

ORBITER STRUCTURAL DESIGN AND VERIFICATION

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ABSTRACT

The Space Shuttle development program provided the opportunity to challenge many of the established practices and approaches used in prior manned-space-flight programs. The most significant accomplishments and resulting precedents which emerged during the structural development of the Space Shuttle and the Space Shuttle Orbiter are reviewed. Innovations in criteria, design solutions, and certification are highlighted, and brief comments on the lessons learned are included.

INTRODUCTION

As the final Space Shuttle concepts matured, the new challenges offered by the Shuttle main engines, the Orbiter thermal protection system (TPS), and the Orbiter avionics systems became clearly visible. The engineering challenges faced by designers of the primary structural system were created by features for which no precedent existed and thus provided the momentum for creative and innovative criteria, approaches, and hardware features. Simply stated, the reusability and mission flexibility of the vehicle, the weight sensitivity of the Orbiter to the mission requirements, and the cost consciousness of the project provided the constant pressure for innovative design challenges. Although the expertise of NASA and the supporting aerospace team had been focused on high-reliability single-use boosters and spacecraft, early studies identified no significant technology issues with structural reusability. As engineering concepts emerged, it became more obvious that a considerable change in the size and design life of the spacecraft would force the emergence of new concepts and precedents.

DEPARTURE FROM FULL-SCALE TEST SIMULATION

Previous experience had led to complete spacecraft vibration and acoustic testing as well as complete entry vehicle thermal testing. Initial proposals emerged for subjecting forward and aft sections of the Orbiter to vehicle-level acoustic tests. After detailed technical and programmatic examination, the differences with the past precedents became clear. Secondary structure and installations were to be designed in accordance with life requirements for the Orbiter. Generic installation concepts would evolve throughout the vehicle and would require additional development and verification. Prevention of acoustic fatigue of the primary structure emerged as a serious requirement (fig. 1). This factor complicated the structural reliability since extensive coverage of the Orbiter with external TPS tiles plus internal insulation made regular inspection much more costly and impractical. The technical issues associated with the complexity of the environmental simulation, the remaining unanswered structural and systems issues, the availability of the Main Propulsion Test Article (MPTA) to aid in vibration and acoustic development, and, of course, the significant saving in test article and development test cost led to the challenge of developing the reusable Orbiter structure without the full-scale spacecraft vibroacoustic test articles. This challenge was addressed in the structural and system acoustic fatigue program and is discussed herein.

The large-scale transient thermal response of the entire Orbiter was also recognized as a significant challenge for the structural discipline. Efforts and approaches such as those implemented by the British Aircraft Corporation for development of the Concorde were technically and programmatically evaluated. The Concorde approach was to convectively heat the entire airframe using a special shroud and heating facility while applying mechanical loads simulating the in-flight load spectra. In this program, creep and fatigue issues for steady-state flight conditions were addressed. The Orbiter design was sensitive to the peak combined transient thermal stress as it related to out-of-plane deflection of the airframe and to airframe panel stability. Full- and quarter-scale airframe test programs were evaluated to address the challenge. This challenge was finally addressed through a combined program of criteria, structural testing, and flight testing. The resulting precedents are discussed herein.

DEPARTURE FROM ABORT AND TRAJECTORY UNIQUE DESIGN

Pre-Shuttle NASA experience had been focused on clearly defined mission models and reference trajectories. Abort scenarios would emerge as the most demanding structural event, and complexity, cost, and weight would be designed into the spacecraft. As the flexibility of the Shuttle was exam-

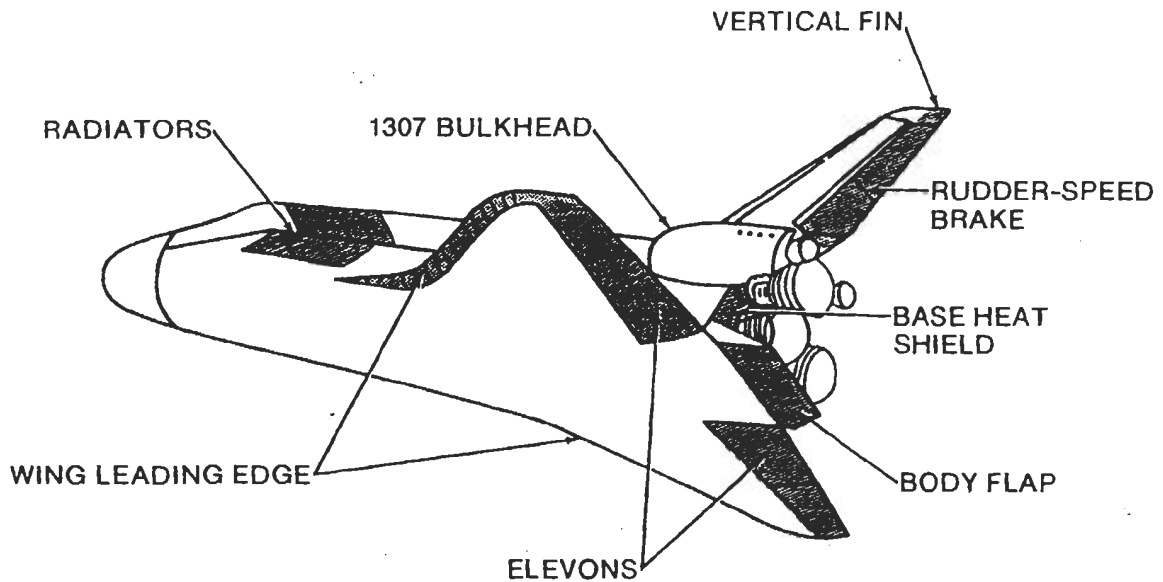


FIGURE 1.- PRIMARY STRUCTURE AREAS SUBJECT TO ACOUSTICS.

ined, it became clear that, as the performance parameters of payload weight, orbital inclination, and orbit altitude were varied over feasible ranges, the flexibility available through use of the engine throttles led to a myriad of variations not easily examined using direct trajectory simulation for design. Concurrently, the required system redundancy specified and the Orbiter weight sensitivity led to the design philosophy of minimum design impact for abort requirements. Such an early NASA design approach was unprecedented. The direct impact to the hardware structural weight was minimized by design conditions for loss of only one main engine using rational statistics to avoid the worst possible case. Descent operational maneuver and landing placards were specified for the resulting heavy-weight entry and landing. This approach, of course, put exceptional pressure on the abort planners to invent concepts which would "stay within" the operational capability - an effort achieved with commendable expertise.

