ADVANCED STRUCTURES TECHNOLOGY APPLIED TO A SUPERSONIC CRUISE ARROW-WING CONFIGURATION

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ABSTRACT

The application of advanced technology to a promising aerodynamic configuration was explored to investigate the improved payload-range characteristics over the configuration postulated during the National SST Program. Highlighted are the results of an analytical study performed by the Lockheed-California Company to determine the best structural approach for design of a Mach number 2.7 arrow-wing supersonic cruise aircraft. The data from this study, conducted under the auspices of the Structures Directorate of the National Aeronautics and Space Administration, Langley Research Center, established firm technical bases from which further trend studies were conducted to quantitatively assess the benefits and feasibility of using advanced structures technology to arrive at a viable advanced supersonic cruise aircraft.

INTRODUCTION

For the past several years, the National Aeronautics and Space Administration (NASA) Langley Research Center has been pursuing a supersonic cruise aircraft research program (SCAR) to provide sound technical bases for future civil and military supersonic vehicles, including possible development of an environmentally acceptable and economically viable commercial supersonic transport.

The design of a satisfactory advanced supersonic cruise aircraft requires reduced structural mass fractions attainable through application of new materials and concepts, and advanced analytical methods. Configurations, such as the arrow-wing (fig. 1), show promise from the aerodynamic standpoint; however, detailed structural design studies are needed to determine the feasibility of constructing this type of aircraft with sufficiently low structural mass. Past design studies have shown that excessive structural mass was required to satisfy the strength and stiffness requirements of the arrow-wing configuration. In addition, aerodynamic complications were accentuated at low-speed by the low aspect ratio, highly swept configuration.

This paper presents the highlights of the study conducted by the Lockheed-California Company to subject promising structural concepts to in-depth analyses, including the more important environmental considerations that could affect the selection of the best structural approach for design of wing and fuselage primary structure of a given Mach 2.7 arrow-wing supersonic cruise aircraft assuming a near-term start-of-design technology (ref. 1).

(Results of a similar study conducted by the Boeing Commercial Airplane Company are presented in ref. 2.)

This aforementioned structural evaluation (ref. 1) was conducted in three phases: 1) a design concept evaluation study wherein a large number of candidate structural concepts were investigated and evaluated to determine the most promising concepts; 2) a detailed engineering design-analysis study of the selected structural approach to define the critical design parameters and conditions, and the estimated structural mass of the near-term technology airplane; and 3) supplementary studies to identify opportunities for structural mass reduction resulting from the application of advanced technology to define a far-term technology airplane.

The structural evaluation of the near-term airplane involved detail analytical studies that encompassed airplane configuration refinement, design/manufacturing cost studies, and a structural evaluation involving the complex interactions between airframe strength and stiffness, static and dynamic loads, flutter, fatigue and fail-safe design, thermal loads, and the effects of variations in structural arrangements, concepts and materials on these interactions. Extensive use of computer programs and their associated math models were essential to perform the large-order analytical calculations required for this study. Math models were used in association with the aerodynamic heating, basic aerodynamics, external loads, internal loads and vibration and flutter analyses. In addition, interactive computer graphic programs were used in the flutter optimization and stability and control assessments.

The impact of the application of the various advanced structures technology to the near-term design airplane displayed performance gains realized by investing the mass savings into fuel/payload or in a resized (smaller) aircraft that would have the same performance at potentially lower cost. These trends, shown in figure 2, provided insight into future research requirements in the areas of advanced lightweight structural design concepts, advanced composite materials, advanced manufacturing approaches and active controls technology to provide a viable supersonic cruise aircraft.

More detailed results of the design concepts study including substantiating data and supersonic airframe technology recommendations are presented in references 1 and 3. A summary of the producibility technology studies is presented in reference 4.

CONFIGURATION

Reference Configuration

The reference configuration shown in figure 1 is a discrete wing-body airplane with a low wing which, in general, is continuous under the fuselage. The external shape of the airplane was defined at the design lift coefficient by a computer card deck supplied by NASA. This referenced configuration had neither a canard nor inboard leading-edge devices, but relied on the horizontal tail for pitch control and trim.

Configuration Refinement

Several areas of concern were identified with regard to the reference configuration, and refinements to these areas were examined and appropriate changes incorporated into the design. To provide suitable passenger accommodations in terms of comfort, baggage storage, cargo and passenger services, the fuselage depth was increased. A decrement in airplane lift-to-drag ratio equal to 0.10 resulted from this modification. A main landing gear concept was adopted which avoided the necessity for deviations from the NASA-supplied external contour, thus avoiding a drag penalty and minimizing the complexity and mass of the wing structure.

