NDE sensitivity	Reassessment of bonds, significance of LDA's, realignment of data base
RSRM 36 nozzle nose/inlet assembly	Aluminum/phenolic unbonds, suspect ascent condition	Reentry/water impact structural/thermal loads	Bond improvement using silane primer, comprehensive team assessment indicated not ascent problem

C. External Tank

The ET, although probably the simplest of the shuttle elements due to its expendable requirement (throw away), had many complex issues and concerns. Since 1 lb of tank structural weight was approximately equal to 1 lb of payload, it was imperative to control and reduce weight. Ensuring tank break up on reentry further complicated the system. Welding and ensuring weld integrity was and is a major concern since there is over one-half mile of welds in each tank and a leak is not acceptable. Insulation to maintain propellant conditioning and propellant management was a major design issue, particularly since debonded TPS could come loose and damage the orbiter reentry heat protection tiles during the ascent maximum dynamic pressure regime. The shear size, coupled with highly accurate geometric dimensions, led to the development of complex accurate tooling and jigs that have been key to tank manufacturing quality.

Several lessons stand out from the tank program. First, NDE and proof test is a safety-critical process and is a key to producing quality hardware. This discipline, fracture control/fracture mechanics, must be a prime requirement for safety-critical hardware, particularly large cyro-propellant tanks. Second, manufacturing is a fundamental part of the design process and must be pursued concurrently with the hardware design in a systems or integrated approach. Third, if the environments are well known and the materials (hardware base products) are well characterized, criteria such as safety factors can be reduced to increase performance. Fourth, process control from the ingot, to the forming, to welding, must be characterized, understood, and controlled to achieve quality products. Since these different steps usually occur in different industrial elements, communications, etc., is vital.

The matrix (tables A-11 and A-12) gives a history of the type of problems that can be expected in the development of propulsion system propellant tankage.

Table A-11. ET problem outline.

Problems are broken down into the following categories;

- 1.0 Systems problems
 - 1.1 Shuttle systems problems
 - 1.1.1 Trajectories
 - 1.1.2 ET disposal
 - 1.1.3 Loads
 - 1.1.4 Environments
 - 1.1.5 Range safety
 - 1.1.6 Launch commit criteria
 - 1.2 ET systems
 - 1.2.1 Preflight problems
 - 1.2.2 Loading problems
- 2.0 Structure problems
 - 2.1 Design requirements
 - 2.2 Design assumptions
 - 2.3 Structural changes
 - 2.4 Fracture control
 - 2.5 Pyrotechnics
 - 2.6 Structural test
- 3.0 Thermal protection systems problems
- 4.0 Ground handling and operations problems
- 5.0 Instrumentation problems

Table A-12. ET problems.

1.0 Systems problem list

Area	Problem	Problem Cause	Problem Solution
1.1 Shuttle systems problems			
1.1.1 Trajectory effects	<p>Design trajectories are conservative</p> <p>Abort and ET disposal targeting leads to more severe induced environments</p> <p>Load relief measures to protect orbiter wind create impact on performance, control, and induced environments</p> <p>Day of launch I-load updates (DOLILU) instituted to base trajectory shaping on measured winds on the launch day in order to improve launch probability</p> <p>Generic certification leads to more severe design environments</p>	<p>Design trajectories based on 3-sigma systems dispersions lead to conservatisms</p> <p>Necessity to achieve abort and disposal targets requires flight path and attitudes that increase environments</p> <p>Max “q” trajectory changed from -2° to -6°</p> <p>Trajectories based on DOLILU protect the vehicle from excessive loads but can increase heating</p> <p>Desire to minimize preflight analysis activities led to more all-encompassing flight conditions for certification and resulted in more severe design environments</p>	<p>Accept conservatism initially and redefine dispersion values as flight data and experience become available</p> <p>Accept increased environments to ensure abort and disposal requirements are met</p> <ul style="list-style-type: none"> • Trajectories are designed for best compromise that minimizes impacts • Wind constraints were instituted • Accept performance and launch probability reductions • Wind constraints were instituted • Accept increased protuberance airloads <ul style="list-style-type: none"> • Algorithms to indicate loads for critical areas were developed to assess the acceptability of the various DOLILU options • Design thermal environments were increased to protect against the heating increases possible with DOLILU options <p>Accept more severe design requirements to minimize operational cost</p>
1.1.2 ET disposal	<p>The ET must be jettisoned after MECO in an acceptable manner</p> <p>Requirement for ET to remain intact until altitude is below 294,000 ft results in more thermal protection</p> <p>Inability to predict ET break-up altitude</p>	<p>The ET (actually its pieces that result after aerodynamic breakup during entry) must be made to impact in preselected ocean areas for ground safety considerations and in accordance with international treaties; this applies to all flight cases, whether under normal flight or abort conditions</p> <p>To minimize debris footprint, ET must not break up until it is below 294,000 ft</p> <p>Uncertainties in analysis caused an inability to accurately predict break-up altitude</p>	<ul style="list-style-type: none"> • Trajectory shaping is required to allow the ET to be released under the proper conditions to impact in the preselected locations • A propulsive “tumble valve” was included on the first 22 flights to cause the ET to tumble during entry and thus provide a more stable entry that neither “skips” out of the atmosphere nor “dives into” the atmosphere (valve was removed after flight experience showed it was not necessary for acceptable debris footprints) <p>Accept additional TPS to comply with break-up requirement</p> <p>Measurements of break-up altitude by DOD resources were used to develop a statistical data base for break-up prediction</p>

Table A-12. ET problems (continued).