SERIAL DESIGN EVOLUTION OF AIRFRAME DESIGN LOADS

Results of early Space Shuttle studies had shown a clear performance benefit with increasing dynamic pressure (q) and minimum throttling. A conscious design decision was made to limit the maximum dynamic pressure to 650 psf nominal and the maximum longitudinal acceleration to 3g axial. The desired effect was to limit the q -sensitive parameters such as peak differential pressure, buffet intensity, aerodynamic noise, aerodynamic loading, control authority, and flutter requirements within manageable bounds. The constant weight sensitivity and development of the emerging flexible capability of the Space Shuttle continued and resulted in changes to the ascent loading requirements late in the design cycle. Changes to the detailed structural criteria were necessary to minimize the impact of updated ascent configurations and the serial update of the required data bases. Although Columbia and Challenger were designed to an early data base designated the 5.1 loads data base, it was clear that a later certification to a 5.4 loads data base would have to be incorporated. The separate certifications of OV-101, MPTA, and OV-102 to different requirements with different data bases and criteria were a significant challenge. Resource pressures forced a serial approach. The Enterprise was certified by analysis and placarded to 80 percent of limit load. The MPTA was certified by analysis. The Columbia was certified by analysis and testing to the 5.4 loads data base following partial modification performed in the field. Further, the Columbia was restricted to a reduced operational envelope before final modification. The Challenger was structurally tested and returned to the fleet. This approach is unprecedented in NASA experience and is discussed further herein.

SERIAL DESIGN OF AIRFRAME AND TPS INTERFACE

The classical design process dictated an early definition of the structural moldlines and structural details before the integration and final sizing of the TPS. Early sensitivity of the TPS to short wavelength, out-of-plane deformations led to the selection of a nonbuckled external skin design. Since the TPS thickness was not defined and would vary depending on final requirements and

aerodynamic moldlines, a significant challenge was an adequate but minimum weight definition of three-dimensional temperature gradients and stresses. Detailed attention to the thermal stress additions which would contribute to skin buckling was also required. The approach to design definition and resulting performance was a worthy challenge and is discussed further herein.

CRITERIA INNOVATIONS

DEFINITION OF DESIGN LOAD ENVELOPES

As the critical loads analysis studies progressed, it became obvious that trajectory simulation of high-q ascent and descent was sufficiently cumbersome to prevent surveying the entire flight envelope for hardware design cases. For ascent, an entirely new precedent, which was analogous to the airframe V-n (velocity versus load factor) diagram and was named the squatcheloid, emerged. The detailed implementation of this approach is discussed in reference 1. Because this concept preceded the development of the ascent control system as well as the multimission and real-wind trajectory simulations, the criterion generated much debate. There was genuine concern that the criterion would force the vehicle to fly through a "tube" of $q\alpha/q\beta$ versus Mach number values which, at that date, the ascent control community had serious reservations about the control system capability to handle. After numerous reviews, the weight penalty and the credibility of the load survey associated with any other alternative resulted in the adoption of the squatcheloid; thus, a new precedent was set and design surveys could proceed.

The impact of this innovation was readily obvious. The Shuttle vehicle loads analysis was capable of surveying hundreds of potential design conditions within the flight envelope. Load surveys and indicators were evaluated even though the trajectory surveys had not been performed. The thrust structure could be designed for compatible engine deflections, consistent with the control system and engine mixing logic rather than with a worst case geometric mix. This criterion also became an active system integration tool. It allowed the performance, flight control, and structure disciplines to work in parallel rather than in the serial manner required with trajectory simulations and evaluations.

A similar situation existed with respect to descent maneuver loading requirements since only nominal trajectory-based data had emerged. Early ballistic trajectories did not require any significant maneuvers and did not provide any meaningful "design to" envelopes of control surface constraints required for load surveys. The initial maximum speed was defined as that required to stall the actuators. The structural criterion was baselined using Mach-number-dependent V-n diagrams illustrated in figure 2. The maximum speed, equivalent to 375 psf, was derived from upsetting the nominal trajectory and recovering and was tempered with understanding of the entry control system limits. Modifications to aircraft maneuver requirements such as limiting the maximum yawing load factor to $3/4g$ were also derived. Envelopes for the control surface were generated so that a complete loading survey could be performed. Since no deterministic flight conditions to justify the myriad of descent cases existed, this criterion also generated serious review. The criterion was found to be the logical alternative and set a precedent of deviating from deterministic ballistic load definition.

DEFINITION OF COMBINED STRESS CRITERIA

Once the trade-off studies showing the relative insensitivity of the unit weight of structural effective thickness plus TPS unit weight to maintain a temperature of 350°F (fig. 3) were performed, the decision to preclude buckled external skin became the baseline approach. Using techniques such as those documented in reference 2, the frame and rib spacings were set for minimum structural weight. As the design concepts evolved, it became apparent that the local skin deflections amplified by the beam column effect over relatively short wavelengths were important to the induced through-the-thickness stresses in the TPS. A consistent approach to and method of combining the stresses were needed to assure adequate and uniform specification of skin peak stress. The criterion of figure 4 was specified after some debate and trade-off study. This level of specified combined stress criterion was unprecedented for design of manned spacecraft. As weight-saving pressures mounted, the assurance provided by the criterion contributed to further sophistication in the skin deformations. The post-heating criterion on skin buckling was relaxed from no buckling at 115 percent of limit load to 100 percent limit load.