The low-speed pitch-up characteristics were examined using an interactive computer graphics technique that simulates, in real-time, the longitudinal behavior of the airplane response to control disturbances. Findings showed that if adequate control authority was provided, it was feasible to use the horizontal tail to provide automatic pitch limiting capability and good handling qualities. However, two requirements must be met: (1) a definite tail size to center-of-gravity relationship must be maintained, and (2) the pitch limiter system must be fail-operative. On the basis of these considerations, a minimum tail volume coefficient of 0.07 would yield an acceptable center-of-gravity range; with the further constraint that the airplane center of gravity be at 55 percent mean aerodynamic chord for the maximum landing mass.

Configuration development studies explored application of leading and trailing edge devices with auxiliary trimming surfaces (canards and horizontal tail) to provide schemes for supplementing the low-speed lift capabilities. The objective was to maximize the usable lift at takeoff attitudes considering in-ground effects. Methods of low-speed pitch stability improvement were also studied. This involved airplane balance, including the fuel system and its related tankage arrangement. On the final configuration a change in wing tip sweep from 1.13 rad (64.6 deg) as defined by the NASA-supplied data to a 1.05 rad (60 deg) sweep was made. This change reduced the demands on the longitudinal stability augmentation system and permitted a more aft center-of-gravity location with the existing horizontal tail power.

Final Configuration

The final airplane arrangement is shown on figure 3. The fuselage accommodates 234 passengers in five-abreast seating with an overall length of 90.5 m (296.9 ft) and a wing span of 40.4 m (132.6 ft). The leading edge sweep of the wing tip has been decreased to 1.05 rad (60 deg). The wing-mounted main landing gear employs a three-wheel axle design and retracts into a well just outboard of the fuselage.

The aircraft is equipped with a three-axis stability augmentation system (SAS) with adequate redundancy to be fail-operative. The primary control surfaces are indicated on figure 3 and includes an all-moving horizontal stabilizer with a geared elevator for pitch control. For yaw control, a fuselage mounted all-moving vertical tail with a geared rudder is provided. The tail volume coefficients for the horizontal stabilizer ($\overline{V}_H$) and the vertical tail ($\overline{V}_V$) are 0.07 and 0.024, respectively. The inboard wing flaps are used as lift devices at low speed. Leading edge flaps are provided on the outer wing for subsonic and transonic speeds, and ailerons on the trailing edge for low speed. At supersonic speeds, the inverted spoiler-slot deflector and spoiler-slot deflectors provide the primary roll control.

Four duct-burning turbofan engines, each with 398,560 N (89,600 lbf) of uninstalled thrust, are mounted in under wing pods having axisymmetric mixed compression inlets and have thrust reversers located aft of the wing trailing edge. The engine characteristics selected were based on the results of a NASA funded systems study (ref. 5). The engines are sized to provide a total thrust-to-airplane mass ratio of 0.36 at takeoff. The engine mounts are located aft of the wing rear beam and are attached to box beams which are cantilevered off the wing structural box.

The tank arrangement shown in figure 3 provides for a fuel storage capacity of 178,500 kg (393,600 lbm) with a significant portion of the total fuel for the 16 tanks stowed within the protected wing center section. Approximately 43 percent of the total storage capacity is contained in these "protected-volume" locations where the upper surface is exposed to the cooled and controlled environment of the fuselage cabin while the wing lower surface is shielded from the outside airstream by a fairing.

DESIGN CRITERIA

Evaluation of structural concepts for the Mach 2.7 supersonic cruise aircraft was based on an aircraft with an economic life of 15 years and a service life of 50,000 flight hours. The environment was determined from a design flight profile for an international mission which is approximately 3.4 hours in duration; three-quarters of that time, or 2.5 hours, is at Mach 2.62 (Hot Day) cruise.

For design purposes, a maximum taxi mass of 340,000 kg (750,000 lbm), a maximum landing mass of 191,000 kg (420,000 lbm), a payload of 22,000 kg (49,000 lbm), and a design range of 7800 km (4200 nmi) were specified for the airplane.

Maneuver loads analyses were based on solution of the airplane equations of motion for pilot-induced maneuvers. Except where limited by a maximum usable normal force coefficient or by available longitudinal controls deflections, the limit load factors (n_z) were as follows: (1) Positive maneuvers: n_z = +2.5 at all design speeds; (2) Negative maneuvers: n_z = -1.0 up to V_C[1] and varies linearly to zero at V_D; (3) Rolling maneuver entry load factors: an upper limit n_z = +1.67 at all design speeds, with a lower limit n_z = 0 up to V_C and varies linearly up to +1.0 at V_D.

