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SPACE SHUTTLE ORBITER
LEADING-EDGE FLIGHT PERFORMANCE COMPARED TO DESIGN GOALS

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SUMMARY

Thermo-structural performance of the Space Shuttle orbiter Columbia's leading-edge structural subsystem for the first five (5) flights is compared with the design goals. Lessons learned from these initial flights of the first reusable manned spacecraft are discussed in order to assess design maturity, deficiencies, and modifications required to rectify the design deficiencies. Flight data and post-flight inspections support the conclusion that the leading-edge structural subsystem hardware performance has been outstanding for the initial five (5) flights.

INTRODUCTION

Conception of a new era in man's advantageous utilization and exploitation of space was realized recently with the successful completion of the four development test flights and the initial commercial mission of the Space Shuttle orbiter, Columbia. Unique design and construction of the orbiter to achieve reusability, a feature previously impractical in space vehicles, were attainable with the progressive development of high-technology materials used in the Thermal Protection System (TPS). Multi-mission capability is the key in achieving cost effective access to space for routine manned operations. Assessment of this capability is now possible with the accrued flight data coupled with the information gathered from post-flight inspections conducted after each flight.

Essential to the total system of thermal protection of the orbiter is the leading-edge structural subsystem (LESS), generally defined as those areas of the wing leading edge and the forward fuselage that exceed maximum temperatures of 2300°F during re-entry. Reinforced Carbon-Carbon (RCC) is one of the new generation materials that is indispensable in providing multi-mission capability in this punishing, high-temperature environment while concurrently maintaining the integrity of the aerodynamic surfaces. RCC is a hard carbon structural material possessing reasonable strength throughout the operational temperature range predicted for the orbiter. Thermal shock and thermoelastic stress effects are minimized with the low coefficient of thermal expansion. Oxidation protection, fundamental to the reusability feature of RCC, is provided to the carbon substrate by converting the outer surface to Silicon Carbide (SiC) in a diffusion coating process. Further enhancement of the oxidation protection is provided by post-coating treatments of vacuum impregnation of the laminate with Tetraethyl Orthosilicate (TEOS).

Success of the initial five flights of Columbia implies that the flight environments were appropriately anticipated and the system response accurately predicted. Although verification of the total system capacity in terms of reusability remains unconfirmed, certain parameters can be evaluated from the acquired flight data to provide forecasts necessary for operational viability for the life of the orbiter. Lessons learned during the early stages of this unique, reusable space vehicle can be used to identify not only areas of the orbiter that need attention to achieve maturity but also technology deficiencies on which to concentrate research and development for future space systems.

LESS DESIGN

The orbiter LESS basically consists of the RCC nose cap and seals, the wing leading-edge RCC panels and seals, the associated metal attachments to the supporting structure, the internal insulation systems, and the interface Reusable Surface Insulation (RSI) tiles. Depicted pictorially in Figure 1, the RCC nose cap and wing leading-edge constitutes approximately 420 square feet of external surface area. Additionally, although not included in the original design, pre-flight modification of the region surrounding the forward, external tank attachment was made to include a RCC cover plate, appropriately identified as the arrowhead illustrated in Figure 1.

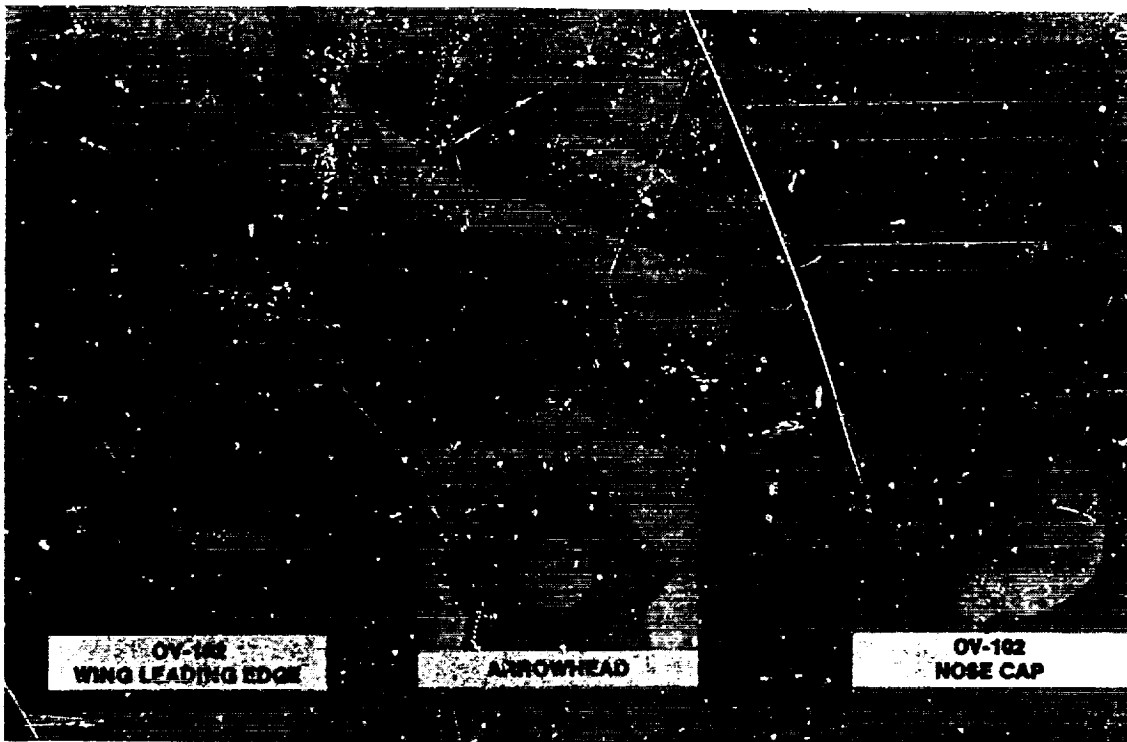


Figure 1.- Leading-edge structural subsystem.

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Basic design goals and purpose for the LESS are to provide thermo-structural capabilities for the regions of the orbiter that exceed 2300°F. Operational requirements include the retention of the aerodynamic shape of the outer moldlines, the control of the aluminum structure maximum temperature to less than 350°F, and the capability to sustain 100 missions with minimal refurbishment. Interface control between the RCC and the RSI tiles was a significant parameter in the design not only to retain the aerodynamic surface for flight quality but also to preclude damaging the more vulnerable tiles. Serviceability was another issue that dictated the field-break design configuration for access to the attachments and easy removal of the RCC components.

Final design configuration of the RCC nose cap assembly is illustrated in Figure 2, consisting of the dome, five (5) gap seals, and three (3) expansion seals. Functional requirements of the seals are to allow thermal expansion and deflections while simultaneously preventing hot gas influx into the cavity and precluding deflections of the RCC that penetrate into the interface RSI tile envelope. Illustrated pictorially by the representative panel-seal set in Figure 3, the wing leading edge consists of twenty-two (22) similar assemblies on each wing. Gap seals are provided between the panels to serve the same function previously described for the nose cap seals. Optimization of the size of the panels included as significant parameters: structural integrity, producibility in terms of tooling and facility requirements, and weight.