1.0 Systems problem list

Area	Problem	Problem Cause	Problem Solution
1.1 Shuttle systems problems			
1.1.3 Loads	Strut loading was excessive SSME startup (and shutdown for abort cases) cause excessive loads SRM thrust variations can aggravate flight loads Change in pressurization method increased loads	Cryo shrinkage of ET affects strut loading Simultaneous startup or shutdown from all three SSME's induces extreme loads SRM chamber pressure oscillations are caused by internal motor acoustic modes and result in increased flight loads, reduced margins on intertank buckling, and caused some welds to be inadequately proof tested Change to fixed orifice pressurization increased LO2 aft dome loading	Struts were pretensioned to minimize loading effects Startup and shutdown of SSME's was staggered • Added factor to SRM thrust loads to account for oscillations • Certified structure for increased loads • Added postproof test x-ray requirements for inadequately proofed weld areas • Added factor to account for increased pressure • Certified structure for increased loads
1.1.4 Environments	Aerodynamic heating prediction uncertainties Plume-induced heating prediction uncertainties Booster separation motors impinged on ET foam during SRB separation Ascent acoustics uncertainties	Trajectory, configuration, and methodology uncertainties, and initial lack of data led to conservatism in predictions and excess TPS weight Trajectory, configuration, and methodology uncertainties and initial lack of data led to conservatism in predictions and excess TPS weight SRB separation motors must point toward ET for proper separation of boosters Trajectory, configuration, and methodology uncertainties and initial lack of data led to conservatism in predictions	As trajectory and configuration matured, more detailed aerothermal wind tunnel testing was conducted • Flight instrumentation was obtained on DDT&E flights • Additional analysis points were used for specific problem areas • Prediction methodology techniques were improved • Radiation measurements were obtained on engine and motor ground tests • Prediction methodology techniques were improved • Flight instrumentation was obtained on DDT&E flights • Additional analysis points were used for specific problem areas • Sep motors were moved to preclude direct normal impingement on ET • Testing was performed to quantify impingement effects on foam • Aluminum content of separation motors was reduced to minimize erosion of foam • An early acoustics test (IS1) provided general acoustics levels • Some flight instrumentation was obtained on DDT&E flights • Additional analysis points were used for specific problem areas

Table A-12. ET problems (continued).

1.0 Systems problem list

Area	Problem	Problem Cause	Problem Solution
1.1 Shuttle systems problems			
1.1.4 Environments (cont.)	Unsteady airflow over cable trays caused structural concerns	Normal flow over raised cable trays can cause “galloping instabilities” and extreme structural responses	<ul style="list-style-type: none"> • Several subscale tests were run • Experts were put under contract • Foam ramps were added to the ET at cable tray locations to protect from this phenomenon
1.1.5 Range Safety	Airloads on foam protuberances caused structural concerns	Airflow to flat-faced foam protuberances can cause them to shed debris	<ul style="list-style-type: none"> • Several subscale tests were run • Foam ramps were added to the ET at protuberance locations to protect from this phenomenon
1.1.6 Launch commit criteria	Commanded destruct of the ET is required for civilian safety; nonfunctioning is unacceptable and has been the cause of launch scrubs; premature functioning would be catastrophic Deletion of the linear shape charges pursued as a means to eliminate a criticality 1 failure point	Commanded destruct caused addition of explosive charges, antennae, electronics, additional TPS, ground stations, and many ground checks Removal of charges only possible if tanks can be destructed by other means	<ul style="list-style-type: none"> • Range safety electronics package was added to ET intertank area and cross-strapped to the boosters • Antennae were added to receive commands • Linear shape charges and fuses were added, which required special thermal protection • Many preflight readiness checks were instituted <ul style="list-style-type: none"> • Linear shape charge on LH2 tank removed as analysis showed it would be destructed when SRB's are destructed • Linear shape charge on LO2 tank could not be removed
	Low temperature damage	Low ambient temperatures can cause ice formation in areas where water has collected and can damage components	Constraints were placed on the shuttle system to prohibit propellant loading and launch if ambient temperature limits are violated
	Lightning damage	Lightning effects can damage or destroy the ET or its components	<ul style="list-style-type: none"> • Lightning rods were placed on the launch pad to protect the vehicle while on the pad • Constraints were placed on the shuttle system to prohibit propellant loading under lightning conditions and to prohibit flight when conditions in the flight path can cause lightning
	Wind damage	Hurricane force winds can damage the ET on the launch pad and high winds in flight can damage ET structure	<ul style="list-style-type: none"> • Shuttle vehicle is moved back into the VAB if hurricane force winds are predicted • Constraints were placed on the shuttle system to prohibit flight if the winds aloft are predicted to cause vehicle load exceedences
	Debris damage	Debris at launch or during flight can damage orbiter tiles or windows	<ul style="list-style-type: none"> • Launch pad has all loose objects removed before flight • Shuttle is prohibited from launch if ET has any external ice in certain areas or over 1/16 in of ice in other specified areas or if the ET has any condition that can shed harmful debris during flight