EMPHASIS ON FRACTURE, LIFE, AND ALLOWABLE DEFORMATIONS

Studies were performed to assess the approach to implementing fracture control on the primary airframe. Fracture control plans were generated by the contractors and approved. The importance of the fracture requirement, the life requirement of the structure, and the ratio of yield to ultimate

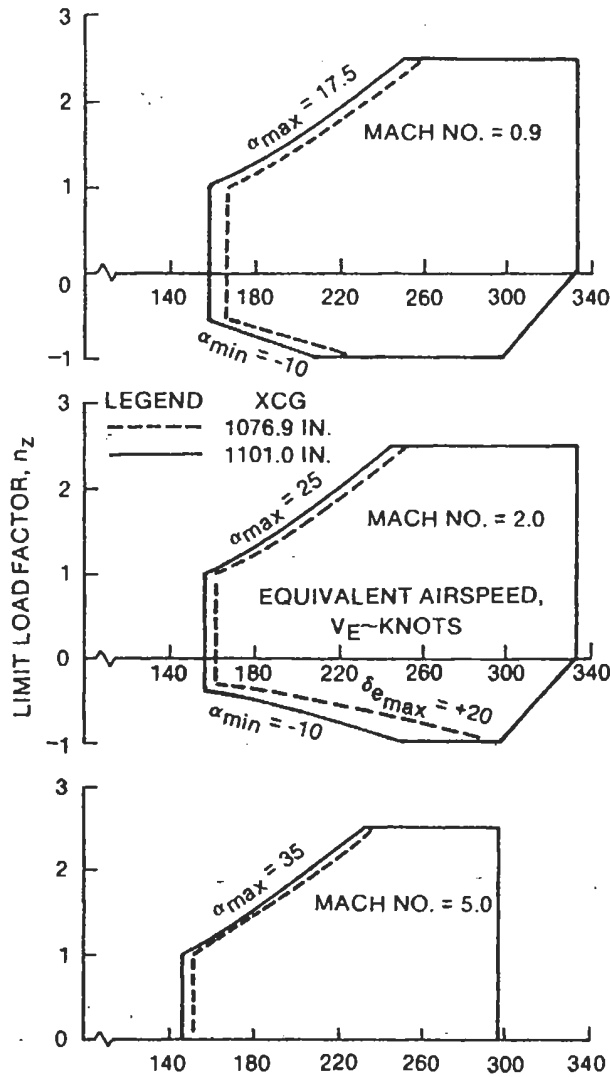


FIGURE 2.- DESCENT V-n CRITERIA.

stress of modern materials led to the unprecedented approach of not specifying a yield factor. It was shown that a yield factor could be misinterpreted during preliminary design and that the preferred approach was to focus on the deformations and on fracture and life requirements.

DEFINITION OF FEASIBLE CREW CABIN WEIGHT

As the crew cabin design evolved, it was noted that the specific design missions compared to the available volume resulted in considerable opportunity for weight growth. Apollo command module volumetric stowage densities were reviewed. The result was that the structural design of the crew cabin was performed using a value of 30 000 pounds even though specific mission requirements could not define weights above 27 891 pounds. This provision proved to be well worth the minor scar weight of approximately 57 pounds and allowed very flexible mission planning with non-weight-limited stowage volumes used for payload and mission integration.

DEFINITION OF DESIGN BUDGET FOR THERMAL STRESS

Results of early studies had not revealed the complexity of defining design-case thermal stresses. With the extreme pressure of "designing out" Orbiter structural weight, redesign to desen-

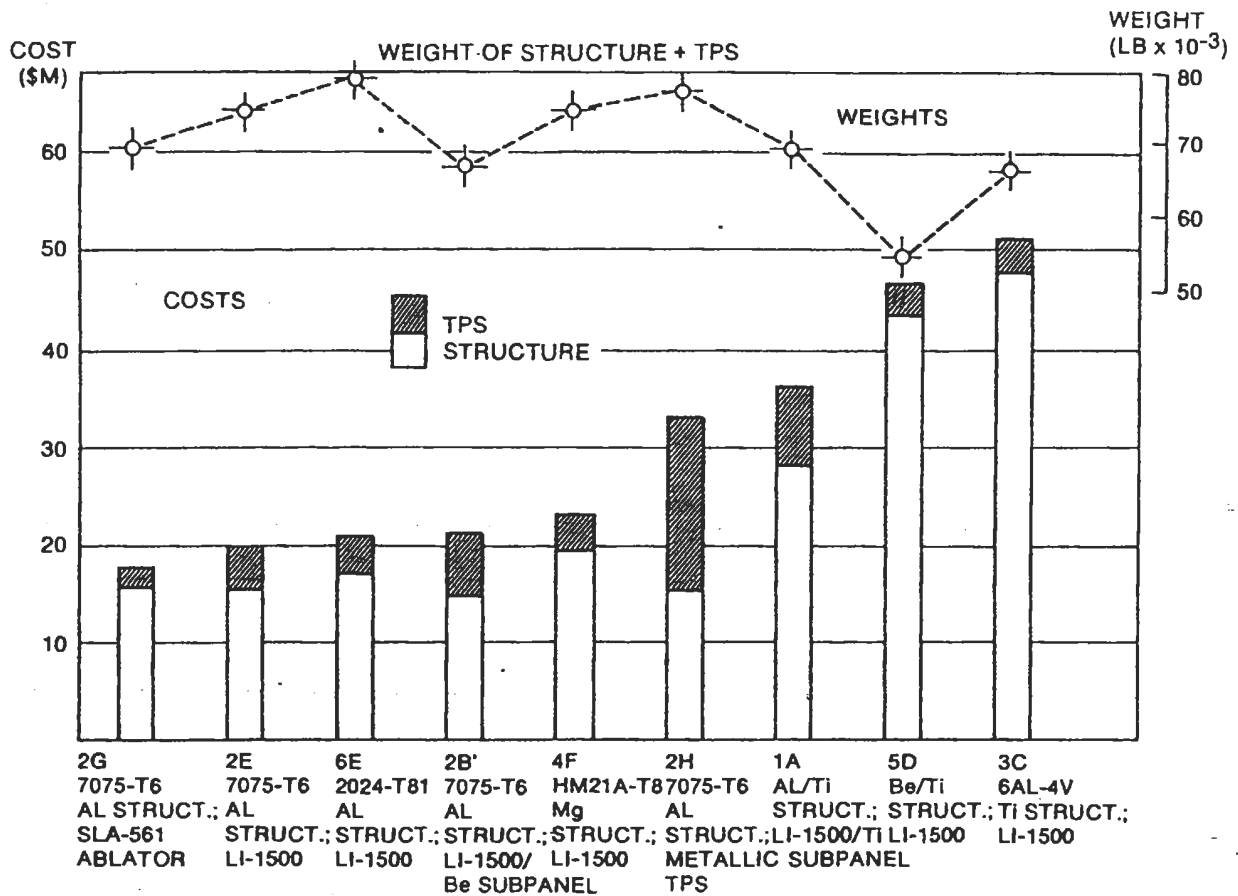


FIGURE 3.- ORBITER STRUCTURE/TPS FIRST UNIT COST COMPARISON.