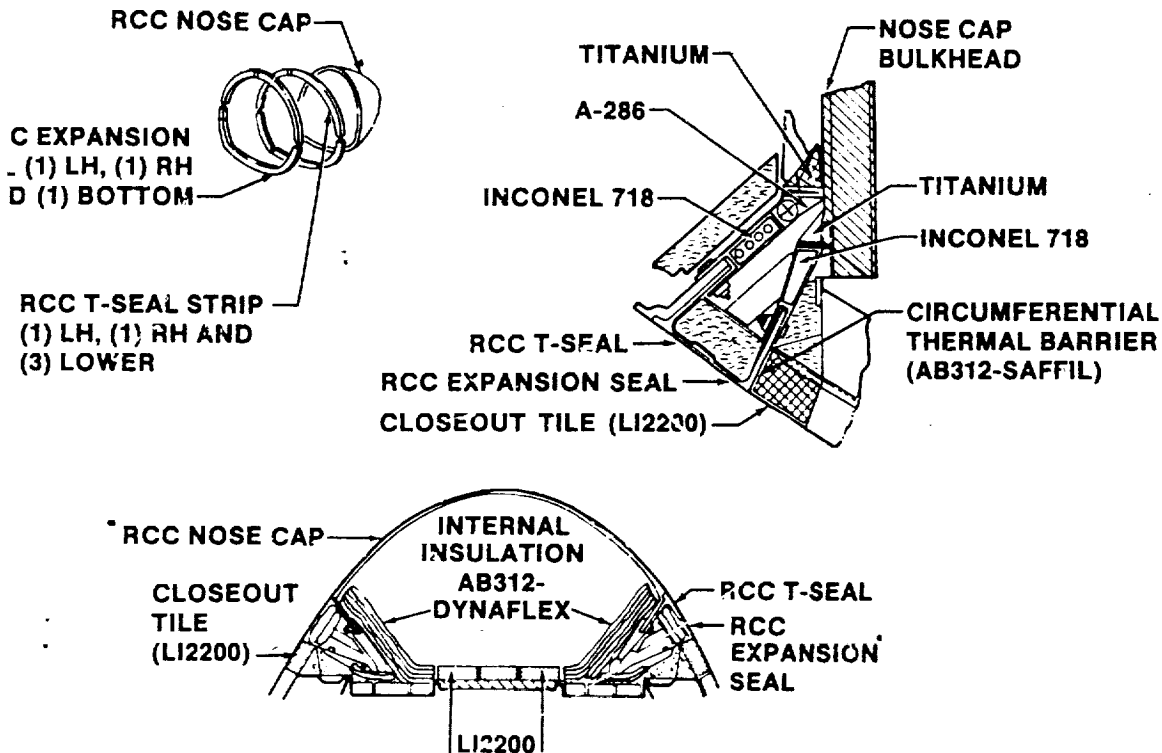


Figure 2.- Nose cap system.

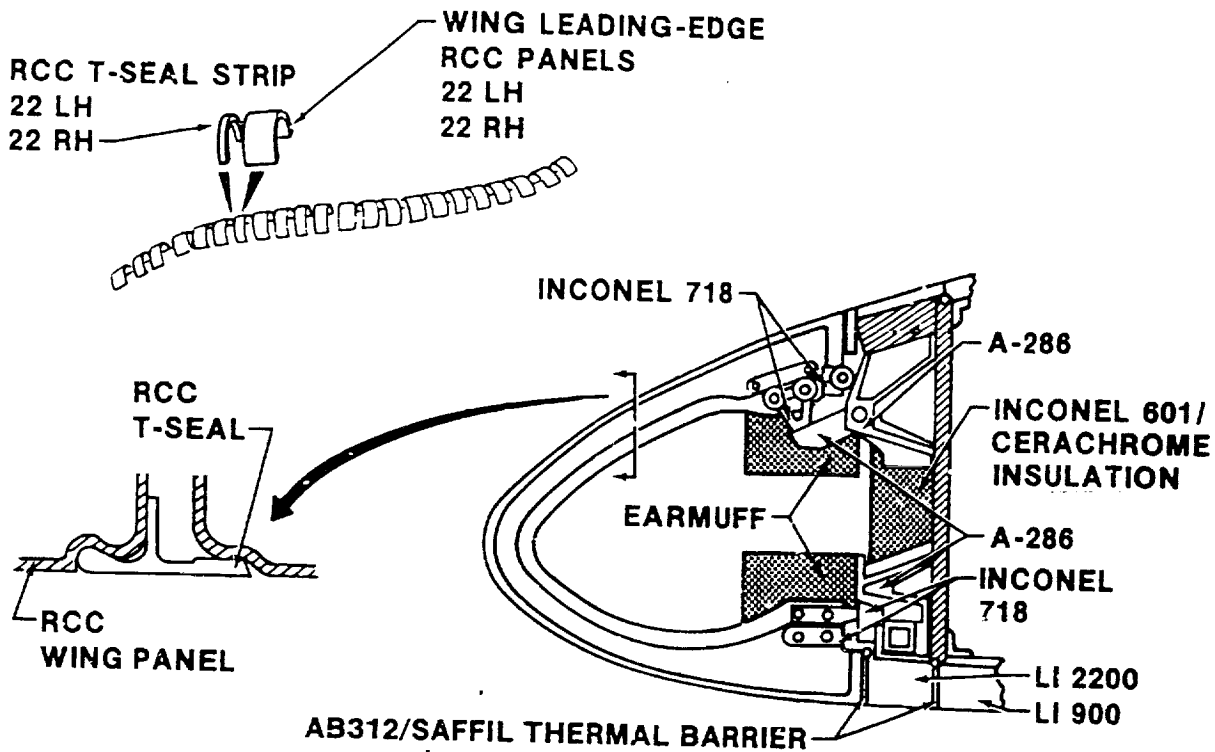


Figure 3.- Wing leading-edge system.

Elevated temperature is the primary factor in the design of the attach fittings as well as the internal insulation system used in the protection of those attachments. Heat resistant metals such as Inconel 718 and A-286 steel are utilized to interface between the RCC and the aluminum support structure. Protection is provided these metal components with various insulation packages composed of Dynaflex, AB-312 ceramic cloth, saffil, or RSI tiles. Dynaflex, contained in formed and welded Inconel 601 foil, is the primary insulation system used in the wing leading edge as illustrated in Figure 3. Blankets of Dynaflex and saffil wrapped with AB-312 cloth are used in the nose cap cavity along with RSI tiles on the forward face of the access door in the support bulkhead as indicated in Figure 2.

Thermophysical properties of the RCC material and the hollow shell design promote internal cross radiation from the hot stagnation region to the inherently cooler regions. This characteristic reduces the stagnation region temperatures and the critical lower lug temperatures and minimizes the thermal gradients in the shell. Paradoxically, the insulation used in the cavity to preclude exceeding the maximum temperature limits established for the metal components also retards the cooling rate of the lugs, contributing to the undesirable oxidation rate.

Oxidation rate is the single most important variable parameter in the determination of mission life of RCC parts. Oxidation of the carbon substrate occurs as a result of oxygen penetrating the protective coating through microscopic porosity or fissures inherent in the coating system. Resultant strength degradation^{1,2} caused by the substrate mass loss restricts the mission life capacity through the inability of the RCC to sustain the predicted loads. Oxidation rate is a function of temperature, pressure, time, and the type of environment, either radiant or convective heating.³ Radiant and convective mass loss correlation curves are presented in Figure 4, applicable to the flanges and the outer shell regions respectively.