Table A-12. ET problems (continued).

1.0 Systems problem list

Area	Problem	Problem Cause	Problem Solution
1.2 ET systems problems			
1.2.1 Preflight problems	Difficulty in mating ET to shuttle stack Difficulty in mating ET to shuttle stack Components and materials have limited lifetimes TPS materials were thought to soak up moisture during on-pad stay time; materials performance degradation and vehicle weight gain are potential penalties Damage to the exterior foam TPS has been experienced at the pad	Lack of space between ET and boosters at forward attachment Ball fittings on ET did not match up exactly with orbiter attach points Range safety pyrotechnics, batteries, and various other components have limited lifetimes Open cell materials can soak up moisture during rain and from humidity at pad Woodpeckers and other birds have damaged the foam TPS	Shaved off some of the ET foam around forward attachment area Procedures put in place to track ages of all limited life items to ensure they will be within their limits at launch The first two ET's were painted with a moisture barrier; the closed cell foam on the exterior of the ET was later shown to be a sufficient moisture barrier and the 600 lb of paint was removed Damage repaired as required and bird repellent devices have been implemented on the pad
1.2.2 Loading problems	Propellant gas boiloff must be ducted away from vehicle The ET shrinks during loading The ET can form an external layer of ice for certain weather conditions Geysers and "water hammer" effects can be caused during the LO2 loading process and can harm the ET All "compartments" on the ET must be protected from the buildup of hazardous gases	GH2 is potentially explosive and GO2 coming from the ET louvers can affect the flammability of the ET foam insulation The cryogenic propellants cause the ET to shrink Cryogenic propellant temperatures will, with time, "cold soak" through the external insulation and cool the external surface of the ET Gas bubbles can form when cold LO2 is pumped through the initially warm, long, and vertical feedline that is used to load the LO2 tank; this can lead to geysers and "water hammer" effects Leakage from propellant line joints could cause an unacceptable concentration of hazardous gases	<ul style="list-style-type: none"> • GH2 is ducted off to a burn pond next to the launch pad • A vent hood is provided to scavenge the GO2 from the nose louvers of the ET where it is mixed with GN2 and ducted away from the vehicle • Attachment structure to the other shuttle elements must rotate to account for the shrinkage • The LH2 tank and LO2 tank must be pressurized during loading to protect structural margins and the TPS bondline integrity • A sufficient thickness of foam insulation is applied to the propellant tanks to preclude ice formation for most loading conditions • An algorithm that predicts ice formation is used to assess whether unacceptable ice is forming during loading • An "ice team" physically inspects the ET before launch • Initially a separate anti-geyser line was installed on the ET; this line was removed after several flights • A gaseous helium injection system was installed on the feedline to mitigate these effects <p>All internal compartments (such as the intertank nose cap) are continually purged with inert gases during the loading process to keep hazardous gases below specified concentration limits</p>

Table A-12. ET problems (continued).