3.2.2.2.6 ULTIMATE COMBINED LOADS. THE MECHANICAL EXTERNAL, THERMALLY INDUCED, AND INTERNAL PRESSURE LOADS SHOULD BE COMBINED IN A RATIONAL MANNER ACCORDING TO THE EQUATION GIVEN BELOW TO DETERMINE THE DESIGN LOADS. ANY OTHER LOADS INDUCED IN THE STRUCTURE, E.G., DURING MANUFACTURING, SHALL BE COMBINED IN A RATIONAL MANNER. IN NO CASE SHALL THE RATIO OF THE ALLOWABLE LOAD TO THE COMBINED LIMIT LOADS BE LESS THAN 1.40

$$K_1 L_{EXTERNAL} + K_1 L_{THERMAL} + K_2 L_{PRESSURE} \geq 1.40 \Sigma L$$

$K_1 = 1.4$ WHEN THE TERM IS ADDITIVE TO THE ALGEBRAIC SUM, ΣL

$K_2 = 1.5$ FOR TANKAGE WHEN THE TERM IS ADDITIVE TO THE ALGEBRAIC SUM, ΣL

$K_1, K_2 = 1.0$ WHEN THE TERM IS SUBTRACTIVE TO THE ALGEBRAIC SUM, ΣL

FIGURE 4.- COMBINED LOADS CRITERIA.

sitize the minimum weight concepts from thermal stresses was not practical. Also considered was the concept of a tolerance on the thermal gradients. Although seemingly logical, a three-dimensional finite-element model was not suitable for a consistent combination approach. An automated worst case would have inferred different temperatures at the same node point. This conflict implied inconsistent added weight, and the concept was rejected. Results of studies performed showed the sensitivity of the initial conditions achieved on orbit. These data showed that approximately 30 percent of the total stress could be attributed to that "locked in" at entry interface.

Tail Sun, top Sun, side Sun, and nose Sun attitudes and initial conditions from a once-around mission were chosen as design initial conditions. Theoretical thicknesses of TPS tiles were specified for approximately 100 three-dimensional models. The gradients were assessed and several time points were selected for detailed analysis. These data were then hand-faired and extrapolated to the entire finite-element grid. This procedure involved judging the temperatures to about 100 times the resolution of the thermal analysis. This experience demonstrated the technology weakness of determining the design-case transient thermal stress for large three-dimensional structures. The design was frozen to this budget of thermal stress. It remains for operational planners to ensure that the operational envelope stays within the budget.

MATERIAL BASELINE OF THE PRIMARY STRUCTURE

In selecting the materials for the "airframe" of the combined airplane/spacecraft Orbiter, the conventional trade-off studies, considering the costs to produce the first unit and the weight, had to accurately reflect the compatibility requirements imposed on the structure by the external TPS. The very low strength, brittle TPS tiles, which were to be bonded to the skin of the Orbiter, required that the skin not buckle when exposed to 115 percent of maximum expected loads (limit load) during ascent and 100 percent of the maximum expected load during atmospheric entry. From a weight standpoint, this skin-buckling requirement made aluminum and titanium equivalent since the ratio of the compression modulus to density is approximately the same for both materials ($E_c/\rho = 10 \times 10^7$ inches). The structure of the Orbiter, sized by flight loads where the material could be worked to high stresses, favored the high-strength materials such as titanium. The third general category considered in the structural material selection was the heat capacitance and the strength at elevated temperatures. When considering the requirements of buckling for TPS compatibility, strength, heat capacitance, conductance, and a few other factors, the combined weight of the structure and the TPS was approximately 15 percent less for titanium than for aluminum as the primary structure; however, the cost for the titanium structure was approximately 300 percent greater than that for aluminum. The much higher production cost and the higher-production development risk for titanium and the system performance studies resulted in selection of aluminum for the basic primary structure.

There were two major areas of the primary structure for which aluminum was not selected - the payload bay doors and the main engine thrust structure. The payload bay doors were designed for maximum reliability in opening and closing in space. To help achieve this reliability, the doors were designed so they could be "zipped" up during closure. This zipping capability dictated that the doors be flexible and have large clearances between door segments and surrounding structure. To help meet these requirements, the Orbiter primary structure was designed so that the doors were not affected by fuselage bending (only by pressure and torsion loads). The large, flexible payload bay doors, which had very few TPS tiles, were then optimized for weight, thermal distortion, and cost. Graphite epoxy was selected as the material.

The thrust structure reacts 1.5 million pounds of Space Shuttle Main Engine thrust load and distributes the vertical stabilizer loads and external tank aft attachment loads into the fuselage. The material selected for this highly loaded structure was titanium. For long truss members that were compression/buckling critical, the weight was reduced by overlaying longitudinal strips of boron epoxy. The thermal expansion compatibility between the titanium and the boron epoxy permitted this composite system. The system reliability was increased by sizing the titanium structure so that limit load could be accommodated on any structural member even if a segment of the boron epoxy became debonded.

The general arrangement of the Orbiter primary structure and the materials used is shown in figure 5.