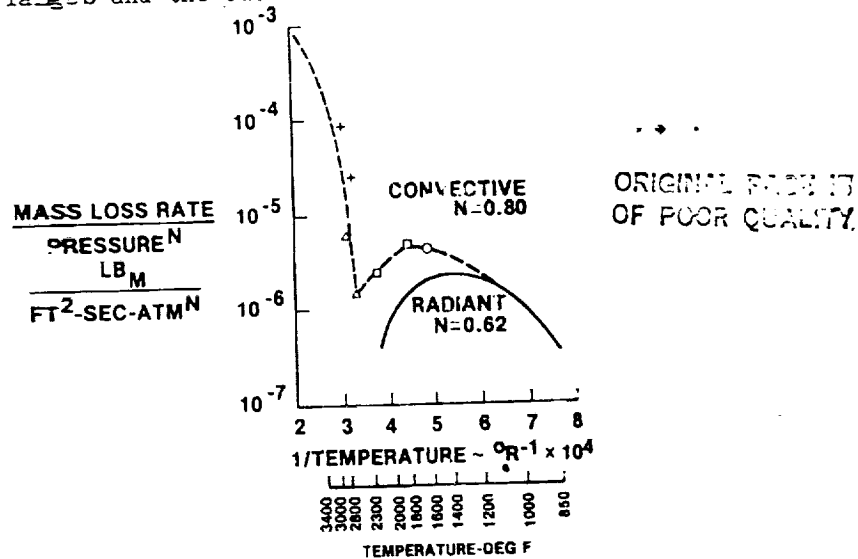


Figure 4.- RCC mass loss correlation.

Thermal analyses were performed on the nose cap and representative wing leading-edge panels to correctly design the temperature sensitive elements as well as to determine the temperature histories at selected locations. Comprehensive two- and three-dimensional thermal math models, developed for the design analysis, were verified in the development and qualification tests and used for flight certification.^{4,5}

Structural analyses were performed on the nose cap and wing leading edge with the basic objective to determine the resultant stresses, deflections, and margins of safety for the applied environments. The complex nature of the design coupled with the deterioration of the mechanical properties of RCC with each repeated exposure to oxidation created unusual analytical problems. Detailed finite element structural models were constructed for the nose cap and representative wing leading-edge panels to insure adequate resolution of the issues. Verification of the analytical methodology was achieved in the qualification test program which led to flight certification.^{6,7} Additional complexity was introduced with the inherent shape of the parts, the variable stiffness of the support structure, and the interaction of the adjacent parts. Critical stresses had to be determined for each part, dependent upon these parameters and sensitive to the distribution of the applied airloads. Typical spanwise variation in the airloads along the wing span can be observed in Figure 5.

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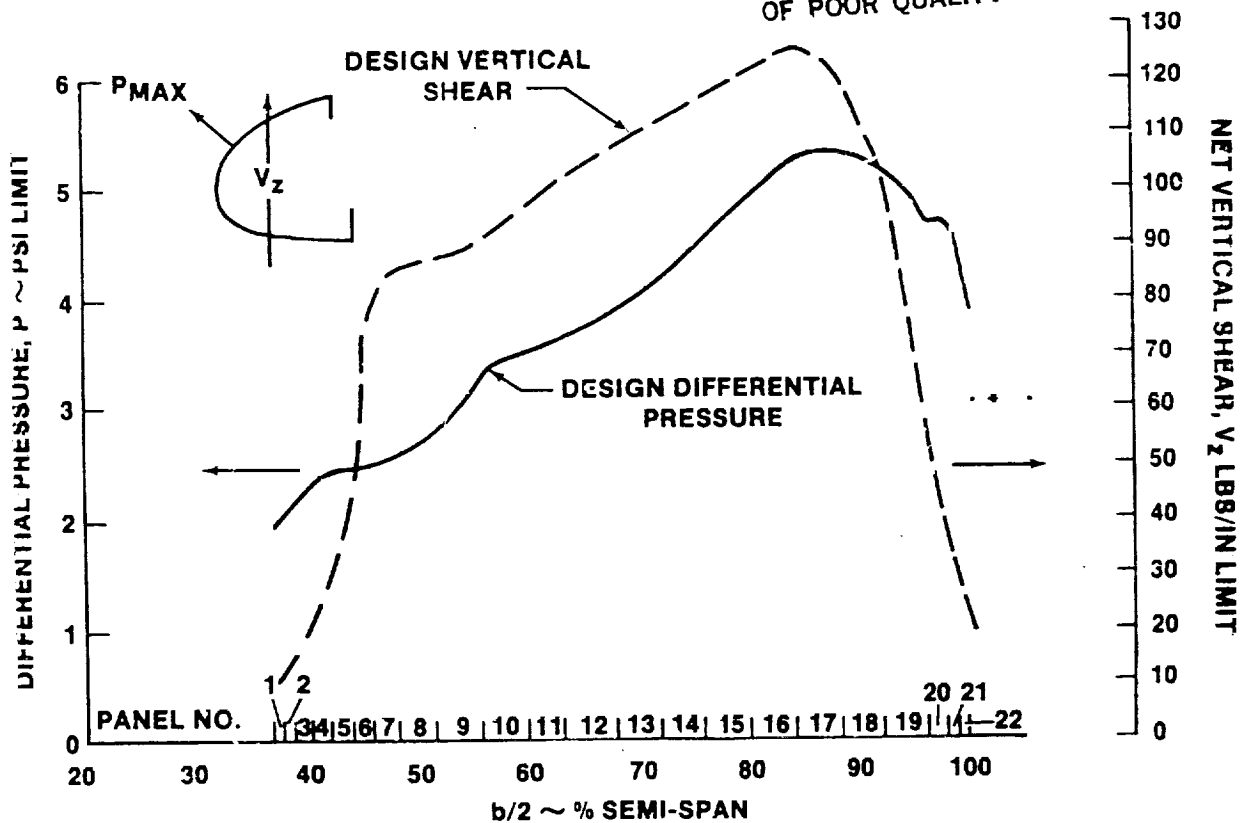


Figure 5.- LESS design airloads.

Thermoelastic stress analyses were also performed on the LESS components at several time cuts in the re-entry trajectory. Thermal-induced stresses in the wing leading edge are minimal with the attachment system providing unconstrained spanwise growth capability, and the thermal gradients are insignificant. However, the nose cap with relatively rigid constraints and high thermal gradients exhibited critical thermoelastic stresses during the initial flights. Coefficient of thermal expansion differences between the RCC and the metal fittings dictated slotted joint designs to eliminate induced stresses. Integrated thermal expansion and combined environment-induced deflections also had to be accurately predicted in order to determine the gap requirements between adjacent parts as well as to avoid RCC to tile interference at the interface joints.

Certification of the LESS for flight was accomplished by analyses verified with development and qualification tests conducted on full-scale hardware. Critical launch and entry conditions were simulated in these tests, cyclically exposing the parts to acoustic, thermal, and airload environments. Comparisons of the predicted versus measured response to the airloads 6,7 and thermal⁴ stimuli resulted in approval of the certification process. Structural assessment of the flight performance must integrate the results of ground tests to substantiate any observation or conclusion from the flight data.

Successful completion of the initial flights of Columbia have provided sufficient data from radiometers, thermocouples, and pressure transducers to appraise the thermal performance. STS flight parameters, especially angle of attack, allowed relatively lower total heat load and heat rate on the LESS than that predicted for the design trajectory. STS and design trajectory differences can be assessed by comparing the heat rate and heat load to a one-foot sphere. Peak heat rate varied from 80 percent of design for STS-1 to 96 percent for STS-4; whereas, heat load varied from 64 percent of design for STS-5 to 92 percent of design heat load for STS-2. Radiometer data presented in Figures 6 and 7 for the nose cap and wing leading edge, respectively, indicates good agreement between the predicted and measured temperatures for the RCC shell inner surface. Measured STS flight temperatures for the panel 9 attachment clevis fitting were lower than both the STS and design predictions as indicated in Figure 8. Evaluation of the STS flight data indicates no degradation in the thermal performance of the LESS components, particularly the insulation systems.