2.0 Structures

Element	Problem	Problem Cause	Problem Solution
Lox Tank	Tank buckled during filling with water (forward ogive) Damping during modal test was very low for high tank fill levels	Filling without internal pressure Modal coupling between structures and fluid modes	Procedure change for operations Accounted for effect in pogo analysis and pogo suppression device design
Hydrogen Tank	Aft section of tank oil canned during dynamic test Aft dome buckled during structural testing of an unpressurized prelaunch condition LH ₂ tank cable tray support cracked at 80% limit load but carried 200% limit load Proof test failure of stabilizing arc frame tension splice SLA (insulation) bonding failed upon LH ₂ loading LH ₂ feedline “milk stool” failed during static test at vendor LH ₂ siphon line failed during static test; also failed random vibration test due to screen separation	Dynamic test was run ambient (flight is cryogenic), thus SRB to ET strut is canted 7° from perpendicular; cant caused rear I-beam (circular) to rotate creating canning The strut loads introduced by cryo shrinkage during hydrogen tank fill caused the aft bulkhead to buckle Glass phenolic broke in bond line due to shift but weak load path Bolt hole pattern was in line, introducing large local loads Thermal distortion coupled with lack of cleanliness during bonding Underdesigned Permanent set in material caused line to buckle	Not an operational range since cryo effects would put strut into perpendicular position Struts were prestressed in compression to reduce load for original tank; the lightweight redesign added material, eliminating the need for strut prestressing TPS trapped debris carried load; redesigned for super lightweight Redesigned bolt hole pattern to not be in line Revised procedures Redesigned Redesigned
Scasols	Pressure transducer vibration qual test failure Temperature sensor vibration test failure	Over test due to wrong input spectrum Failure in protective cover	Ran new unit to correct levels Cover not required, eliminated and solved problem
Other	Gaseous oxygen (GO ₂) pressurization sliding bellows collapsed during pressure test Protuberance airload ramps (PAL) cracked under applied deflection GO ₂ press line lower flex joint failed during static load test Range safety system (RSS) panel failed in random vibration test	Underdesigned Test up simulator of tank skin was too long and did not represent actual situation Rupture of bellows support structure (BSTRA) bellows; BSTRA was underdesigned Insulator was inadequate	Redesigned Retested with proper simulation Redesigned BSTRA Redesigned to be isolators; retested and passed
Nose Cone	RSS panel failed in random vibration test Lightweight replacement composite nose cone failed during development test	Response was larger than predicted Missed coefficient of thermal expansion and inadequate design of mating bolt holes; breakdown in system requirements	Redesign Redesigned unit to proper systems requirements

Table A-12. ET problems (continued).

2.0 Structures

Element	Problem	Problem Cause	Problem Solution
Fracture Mechanics	Concern over crack acceptance criteria after <i>Challenger</i> accident	Formed Blue Ribbon Panel of fracture experts to review approach and make recommendations	Criteria and procedures were changed; supported by detailed analog coupon testing of cracked specimens
Super Lightweight Tank (SLWT)	Welding problems occurred on the new AL-Li material; priming and repair Fracture toughness requirements in conjunction with separation of yield and ultimate was hard to achieve	Material properties invalidated weld procedures The chemical makeup and heat treat properties were very sensitive to these parameters	New procedures were developed Tight procedures control; lot acceptance of each forging

3.0 TPS problem list

Area	Problem	Problem Cause	Problem Solution
TPS problems	Foam surface irregularities on acreage areas Foam cannot cover all metal surfaces, thus allowing ice formation Foam becomes debris Foam becomes debris Foam becomes debris Foam becomes debris Foam becomes debris	Spray system leaves waves on surface due to spray pattern Areas which require motion, such as rotating brackets, cannot be covered with foam Voids under foam can expel foam pieces during flight Cracks in foam can cryopump air under the foam, which can expel foam pieces during flight Spraying of foam over stringers resulted in “rollover,” which left “lobes” of foam around the stringers that were a debris source The “two-tone” foam design for the intertank area suffered voids between the layers and shed foam pieces during flight Areas of foam over ablators could be expelled during flight due to trapped gases from ablators outgassing during the foam spraying process	<ul style="list-style-type: none"> • Spray pattern refined to minimize wave height and period; waviness factors (to account for increased heating) were introduced into heating rates • Criteria were placed on production sprays to achieve acceptable wave height and period • Areas that cannot be foam covered were minimized • Ice debris from these areas were proven acceptable to the shuttle system <p>Application procedures were refined to minimize void formation</p> <p>Application and repair procedures were refined to not allow cracks</p> <p>A “two-tone” foam system was instituted that filled in the stringer areas with one foam and then applied a cover layer of foam over the now noncorrugated stringer area; this avoided the fragile “lobes” of foam</p> <ul style="list-style-type: none"> • Hundreds of vent holes were drilled in the possibly affected areas to vent the voids and avoid the mechanism to eject foam pieces • A new spray method was developed to allow a design change to a single foam layer <p>Procedures were changed to provide vent holes around the ablators during the foam spray application; after the ablators complete their outgassing, the vent holes are closed out</p>

Table A-12. ET problems (continued).

3.0 TPS problem list

Area	Problem	Problem Cause	Problem Solution
TPS problems (cont.)	Foam becomes debris	Foam over protuberances that presents a flat face to the aerodynamic flow will experience airloads that will break off pieces	Ramps made of foam were placed in front of these protuberances to shield them from the aerodynamic flow
	Ablator causes debris	Ablator bondline “glassifies” at cryogenic temperatures and will break when the ET is pressurized to flight levels and expands from a nonpressurized cryoshunk condition	Procedures were instituted to pressurize the tanks during loading
	Ablator causes debris	Ablator spray formed on cable tray covers shrunk during cure process and created unacceptable bondline stresses	Process was changed to adhesively bond “preshrunk” ablator pieces onto the cable tray covers
	TPS materials must comply with EPA mandates	Original foam systems do not comply with EPA requirements for environmentally friendly materials	Large effort underway to develop new foams that do not use CFC-based blowing system