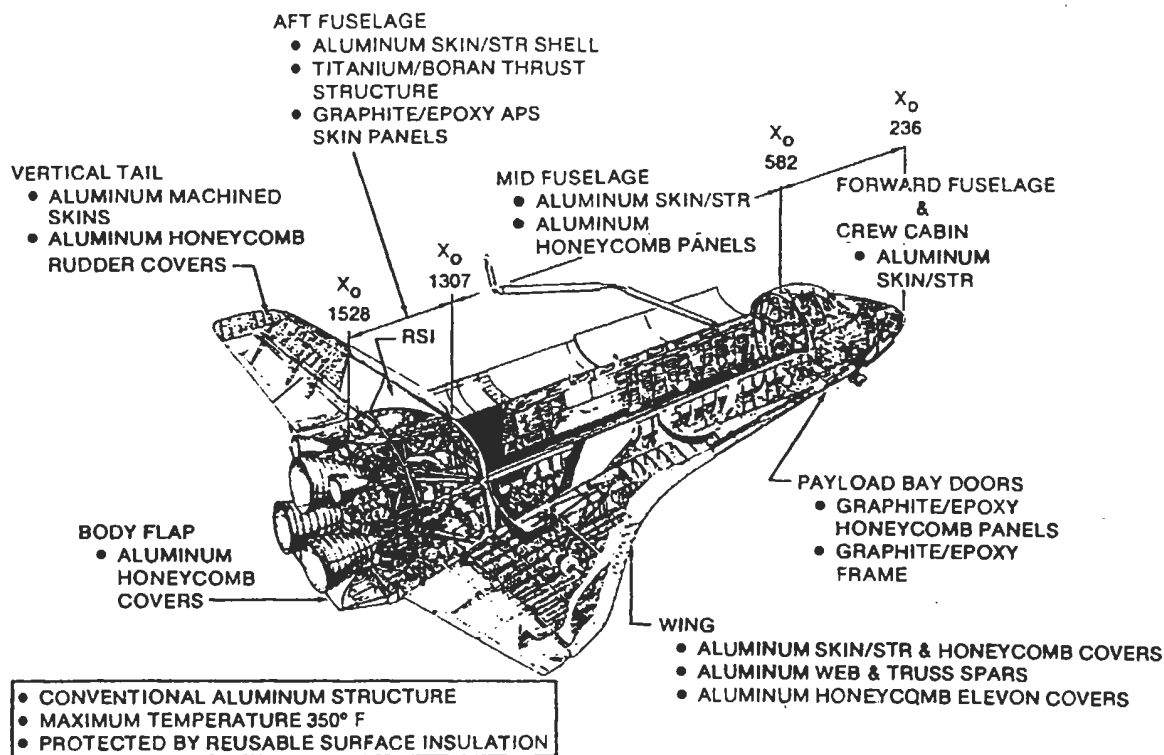


FIGURE 5.- ORBITER STRUCTURE.

DESIGN INNOVATIONS

ORBITER WINDOW SYSTEM

The Orbiter window system shown in figure 6 represents a significant and unheralded achievement in glass structural technology. The Corning Glass Company agreed to produce the windows as a national service. Fracture mechanics criteria were used to design the untempered 1200 pounds of fused silica glazings. Sustained load flaw-growth requirements set the proof test requirement at 8600 psi to screen a flaw of 0.0018 inch. Initially, serious technology reservations were expressed because of the "mass effect" of the large boules weighing approximately 2000 pounds each. Development of the processes and coating verification required 2 years of challenging engineering. Because of changes in aerodynamic data and lessons learned from the flight data, critical inspection and polishing of the operational windows are necessary to assure the life of the window.

MAJOR STRUCTURAL CONCEPT STUDIES

Several major studies were performed to define the minimum-weight structural concepts. These studies had been identified as a result of the phase B options that had been proposed. Trade-off studies were performed by the contractor and by in-house NASA engineering groups. For example, it was shown that a space-frame concept for the thrust structure would save approximately 1730 pounds as compared with a competing plate girder concept. Similarly, it was shown that integration of the aft carrythrough spar with the 1307 bulkhead would save approximately 450 pounds compared to a floating carrythrough and would require a corrugated bulkhead web to limit the stress concentrations. A single-point drag attachment between the Orbiter and the external tank was studied since it achieved a statically determinate interface. It was found to be heavier and more difficult to integrate with the natural load paths from the thrust structure. Results of other studies confirmed the weight effectiveness of the separate crew module as compared to an integrated fuselage and crew module. The requirement for the payload bay doors to react body torsion while precluding vehicle bending was confirmed. Weight and cost studies led to the selection of the composite material systems used in the payload bay doors, the Orbital Maneuvering System (OMS) pod external shell, frame and rib tubes, and the thrust structure. These concept selections resulted from penetrating engineering trade-off