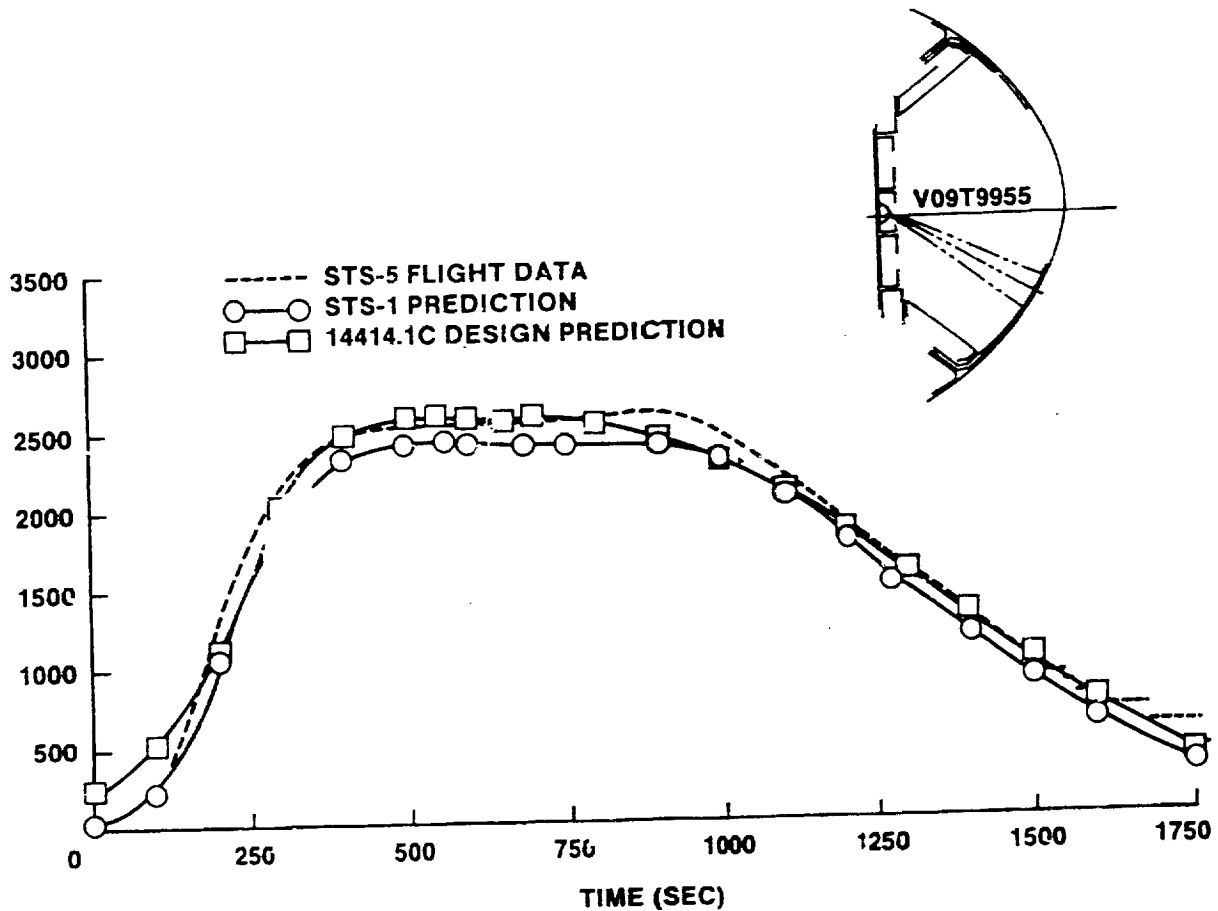


Figure 6.- Nose cap RCC inner moldline (IML) temperature.

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I STS FLIGHT DATA
● STS-2 PREDICTIONS

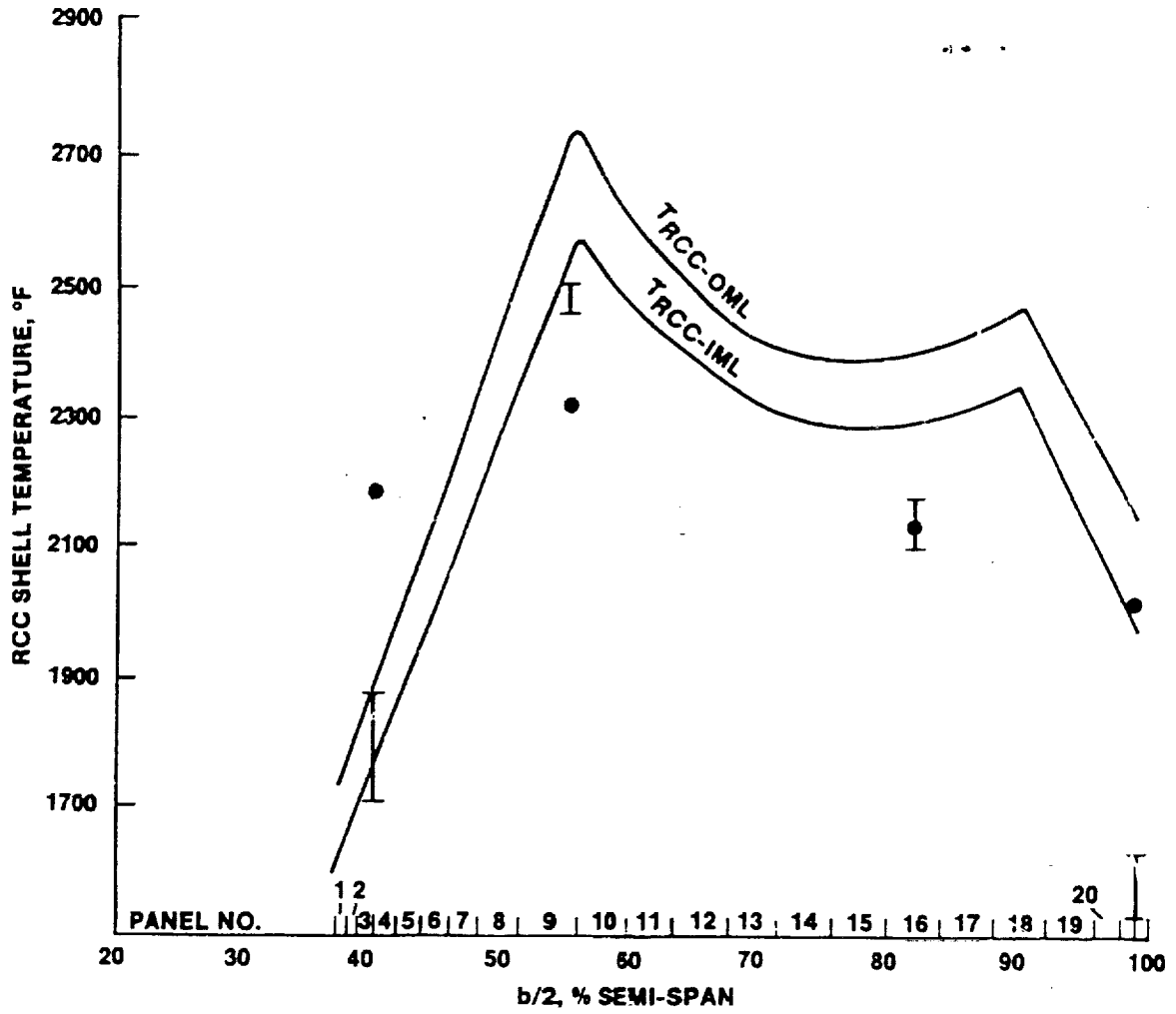
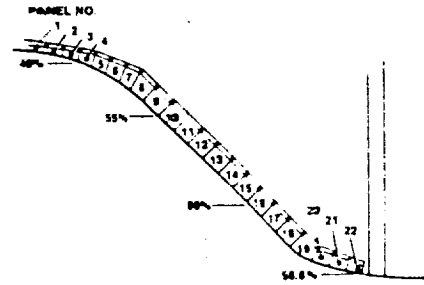


Figure 7.- Wing leading-edge inner moldline (IML) temperatures.