4.0 Ground handling and operations problem list

Area	Problem	Problem Cause	Problem Solution
Ground and operations problems	14,000 lb of lox inadvertently drained from ET; could not recover before launch window closed	Operator did not close valve at proper time	Procedures and training improved to avoid a recurrence
	GO2 vent louvers build up unacceptable ice	Cold GO2 vapors exiting louvers condense moisture from atmosphere on louvers and form ice	A vent hood system was installed on the launch pad to duct away the GO2 and provide a warm, dry nitrogen purge around the vent louvers
	A part of the launch pad vent hood system, a plasticized fabric “collar” around the louvers, can adhere to the topcoat on the foam and pull out foam pieces at vent hood retraction	When the topcoat is too thick, the topcoat adheres to the warm surface of the bag and pulls away from foam at retraction	Controls have been placed on the thickness of the topcoat layer

5.0 Instrumentation problems

Area	Problem	Problem Cause	Problem Solution
Instrumentation problems	Ullage pressure sensors had erratic readings	“Stiction” problems with that type of sensor caused erratic readings	New type of sensor that does not suffer “stiction” problems was qualified for use
	Development flight instrumentation installations were said to allow ice formation	Foam could not be placed over “instrument island” locations or data would be compromised; ice could form on these islands	<ul style="list-style-type: none"> • DFI islands were removed from STS-1 if they were on the orbiter side of the ET • Island mountings were improved to reduce ice formation potential

V. HUBBLE SPACE TELESCOPE

The Hubble requirements are: (1) long-term lifetime on orbit with planned refurbishment and orbit reboost and (2) highly accurate pointing control for science created many design and operational

problems; all of which have been solved or worked around, producing a very successful program. HST is a program that has demonstrated the merit of designing for refurbishment, as was successfully demonstrated by the repair mission and the subsequent science achieved. The lesson is clear: design high-performance systems for operations and maintenance so that success of the mission is ensured. Hubble taught the merit of Government/contractor teams working together at the industrial plants to solve critical problems.

Hubble reemphasized the performance requirements/sensitivity lesson, adding the footnote that for the high-performance systems, nothing can be left to chance, from design, to manufacturing, to operations. Each is as critical as the other and must be pursued with diligence. The previous discussion, in conjunction with the following problem matrix (tables A-13, A-14, and A-15), should give a good feel for the thermal, structures, and pointing control areas. It is also obvious from the Hubble experience that the technology is available to develop highly accurate pointing control systems for large telescopes of the future.

Table A-13. HST pointing control problems.

Component	Problem	Cause	Solution
Solar array	Solar array did not meet its disturbance torque requirement	Thermal deformations caused energy to be stored and released suddenly throughout the orbit, producing large pointing errors	Since the disturbance occurred at the two lowest natural frequencies of the solar array, the pointing system control law was redesigned to produce an augmented gain at these two frequencies and was uplinked to the HST computer; this fixed the problem, enabling the pointing system to meet its specification
Antenna pointing system	When commanded to certain regions of its gimbal domain, the motor torque limit was exceeded and the HST entered safemode	The cable that provides power and data for the antenna and its pointing system was improperly dressed and protruded into the gimbal space, interfering with antenna motion	The region where interference occurred was restricted from use in the ground software, reducing the allowable gimbal space by a few percent
Reference gyro assembly	Over the course of the first 3 years of the operation, three of the six gyros failed, requiring elimination of their use	Electronics failures in wheel drives and torque rebalance loops	The failed components were replaced by redundant units with no effect on system performance
Reference star selection software (ground)	Stars tracked in the fixed-head star trackers were often not the correct reference stars, causing errors in attitude reference and failure in finding guide stars	The ground software used to select reference stars did not adequately allow for magnitude and position uncertainties in the star catalog	The ground software, used to select reference stars, was modified to take into account catalog peculiarities

Table A-14. HST structural problem outline.

- 1.0 Metering Structure Joints
 - 1.1 Metering Truss Thermal Instability
 - 1.2 One-Half Scale Metering Shell Thermal Instability
- 2.0 Secondary Mirror Support Structure
 - 2.1 Thermal Instability
- 3.0 Bathtub Fittings
 - 3.1 Flight Focal Plane Strength Failure
 - 3.2 In-House Development Program Strength Failure
- 4.0 Laminate/Ply Control
 - 4.1 Metering Truss Strength Failure
 - 4.2 Fine Guidance Sensor Warped Parts
- 5.0 Composite Structure Damage
 - 5.1 GSE Impact Loading of Focal Plane Structure
- 6.0 Metal Matrix Structure
 - 6.1 High Gain Antenna Welding Microcracks
 - 6.2 High Gain Antenna Process Control Failure
- 7.0 Honeycomb Structure
 - 7.1 SSM Aperture Door and Aft Shroud Bulkhead “Dings” and Debonds
- 8.0 Metallic Structures
 - 8.1 OTA Main Ring Loose Joints
 - 8.2 OTA Main Ring Attachment Joint Installation Problem
- 9.0 Nonmetallic Properties
 - 9.1 Unclear Property Requirements

Table A-15. HST structural problems.