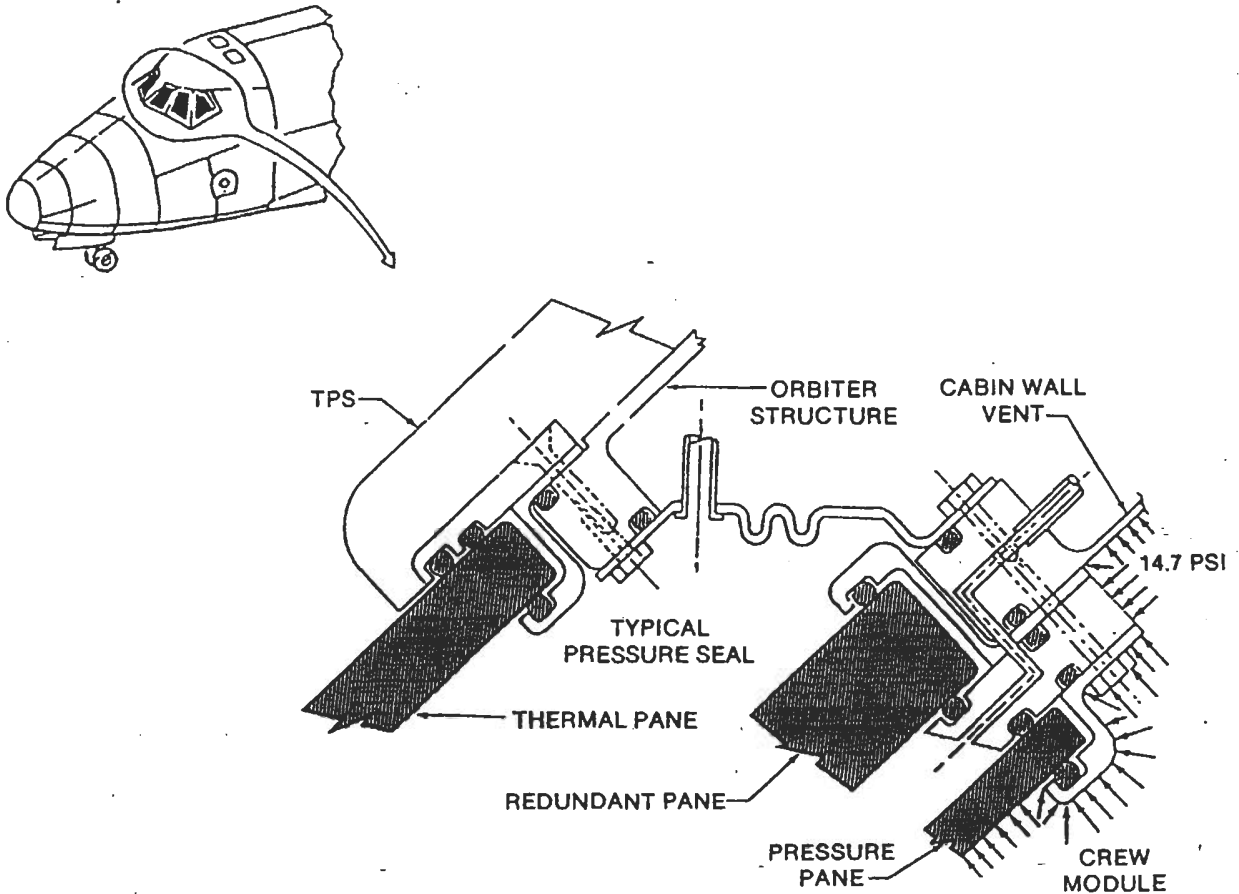


FIGURE 6.- TYPICAL ORBITER WINDSHIELD CONFIGURATION.

studies. The weight "scraping" which followed was the next significant milestone as the loads matured. Significant weight reduction was achieved on Orbiter vehicles OV-103 and OV-104. Attention to detail, with the successful development period confirming the stress distributions, led to structural weight reduction of approximately 1000 pounds.

ORBITER VENTING SYSTEM

Design of the Orbiter venting system involved challenges not previously encountered. Previous spacecraft experience dictated a design which would have connected the entire volume and vented it through base vent areas. Because of the requirements of contamination and cleanliness of the payload bay and the potential hazards of hydrogen concentrations, a multicompartment scheme of self-vented, passively vented, and controlled venting compartments evolved. This arrangement required the vent areas to be distributed around the vehicle and required that they be placed and sized so as to accommodate significant variations in pressure coefficient at the vent. The definition of design differential pressures across the internal bulkheads required extensive analysis to identify the critical combination of venting parameters. The resulting design and performance are discussed in reference 3. Overall, the system performed as designed, is tolerant of failures and variations, and has adapted well to changes in design data and initial requirements. Although it has been largely unnoticed, it represents an outstanding example of a thorough, innovative design achievement.

ACOUSTIC FATIGUE

Fatigue of the Orbiter structure was considered by many engineers and managers to be inconsequential since the design life was for 100 missions and only a few hours of flight in the atmosphere. By comparison to the classical airplane ground-air-ground loading cycles for a 20 000-hour airframe, the Orbiter did have reduced requirements; but, when considering the high acoustic levels (fig. 7), high cycle, low-stress fatigue could not be ignored.

The challenge was in certifying this large, complex, multimaterial, multiconfiguration, multi-manufactured source structure for multicombinations of loads for the high acoustic levels. The obvious solution was to certify by testing as is done for airplanes - apply a spectrum of flight loads to a dedicated test airframe for as many as four times the number of expected load cycles. The Concorde airplane, which was designed for elevated temperatures (200° C), was fatigue-tested in the classical manner while simultaneously being convectively heated. Such a test program for the Orbiter would have required a dedicated structural test article (costing approximately \$100 million), a nonexistent acoustic facility, and a method of imposing rapidly changing temperatures on the structure. The cost, schedule, and technical capability were out of reach.

The acoustic fatigue certification program established was truly innovative. The concept was to test representative structure of various forms, materials, and construction in a representative acoustic environment until the test article failed. (The regions to be tested are shown in fig. 8.) This procedure would result in establishing an acoustic fatigue damage allowable for each type of material and construction. The allowable damage would then be reduced analytically to account for the damage induced by the flight loads and elevated temperature. The test articles were sized so that only one-third of the specimen was the test region - the remaining two-thirds of the specimen was compromised because of the boundary conditions. Several of the test articles are shown in figures 9 to 11.

This approach was modified for some test articles like the payload bay doors, which have great fatigue durability because of the graphite epoxy construction. The compromise consisted of not testing to failure but instead of analytically modeling the structure and correlating with strain measurements from the test. The test articles were then used as flight hardware.

Additional acoustic fatigue certification objectives relative to secondary structure, system installations, and TPS were also accomplished on many of the test articles. This additional hardware was "piggybacked" on the test articles.