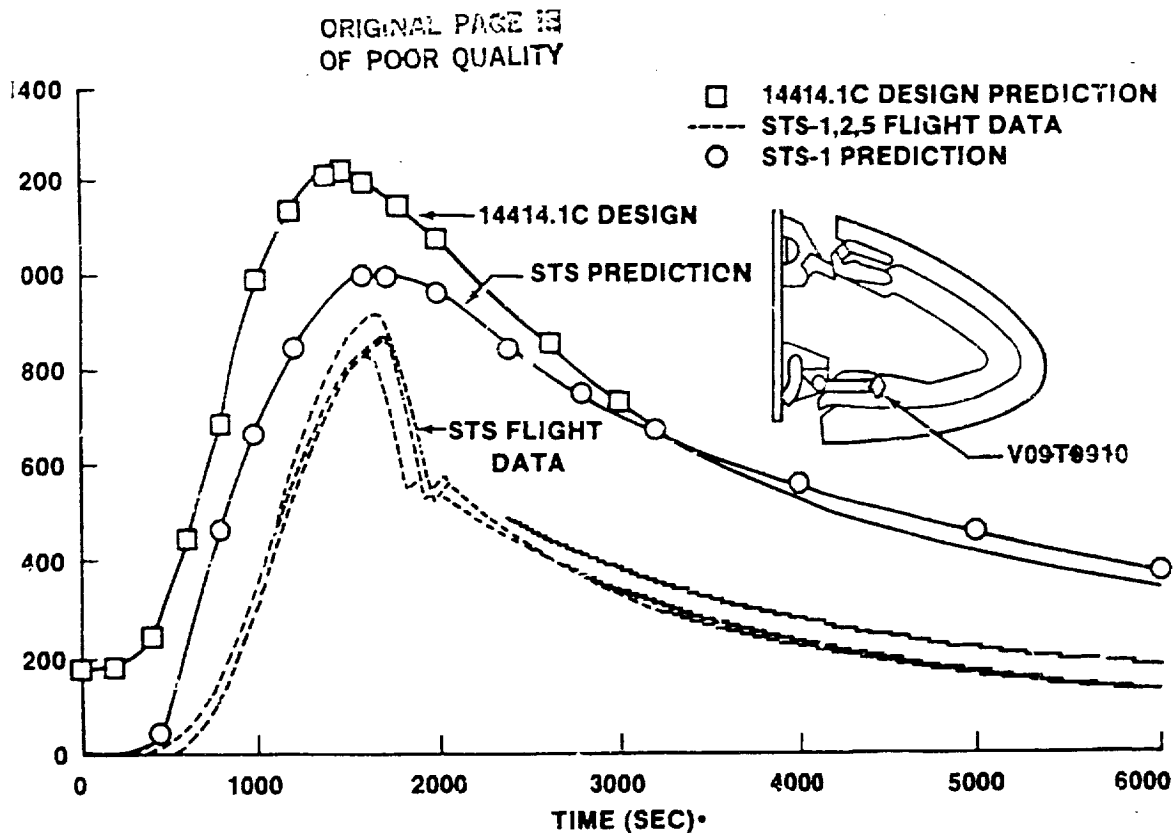


Figure 8.- Wing leading-edge panel-9 lower attach clevis temperature.

Structural performance evaluation requires a more subjective appraisal since the elevated temperature environment precludes the direct acquisition of flight airload data in the critical regions. Strain measurements were limited to special, instrumented attach fittings on the wing leading-edge panel 13, calibrated to determine the magnitude and direction of the flight airloads during ascent. Effective loads are combined external aerodynamic pressures and internal cavity pressures. Load vectors, parallel to the wing front spar, are developed by integrating the differential pressure over the surface area of a panel. Peak ascent loads occur in the maximum dynamic pressure regime coincident with transonic speeds between 1.0 and 1.5 Mach number. Ascent load vectors, developed from flight strain measurements on panel 13, average about eleven (11) percent higher than the anticipated loads as indicated in Figure 9. This is considered to be excellent correlation and indicative that realistic airloads were used in the LESS structural analyses.

Vibroacoustic environments that were used for the LESS design analysis and flight qualification procedures were determined to be conservative from the flight data. Wing leading-edge flight acoustic data present in Figure 10 is comparatively less than the design environment. Impact of this difference is considered negligible at this time but indicative that the critical margins of safety for the RCC lugs are conservative from the vibroacoustic effects perspective.

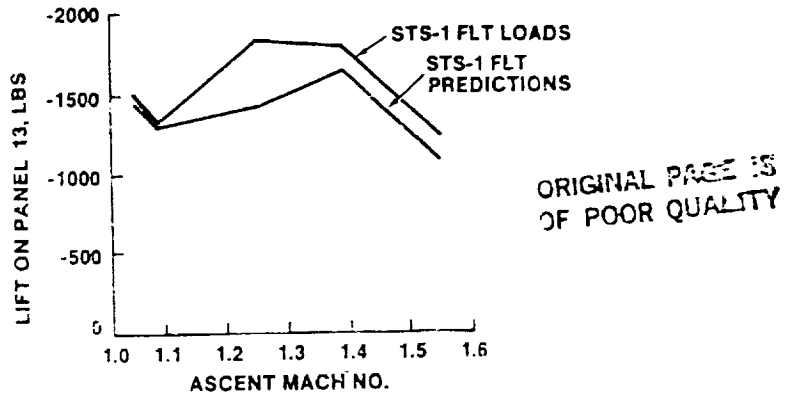


Figure 9.- Wing leading-edge panel-13 airloads comparison.

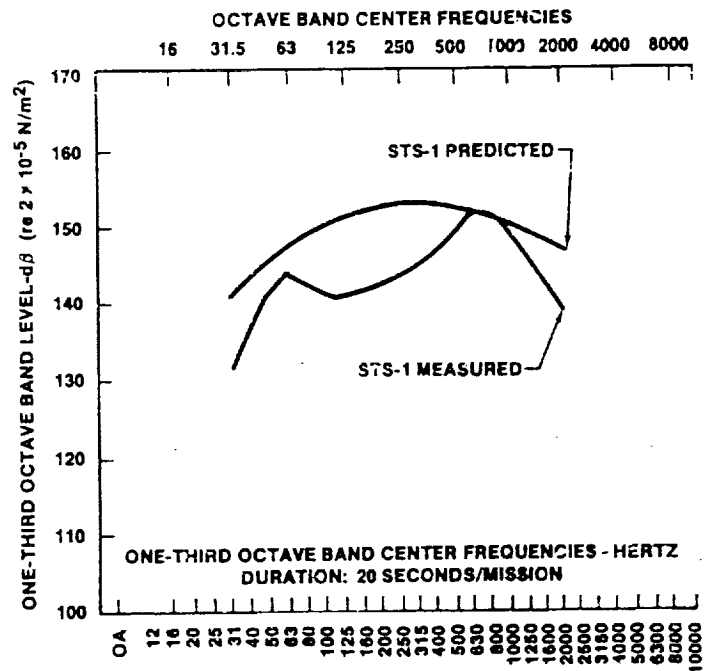


Figure 10.- Wing leading-edge acoustic environment comparison.