Component	No.	Problem	Cause	Solution
1.0 Metering structure joints Metering truss thermal instability	1.1	Thermal growth of truss out of specification	Laminate plies perpendicular truss axial dimension	Removed laminate from line of action; strut passes through slot in ring and clip ties ring to gusset
	1.2	Axial thermal growth of shell out of specification	Field splice ring had laminate plies perpendicular to axial	Removed field splice; added a shell bay
2.0 Secondary mirror support structure Secondary mirror support structure thermal instability	2.1	Unacceptable thermal effects from uniform and gradients	Uniform motion of forward ring moved mirror along longitudinal axis; gradients across forward ring tilted mirror relative to longitudinal axis	Redesigned mirror support structure symmetrical about plane of forward ring
3.0 Bathtub fittings Flight focal plane strength failure	3.1	Failure of fittings during structural load tests	No composite fibers, only epoxy between base and gusset sides	Replaced composite fittings with titanium ones
	3.2	Inconsistent failure loads even with continuous fibers and epoxy bonded on corners	A small business source was not able to control processes very well	Current fitting designs are not amenable to composite fiber/epoxy processes
4.0 Laminate/ply control Metering truss strength failure	4.1	Failure of cylinder strut to meet thermal exposure tests	One ply had been omitted in layup processing	Changed fabrication procedure; kits with prescribed with preassembled parts
	4.2	Numerous warped composite flat plate parts	Lax receiving inspections	Do not allow subassemblies to be adjusted to account for warped parts
5.0 Composite structure damage GSE impact loading of focal plane structure	5.1	During removal of FPS from thermal bakeout, hardware was impacted by overhead hoist bracket	Inadequate maintenance and inspection of GSE hardware	Scheduled inspections of GSE; kinetic energy analysis of impact NDE
6.0 Metal matrix structure High gain antenna welding microcracks	6.1	Extensive microcracks in weldment of 6061-T6 HGA	Not clear since material is very common, no chill bars used during welding	Low stresses, adequate life, and proof tests allowed use of cracked hardware
	6.2	Inability of vendor to build a spare component with the exact equipment, processes, etc.	Contractor lost control of processes; a second vendor also unable to complete part	Composite processing is not a pure science; a preproduction part was used
7.0 Honeycomb structure SSM aperture door and aft shroud bulkhead “dings” and debonds	7.1	Inspections revealed numerous debonds and “dings”	Due to size of components and difficulty with tooling; also prime had poor relations with sister fabrication facility	Aft bulkhead accepted as is because of small debonds; aperture door was rejected; new flight unit built
8.0 Metallic structure OTA main ring loose joints	8.1	Hollow box section Ti ring had loose or missing blind bolt collars on joints and component attachment	Used blind bolts with no preload; inability to properly use bolt installation equipment; use of inexperienced people for less profitable programs	Fixed loose collars by welds; some were drilled out with new fasteners installed; more surveillance by quality in these cases
	8.2	Helicoils protruding above surface	Poorly installed helicoils; use of inexperienced people for less profitable programs	Driven below surface where tool could grip; others were removed and new ones put in
9.0 Nonmetallic properties Unclear properties requirements	9.1	Inconsistent material properties and allowable with contractors; resulted in varying number of material test samples available	Program failed to specify the exact requirements desired for the hardware	Clearly define requirements of property and allowables; i.e., “A basis” or equivalent versus a typical of nonstatistical value

VI. SPACELAB

The Spacelab has demonstrated, in an excellent manner, the concept that a manned/unmanned reusable laboratory for the space shuttle can accomplish a variety of scientific missions with low maintenance and refurbishment of the basic hardware. The Spacelab has not been without problems since it first of all has a manned laboratory module that not only has to have life support, but also must be a laboratory with power, thermal control, etc., for running and observing scientific experiments. In other words, it is a scientific laboratory, with all that implies, operating in the space environment. Those features are not just limited to the manned modules, but include the pallets located in the orbiter payload bay, used for remote sensing, etc., of Earth and space and for launching satellites. The system not only must be rugged enough to withstand launch and landing environments, but also must operate in zero g. In many cases, it must isolate the experiments from the disturbances caused by the orbiter and the crew.

The matrix (table A-16) lists some of the problems that have occurred during the program. These problems and their solutions were complicated somewhat since Spacelab was built by the ESA and operated by the United States. The cultural/communication problem was solved and a highly successful program has resulted.

VII. TETHERS IN SPACE (TSS-1)

Tethers offer many advantages for certain on-orbit tasks; however, the complexity of the dynamics and control problems are great (see the previous discussion). The design problems were compounded by the fact that the system could not be fully verified on ground due to the large length, vacuum, thermal, gravity gradient, zero g, and other environments that could not be simulated on Earth for this large system.