STRUCTURAL TEST APPROACH

Detailed planning of the structural test conditions was well underway for a classical static test program followed by fatigue investigations when a new opportunity emerged. The Orbiter structure had evolved under such weight-saving pressure that virtually all the primary structure had a significant thermal stress component. Attempts to factor mechanical loading into equivalent thermal loadings resulted in inconsistent stress distributions which were not meaningful simulations. Thus, it became clear that the classical demonstration of design ultimate strength was not feasible without a thermal simulation. Such a simulation was not practical and probably not possible for the transient cases of interest. The objective of the test then had to emphasize the correlation of the stress analysis with the measured strain data. It also became evident that a test to ultimate load (1.4 times limit) would not achieve design ultimate stresses but would probably result in deformations and strains to render the airframe unusable for flight. At the same time, it became clear

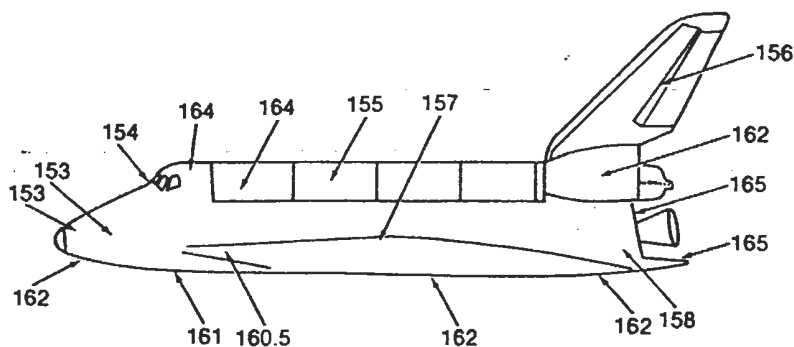


FIGURE 7.- ORBITER AERODYNAMIC-ACOUSTIC NOISE LEVELS.

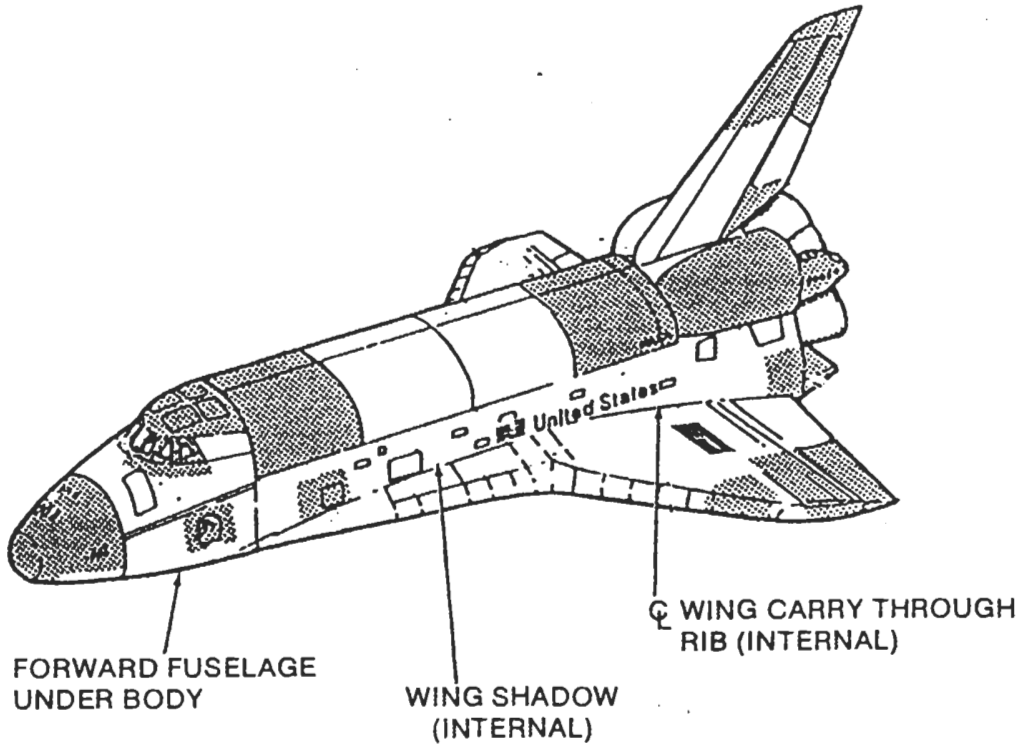


FIGURE 8.- ORBITER ACOUSTIC FATIGUE TEST ARTICLES.

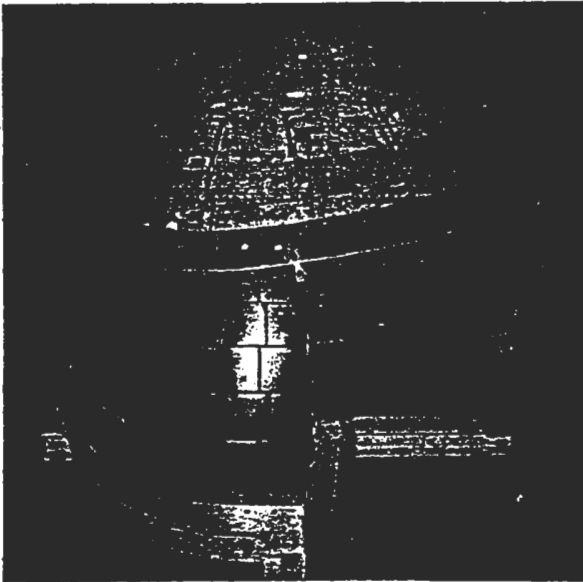


FIGURE 9.- NOSE CAP TEST ARTICLE.

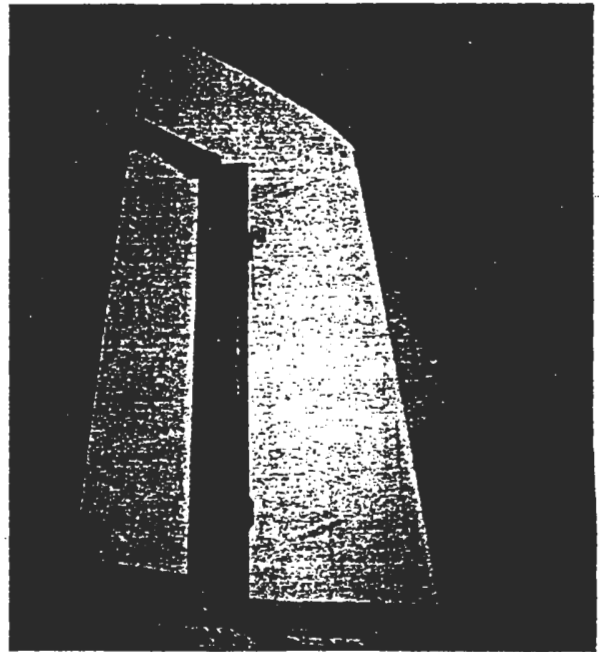


FIGURE 10.- VERTICAL STABILIZER TEST ARTICLE.