General flight performance characteristics can also be deduced from the flight data and the post-flight condition of the parts. Structurally, the leading edge has apparently achieved its purpose of maintaining the aerodynamic shape throughout the flight, inclusive of the elevated temperature regime during atmospheric re-entry. Elasticity of the RCC components can be deduced from the fact that the parts returned to their original shape and position after aerodynamic pressure and thermal-induced distortions. Noticeable markings on the side load restraint pins and on the RCC panel to RCC gap-seal mating surfaces confirm that the parts experience motions and displacement patterns compatible with the analytically predicted response. Although displacement magnitudes cannot be determined, visual inspections have revealed no anomalies and no conditions generating concern for the structural integrity of the leading-edge system.

LESSONS LEARNED

Combined results of LESS design studies and analytical iterations, development tests, qualification tests, and flight tests during the design, development, testing, and evaluation (DDT&E) phase of the Space Shuttle program have not only confirmed basic concept and design sufficiency but also revealed areas of design deficiencies. Modifications have in most cases been adopted to rectify those deficiencies.

Gap heating development tests on LESS interface geometric design concepts revealed a thermal anomaly at the RCC to RSI interface. Hot gas intrusion around the thermal barrier between the RCC and the RSI necessitated a redesign to incorporate a flow stopper. Evidence from STS-1 and STS-2 indicated the continuance of hot gas emanating from the lower interface region and flowing through the LESS cavity as illustrated in Figure 11. Modification of the flow stopper to eliminate this flow-through was incorporated and the effectiveness verified in the subsequent flights as indicated in Figure 12. Heating phenomena at the lower centerline region of the nose cap during STS-5 created a condition in the interface tiles unique to this flight. Hot gas penetration into the gap between two interface tiles caused slumping/melting of the tiles, thermal damage to the gap fillers, filler bars, and flow stoppers, and local melting of the aluminum carrier plate as depicted in Figure 13. Penetration of the hot gas damage into the insulation blankets was minimal, and no evidence of overheating of the nose cap Inconel attachments was found. Gap heating damage in this region during previous flights had been limited to tile slumping and filler bar overheating; therefore, this is assumed to be a problem associated with flow stopper and gap filler mission life.

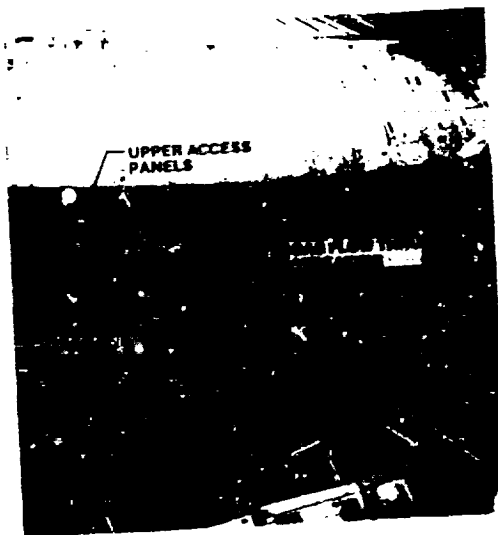


Figure 11.- Wing leading-edge hot gas flow.

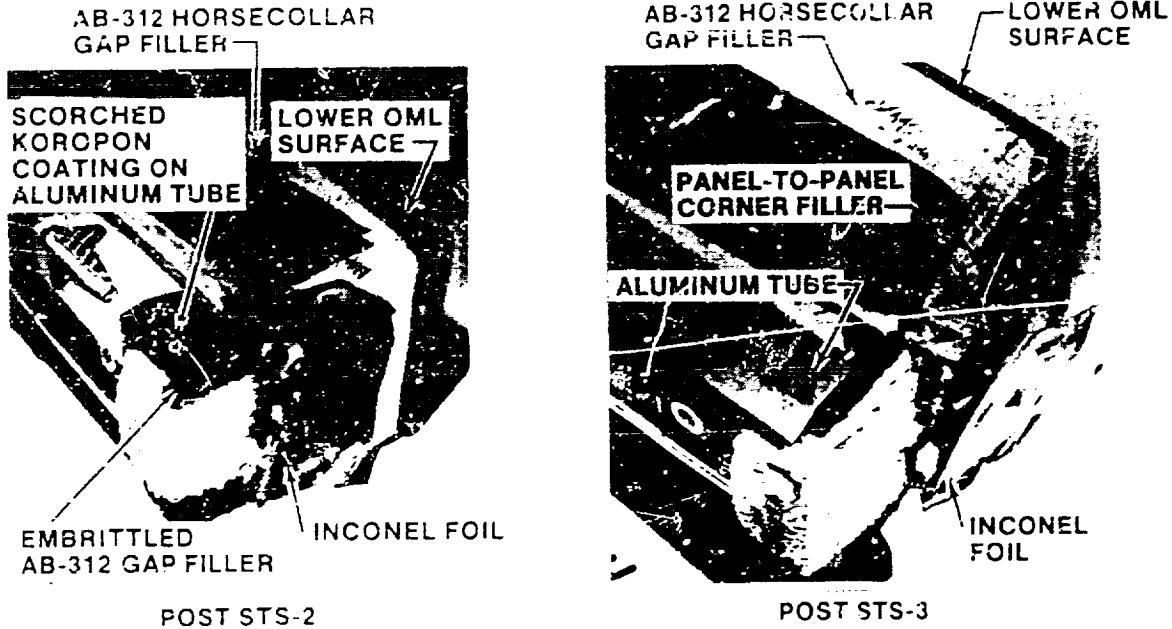


Figure 12.- Wing leading-edge panel-10 heating effects comparison.

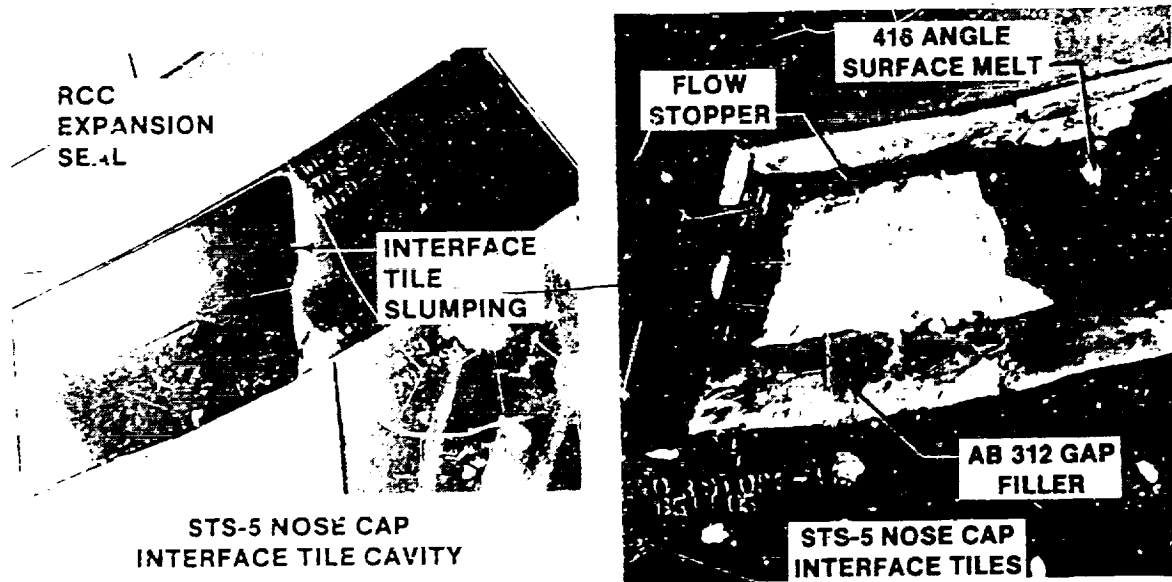
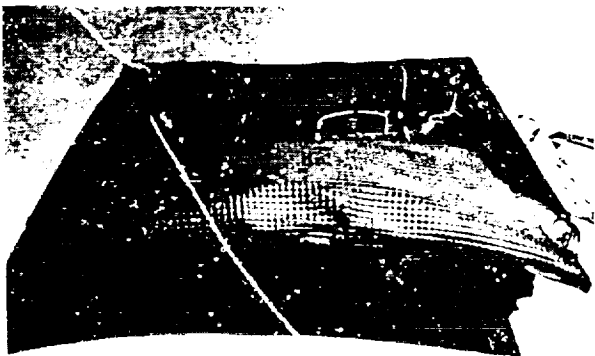


Figure 13.- Interface tile gap heating.