The one great lesson TSS-1 taught was “always verify even the smallest change made to the hardware late in the program and particularly at the launch site.” The changing of one bolt, while TSS-1 was awaiting installation in the orbiter for launch, without verification resulted in a basic loss of the mission (table A-17).

A complex mission such as TSS-1 requires extensive planning and crew training to ensure a successful mission. Nothing can be left to chance. This planning and training must start long before the mission. In fact, it of necessity should be a fundamental part of the design process, as occurred for TSS-1.

Another key lesson: utilize the expertise that exists in the institutions and industrial complexes as well as Governmental agencies to help understand and solve problems, as well as design issues. This was done for TSS-1 and contributed greatly to the solution of the tether problems.

VIII. INTERNATIONAL SPACE STATION

It may be premature to discuss *ISS* problems since it is still in development; however, some of the problems that have occurred offer valuable insights and lessons. A major problem has been the many changes mandated by the changing economic problems of the United States. Adjusting to the budget constraints and redesigning to meet these constraints would be an interesting study within itself, but is not included in this report. As the concepts, designs and hardware testing have evolved, problems have occurred, particularly in the life support systems, which must function long term in space with minimum maintenance and refurbishment. The same is true for the meteoroid debris protection system, which also must function in space environments for extended periods. In other words long-term space stations require the proper life support for the crew and protection from the harsh environments such as debris. In addition to the previous discussion, the matrix (tables A-18 and A-19) lists some of these additional problems.

Table A-16. Spacelab problems.

Element	Problem	Problem Causes	Problem Solutions
Spacelab Module Rack	DPM has high stress levels for (1) MPE-to-DPM mechanical assembly (MA) interfaces, (2) MPE-to-Spacelab rack interfaces, and (3) DPM MA-to-Spacelab rack interfaces	High stress levels due to the stiffness and large mass of the DPM in lower portion of rack 8 (especially the MA)	Minimum “close fit” interfaces were maintained for the interfaces between MPE, DPM, and Spacelab rack using match and drill for MA and epoxy in selected slotted holes in the rack
Spacelab Module Underfloor	Negative margins calculated	Utility equipment located in the underfloor in front of rack 9 is heavier than previously analyzed	Used maximum torque on bolts; showed friction would hold bolts; had redundant bolts
Spacelab Module Rack	Negative margins calculated for GFFC front shear panel-to-rack interface	Weight of GFFC inducing high stress levels to rack	To obtain a “close fit” interface between the shear panel and rack, eccentric bushings used in rack bolt holes
Spacelab Module Floor Support Structure	During the roll-in of the floor into the module on IML-1, interference between the floor and floor support structure occurred	Threaded portions of the bolts protruded beyond the nut	Remove the threaded portion of the HIGH LOK bolts that protrude beyond the mating nut
Spacelab Module Floor Support Structure	Measurements of the USML-1 floor showed that the floor exceeded the SPAH envelope	Interference between the payload cable trays and the floor structure; measurements indicate that cable trays exceed the SPAH envelope	Remove the threaded portion of the HIGH LOK bolts that protrude beyond the mating nut
Spacelab Emergency Breathing System (SEBS)	According to analysis, the SEBS bracket will not carry the current loads being applied	With the IML-1 loads applied, the lower position mounting bracket would yield, then fail	A safety strap will be added to the SEBS mounting configuration to ensure the SEGS will not separate from the forward and aft end cone mounting bracket
CGF Experiment	CGF size and weight would not fit within normal Spacelab rack requirements		Special rack designed for GCF that meet Spacelab rack safety and interface requirements
AEPI Experiment	AEPI landing unlatched in payload bay	Latching mechanism would fail to relatch after experiment operation	Dynamic and static assessment determined that landing unlatched would not be a hazard to orbiter
Atlas-1 Orthogrid Strut Gapping/Looseness	Strut gapping effects not included in Atlas-1 pallet verification coupled loads models; MPE modal surveys indicated some struts had reduced effective stiffnesses due to gap/looseness	Strut gaps result from a buildup of tolerances between end clevises, clevis pins, and monoballs	Dynamic analyses performed on VCLA models with struts modified per static analysis results; minor frequency shifts were within established guidelines; therefore, VCLA models are considered test verified
Spacelab Rack Overhead “X” and “Y” Rod Attach Hardware	Attaching racks to overhead support structure is difficult task	Use of a one-time-only tab washer; access area for working with washer is severely limited and shear carrying bolt is too short	Modification: supplying correctly dimensioned bolts, eliminating the tab washers and replacing the nuts with steel lock nuts
Spacelab Module Rack	Spacelab rack and rack-to-module load carrying capability needed to be increased	Acceleration and load levels that Spacelab must withstand during shuttle launch and landing conditions has increased due to revised forcing functions	Strength improvement to module rack-to-rack and rack-to-overhead support attachment

Table A-17. Tethers in space problems.