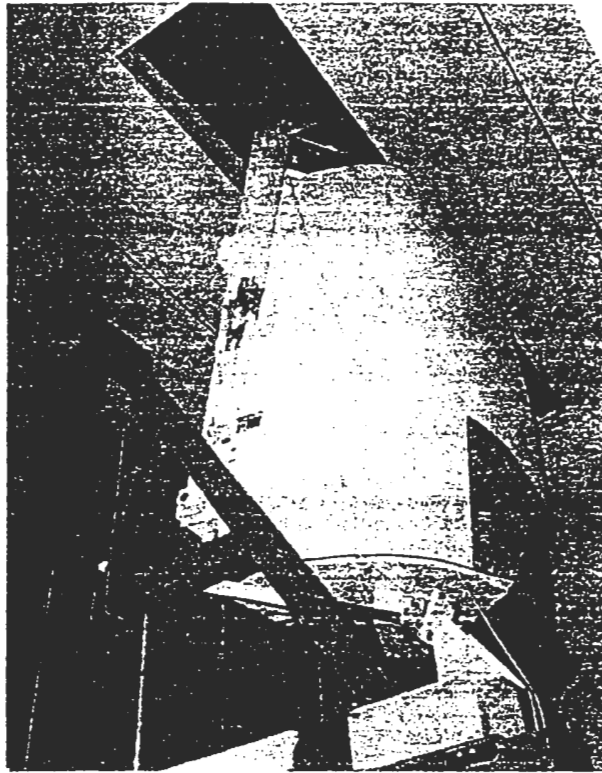


FIGURE 11.- OMS POD TEST ARTICLE.

that the structural test article (OV-099) had to be proved acceptable with the 5.4 loads data base even though it had been designed for the 5.1 data base. From this situation emerged a precedent-setting proposal: (1) test to 1.2 times the 5.4 loads, (2) require detailed stress analysis of each test condition, (3) include unit load test evaluations where required, (4) perform a combined thermal and mechanical test of the forward fuselage, (5) instrument sufficiently to compare with analysis and identify peak critical stresses, (6) perform additional component test investigations for ultimate strength and fatigue of identified critical interface hardware, (7) perform and document a critical post-test inspection, and finally, (8) refurbish the airframe structure for use as a flight vehicle. Daring as this proposal initially sounded, the technical detail and innovative imagination was sustained through complete management review as well as special technical review teams.

The test was a remarkable accomplishment. Thirty-nine test conditions simulating 32 critical design conditions were used. The engine thrust loads required innovative design concepts to accurately position the engine thrust vector. A typical test involved control of 256 jack loads which were distributed over 836 load application points. Because of lack of agreement with analysis, a complete influence test program was performed on the thrust structure; this program resulted in a high-fidelity empirical math model. Partial influence coefficient tests were performed on the payload reactions. The thermal test was accomplished on the forward fuselage by heating the external skin with external resistive blankets controlled in six zones. Cooling was provided by gaseous purge of the internal frame caps. When the test temperatures and gradients had been achieved, mechanical loading was applied to simulate the nose gear impact inertial loads.

The results of the testing were quite impressive and may be summarized as follows. The stress distributions in the critical test regions compared within 10 percent of the analysis with few exceptions. Exceptions were noted in the external tank forward and aft fittings and support structure, in the distribution of reaction between the front and aft spars of the vertical stabilizer, and in the y-load distribution of the main gear reaction. Supplementary instrumentation and component test articles were defined to fully instrument and certify these regions. The thermal test measured stresses compared well (10 percent) with those measured on the frames and internal structure. The skin stresses in the circumferential directions were considerably less (30 percent) than predicted analytically and in good agreement longitudinally. Hardware modifications resulting from the test data were identified. The forward reaction control system tank support structure, which failed in

test because of inadequate lateral stiffness, was redesigned, retested, and implemented. The empirical load distribution between front and aft spars of the vertical stabilizer resulted in drag angle modifications, which were implemented after component retest. The lateral load sharing of the main gear reaction was used to define the shimming and rigging specifications. The deflections measured around the nose landing gear door were used to define the rigging requirements.

The proof test concept used on the Challenger vehicle continues to achieve significant cost savings as it has been applied to several Space Transportation System (STS) payloads. This concept is a recognized verification concept for payloads as specified in JSC-14046. It is emphasized that use of this approach requires detailed engineering with attention to the structural details, sufficient instrumentation, and rigorous post-test inspection.

LESSONS LEARNED

THERMAL STRESSES

Mathematical surveys of peak transient thermal stresses were not technically feasible using the operational tools of the early 1970's. Technology and computational capacity are now available to perform integrated thermal/thermal stress surveys analogous to those used to identify the critical aerodynamic loading conditions. These analysis tools should be exercised and tested against flight data. Such studies would determine the criteria, extent, and accuracy of the methods before their use on the next major program.

GRAPHITE EPOXY MOISTURE

Graphite epoxy structure will absorb moisture by diffusion and will degrade in strength properties at temperatures above 250° F. It should also be noted that the honeycomb configurations can absorb enough moisture to be susceptible to failure by vaporization of the moisture and rupture of the honeycomb panel. Flight data appear to indicate that the thermal conductivity of such panels is complex. During the STS-1 mission, vaporization at a temperature exceeding 250° F caused the failure of a portion of the OMS fairing.

WINDOW STRUCTURE

Window design stress and sustained load flaw-growth parameters are sensitive to detailed knowledge of the loading environment, and the windows are susceptible to damage in flight as well as on the ground. Future technology thrusts are warranted to develop high-temperature glazings which are effectively tempered, perhaps by preloading concepts rather than conventional techniques. Annealed glazings should be used with comfortable conservative margins.

STRUCTURAL INSPECTION

Detailed routine structural inspection of large spacecraft will continue to be laborious and costly because of the required insulation systems. Technology advances are required to identify structural damage without requiring removal of the thermal systems. An innovative technique could be effectively tested and implemented during the STS operational era.

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