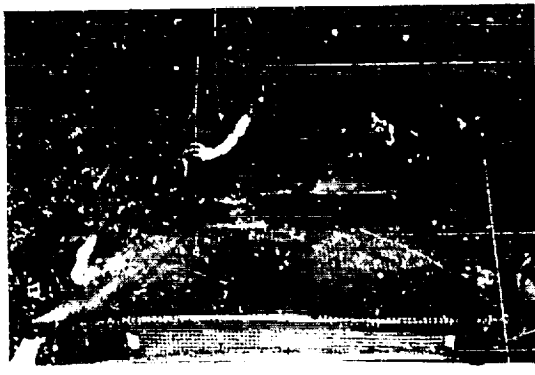
Cyclic exposure of the wing leading-edge internal insulation system to thermal and acoustic environments in component tests confirmed the reusability capability but also revealed a limited life of about thirty (30) missions. A comparison of the qualification test article after twenty (20) cycles to a typical Columbia insulation package after five (5) flights is presented in Figure 14. Both systems are capable of sustaining additional cycles.

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Insulation blankets of AB-312 wrapped Dynaflex used in the nose cap have a higher temperature capacity than the Inconel sheathed Dynaflex used in the wing leading edge; however, it is more susceptible to installation handling damage. Exposure to temperatures in the range of 2300°F causes the AB-312 fabric to become friable. Damaged insulation as a result of handling in the nose cap qualification test article after five (5) cycles is illustrated in Figure 15 and compared to the Columbia insulation after four (4) flights. Inspections of the insulation systems on the Columbia after four (4) flights reveal no deterioration, and the flight data has revealed no thermal performance degradation. Additional life tests will be used in conjunction with continued, scheduled inspection of the flight insulation system to establish the actual life capacity.



WING LEADING SPAR INSULATION
PHASE B THERMAL TEST
(20 OXIDIZING CYCLES)

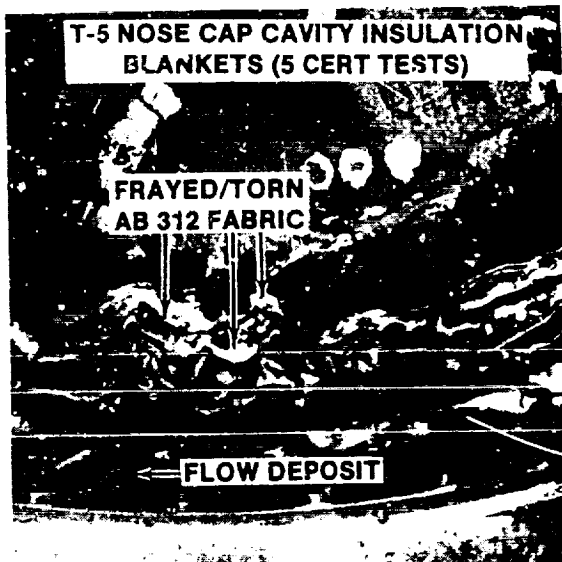


STS-5 WING LEADING-EDGE
SPAR INSULATION

Figure 14.- Wing leading-edge insulation system thermal effects comparison.



STS-4 NOSE CAP CAVITY
INSULATION BLANKETS



T-5 NOSE CAP CAVITY INSULATION
BLANKETS (5 CERT TESTS)

FRAYED/TORN
AB 312 FABRIC

← FLOW DEPOSIT

Figure 15.- Nose cap insulation system thermal effects comparison.

Related to the insulation system is a phenomena occurring on the wing front spar where the Inconel sheathed Dynaflex insulation is attached directly to the aluminum honeycomb spar. Corrosion of the aluminum, appearing as blisters in the Super Koropon paint as illustrated in Figure 16, is assumed to be a result of galvanic activity caused by the contact of dissimilar metals with the source of moisture being the humid air flowing through the LESS vent system. Direct exposure to the salt atmosphere could also be a contributing factor to the corrosion problem. The OV-102 wing front spar is constructed of aluminum honeycomb with face sheet thicknesses ranging from 4 mils to 120 mils that is painted with 1 mil of Super Koropon for corrosion protection. The insulation is contained in 4-mil thick waffled Inconel foil. Corrosion, occurring in discrete areas, created pits in the aluminum 80 to 100 mils in diameter and 1 to 14 mils in depth, some of which penetrated the face sheet. Ramifications of the corrosion range from no impact for the minor pitting to potential structural damage for the areas under major attack. Viable solutions include additional coats of Super Koropon and RTV to the painted aluminum surface, thereby providing a much more tolerant protection system.



Figure 16.- Wing front spar corrosion.

Moment constraint fittings, illustrated and compared to the basic design in Figure 17, were retro-fitted to the LESS design on several wing leading-edge panels as a result of a substantial increase in the predicted airloads. Joint tolerances and the design concept required by the temperature environment caused some speculation on the effectiveness of these fittings in reducing the critical RCC stresses. Substantiated by qualification tests and the absence of problems with the flight performance, the moment constraint fitting concept has proven to be an effective design "fix."

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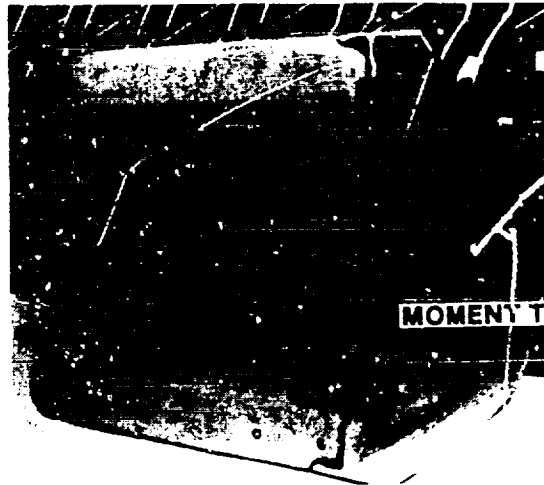


Figure 17.- Wing leading-edge panel attach hardware.

Wing leading-edge RCC panels are cantilevered off the front spar at four points by high-temperature, A-286 steel fittings. Figure 18 shows the general configuration. The initial material selection and sizing for these fittings were driven by expected temperatures (about 1000°F) and large airloads. In addition, producibility required a minimum gage of 100 mils for this very tough material. As the design environments matured and the analysis methods became more refined, peak temperatures dropped below 600°F. This allowed 6AL-4V titanium to be used in lieu of steel. At the same time, the load paths were optimized, resulting in one piece fittings. Producibility gains allowed the minimum gage to drop to 60 mils compared to the 100 mils required in the A-286 steel design. A weight reduction of about 300 pounds per shipset was realized with this change.