Component	Problem	Problem Cause	Problem Solution
Tether	Skiprope	Tether current flow interaction with Earth's magnetic cause	Observers, orbiter yaw maneuvers, passive dampers, phased current flow
Tether	Skiprope observation	Need for intelligence about skiprope for corrective actions	Satellite attitude data to input to Kalman filtering and fast Fourier transform techniques
Tether	Libration	Interaction between Coriolis and gravity gradient forces	Smooth and continuous reeling command profiles, observers and crew techniques for intelligence for orbit translational maneuvers
Tether	"Bobbing" tension oscillation	Interaction between endmass accelerations and stretch of the tether	Minimize thruster firings and active damping through reel motor back EMF
Satellite	Satellite attitude oscillations	Interaction with tether tension and slack tether, through modal interaction and endmass movement, thruster misalignments	Satellite attitude reference, active automatic and manual rate damping by thrusters
Structure/level wind	Interference	Inadequate clearance after late modification	Redesign for clearance
Tether/guide pulleys	Tether jam	Inadequate tension management, lapses in guides	Improved guides and tension management through mainly procedural changes
Deployer system/satellite	Hazard of orbiter collision	Deployer hardware failure causing uncontrolled reel-in of satellite	Redundant overspeed detection circuits to automatically shut down reel motor
Tether	Hazard to orbiter of tether severance and recoil toward the orbiter	Inadvertent severing of tensioned tether by orbital debris of meteoroids	Automatic detection of slack tether conditions through tension monitoring at both deployer and satellite for rapid execution of orbiter evasive maneuvers

Table A-18. Space station structural problems.

Component	Problem	Problem Causes	Problem Solutions
Rack	Low fundamental frequency (<25 Hz)	Open box construction due to crew utilization of requirements	Additional stiffening added to rear of rack and utility panel made structural, which raised frequency to 25 Hz, while retaining capability for front access
MPLM/rack coupled loads	High interface loads	Rack interface fittings not designed to high load levels, which the coupled loads analysis discovered	Aluminum fittings were replaced by titanium
Meteoroid/orbital debris shielding for space station manned modules	Inability to meet PNP requirements in U.S. manned modules without complete shield redesign and excessive thickening of module skin Safety concerns with fully assembled international module cluster having high likelihood of penetration	Growth in debris environment at station orbits Growth of debris and exposed size of module cluster taken as a whole	Redesigned M/D shield employing aluminum plus composite sandwich construction, integrated within existing structure and envelope raises PNP back to original requirement; verified via impact testing to predict protection enhancement and thermal performance, via mockups to verify physical interface with existing module structure Developed analytical tool to quantify and maximize crew survivability following orbital debris penetration; verified via internal module impact effects testing, hazards effects analysis, including cooperation with DOD in crew survivability; implemented improved crew operations with JSC astronaut office and Russian station partners
Structural design and verification criteria	Lack of common design and verification criteria for NASA and international partners	Multinational program managed by three different NASA field centers	Common criteria developed by NASA and contractor teams led by senior NASA managers <ul style="list-style-type: none">• Structural design and verification• Fracture control• Window design and verification
Glass window design	Sustained stress over 15 years due to on-orbit pressure causes static fatigue	Moisture content of ambient air causes stress corrosion effect in glass microcracks under stress	Conservative fracture mechanics methodology employed in window design NASA/FLAGRO updated with glass equations Proof testing of each pane required to screen initial flaws larger than those assumed Windows designed such that tension is not on inside surface of the pane (damage tolerant) Cavity between primary and redundant pressure panes vented to space to reduce exposure to moisture

Table A-19. Space station ECLSS problems.

Element	Problem	Problem Causes	Problem Solutions
Hab element	Incomplete waste water chemical constituents for ground testing	Ersatz mixtures utilized for certain waste streams	Use of “real” waste water sources when available
Hab element	Water system not completely sterile prior to testing	Microbial growth in test systems (particularly in plumbing dead legs)	Elimination of plumbing dead legs, sterilization of system with hot water at 250 °F
Hab/Lab element	Cyclic variations in cabin temperature, CO ₂ partial pressure, and dew point	CO ₂ removal subsystem return air exhausted into cabin	CO ₂ removal subsystem exhaust routed to upstream of condensing heat exchanger
Hab/Lab element	Entrained water generated upstream of CO ₂ removal subsystem	Silica gel breaks down when subjected to liquid water	Modify CO ₂ removal subsystem by adding layer of Zeolite 13X™ upstream of silica gal
Hab/Lab element	Trace contaminant control subsystem (TCCS) plumbed downstream of CO ₂ removal subsystem	Trace contaminants removed by CO ₂ removal subsystem rather than TCCS	Plumb TCCS so that cabin air is pulled directly into TCCS
Hab element	Overall system complexity, component redundancy	Separate loops utilized for potable and hygiene water reclamation	Potable and hygiene loops combined into single water reclamation loop

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