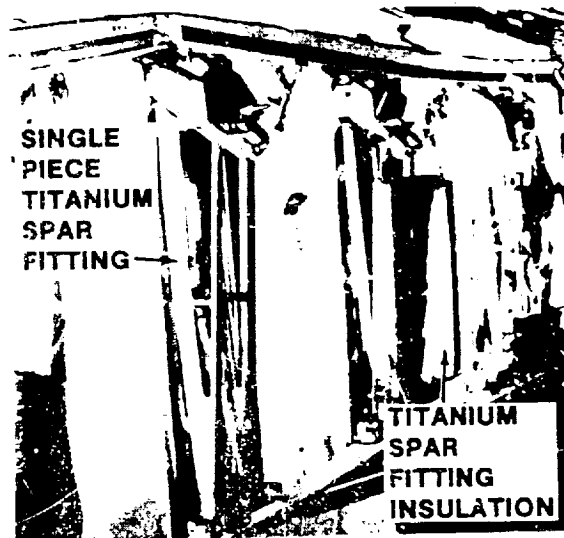
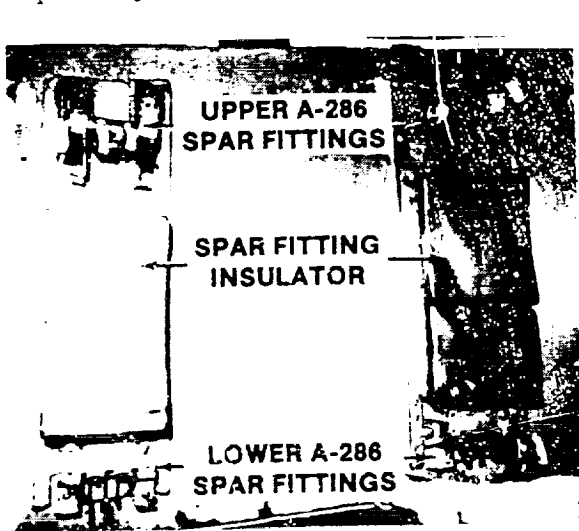


Figure 18.- Wing leading-edge spar fittings.

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Subsequent to the delivery of Columbia, element tests revealed the possibility of getting porous substrate in some areas of the production parts. High porosity in the substrate reduces the effectiveness of the basic Silicon Carbide (SiC) coating and the Tetraethyl Orthosilicate (TEOS) impregnation. Predicated on the local time at temperature history in the autoclave cure cycle, the high porosity is generally restricted to the external surface shell region. The consequence is an increase in the oxidation rate in the porous region and in some cases a reduction in the mission life of the affected parts. Reconciliation of this undesirable feature was achieved with a post-coating treatment of a sodium silicate and graphite fiber surface sealant (Type A). Potential mission life enhancement with the Type A surface sealant has been accomplished on all subsequent vehicles. Rework of the Columbia parts to add the Type A coating has been initiated after STS-5.

Susceptibility of the SiC coating to chipping, primarily ground handling damage on the edges of the RCC parts, necessitates a repair capability. Development of a repair procedure included a repair for major type damage that would be performed at the manufacturer's facility and a repair for minor type damage that could be performed at the launch site. Differentiation between major and minor damage is primarily determined by whether the black carbon substrate is exposed by the coating damage. Although the launch site repair would provide some protection from local oxidation for damage exposing the substrate, limitations have been placed on the procedure restricting it to one flight only. Several major coating repairs were made on the Columbia prior to STS-1, typically pictured in Figure 19. Additional launch site repairs were made subsequent to each of the first three missions. Flight exposure of these repairs has allowed several observations. Performance of the major repairs has been consistent with ground test results in that the repairs have remained intact with no appearance of shrinkage, craze cracking, or any other deleterious anomalies. Assuming congruency with test results, absence of flaws in the repair also suggests that the substrate is adequately protected from oxidation. Durability of the launch site repairs applied to minor damage areas also confirms the validity of this procedure to achieve and maintain the aerodynamic surface. Launch site repairs, utilizing the re-entry thermal environment to complete the cure process, will occasionally require touch-up to remove flaws that developed due to shrinkage or flow of the repair material.

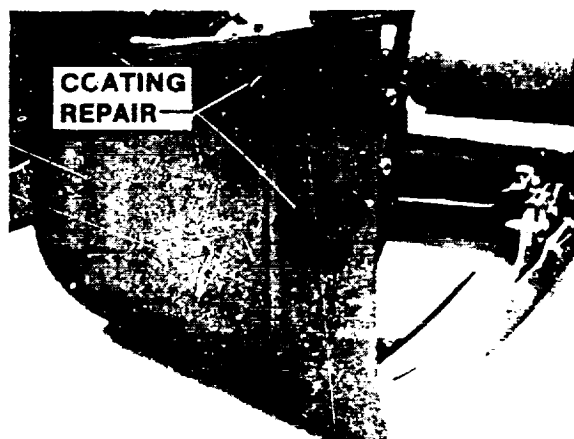


Figure 19.- Typical coating repair.

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The RCC arrowhead, illustrated in Figure 20, was a redesign and retro-fit of the original RSI tile design that failed the qualification tests of the explosive separation of the external tank. Configured in two pieces to facilitate installation around the attach mechanism, the RCC arrowhead parts are independently attached to the carrier plate with fasteners countersunk into the outer RCC surface. Design alterations required a rework of the Columbia arrowhead to provide a 45° bias joint instead of the joggle overlap at the interface of the two RCC components. Removal of the flange necessitated a major coating repair which, due to the lack of flight experience of coating repairs, caused a one-flight restriction to be placed on this particular assembly. Performance of the RCC arrowhead during STS-1 was superb, not only surviving the punishing explosive separation but also providing its primary function, along with the internal insulation, of thermal protection of the metal structure. Contrary to pre-flight, pessimistic expectations, the extensive coating repair exhibited no shrinkage or obvious detrimental effects from the initial flight exposure. Approval was therefore granted for an additional mission. In fact, this arrowhead eventually flew three missions prior to being replaced and used as a "guinea pig" to subjectively evaluate the multi-mission capability of an extensive coating repair and the integrity of the substrate around the attach holes. Sections taken in several areas revealed that the substrate around the attach holes locked good, contrary to the condition of the ground test article subsequent to the simulated separation tests. Interface conditions between the substrate and the coating repair were not ideal, but the repair was in excellent condition. The presence of localized mass loss was minimal and could actually have been caused by shrinkage of the repair material rather than oxidation of the substrate.

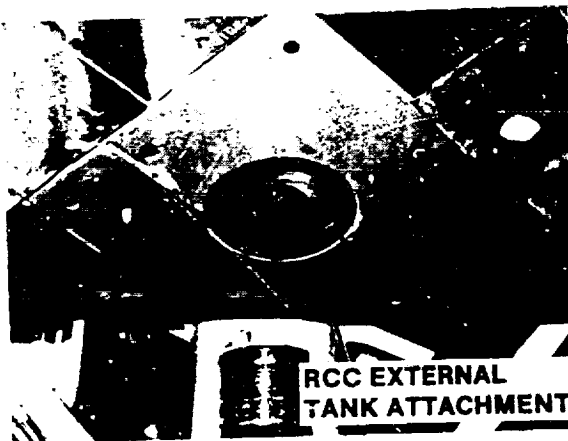


Figure 20.- RCC arrowhead.

CONCLUDING REMARKS

Successful completion of the four (4) development test flights and the initial commercial mission has demonstrated the adequacy of the Orbiter LESS design. Comparisons of measured and predicted temperatures and airloads have verified the analytical models used in the certification of the LESS for operational missions. Post-flight inspections not only confirmed the basic design concepts but also revealed areas of design deficiencies which have been modified to eliminate any potential operational problems. In summary, the LESS hardware performance has been outstanding with no degradation after the initial five (5) flights.

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