[1]Symbols and abbreviations not defined in the text and figures may be found in the appendix.

Fatigue analyses were based on a representative loading spectrum developed for the National SST Program which provides a moderately conservative representation of a loading history for supersonic cruise aircraft. The reference load levels and oscillatory flight loads included representative tensile thermal stress increments and ground loadings. The basic fatigue criterion was to provide a structure with a service life of 50,000 flight hours. Appropriate multiplying factors were applied to the design life for use in establishing allowable design tension stresses. For structure designed by the spectrum loadings, the allowable stresses were defined using a factor of 2 times the design service life of 50,000 hours. For areas of the fuselage designed by constant amplitude cabin pressure loading, the allowable stresses were based on 200,000 flight hours of service (50,000 x 4).

A fail-safe design load of 100-percent limit load was used for the analysis of the assumed damage conditions. The residual strength of the damaged structure must be capable of withstanding these limit loads without failure.

The selection of minimum gages for regions not designed to specific strength or fatigue requirements was based on consideration of the structural concept employed, fabrication constraints, and foreign object damage (FOD) effects.

STRUCTURAL DESIGN CONCEPTS

A spectrum of structural approaches for primary structure design that have found application or had been proposed for supersonic aircraft, such as the Anglo-French Concorde supersonic transport, the Mach 3.0 plus Lockheed YF-12, and the proposed Lockheed L-2000 and Boeing B-2707 supersonic transports, were systematically evaluated for the given configuration and design criteria.

Design and manufacturing concepts studies (ref. 4) established feasibility of the application of advanced manufacturing techniques to large-scale production. Basic design parameters and design guidelines were established for each structural arrangement and each concept to provide consistency between manufacturing design studies and analyses.

Candidate materials included both metallic and composite material systems. Alpha-Beta (Ti-6Al-4V) and Beta (Beta C) titanium alloys, both annealed and solution treated and aged, were evaluated to identify the important characteristics for minimum mass designs as constrained by the specified structural approach and life requirements.

Among the composite materials considered were both organic matrix (graphite-polyimide, boron-polyimide) and metallic matrix (boron-aluminum) systems. Selective reinforcement of the basic metallic structure was considered as the appropriate level of composite application for the near-term design. Furthermore, based on the principle of maximum return for minimum cost and risk, the application was primarily unidirectional reinforcing of members carrying axial loads, such as: spar caps, rib caps and stiffeners of wing and fuselage panel designs.

Wing Structure Concepts

The structural design concepts for the wing primary load-carrying structure are shown in figure 4 and consists of two basic types: monocoque and semimonocoque constructions.

Monocoque construction (fig. 4a) consists of biaxially-stiffened panels which support the principal loads in both the span and chord directions. The substructure arrangement consists of both multirib and multispar designs.

The biaxially-stiffened panels considered were the honeycomb core and the truss-core sandwich concepts. The honeycomb core panels were assumed to be aluminum brazed (Aeronca process); whereas, both diffusion-bonded and welded (spot and EB) joining process were assumed for the truss-core sandwich panel configuration.

In the monocoque concepts, as well as in all the other primary structure concepts, circular-arc (sine-wave) corrugated webs were used at the tank closures. Truss-type webs were used for all other areas. The caps of the spars and ribs are inplane with the surface panels for the monocoque concepts to minimize the effect of eccentricities.

The two types of semimonocoque concepts are: (1) panels supporting loads in the spanwise direction (fig. 4b), and (2) panels supporting loads in the chordwise direction (fig. 4c). Both contain the same type of rib and spar webs as the monocoque structure. Discrete spar and rib caps are provided for the semimonocoque concepts since the spar cap or rib cap must support the inplane loads acting normal to the panel stiffeners.

The spanwise-stiffened wing concept is essentially a multirib design with closely spaced ribs and widely spaced spars. The surface panel configurations shown in the figure have effective load-carrying capability in their stiffened direction with smooth skins required for aerodynamic performance. Zee-and hat-stiffened designs are examples of these wing concepts.

The chordwise-stiffened arrangement is essentially a multispar structure with widely spaced ribs. The surface panel concepts for this arrangement have stiffening elements oriented in the chordwise direction. Structurally efficient beaded-skin designs were explored, e.g., circular-arc convex beaded skin concept. These efficient circular-arc sections of sheet metal construction provide effective designs when properly oriented in the airstream to provide acceptable performance, as demonstrated on the Lockheed YF-12 aircraft. The shallow depressions or protrusions provide smooth displacements under thermally induced strains and operational loads and offer significant improvement in fatigue life. Panel spanwise thermal stresses are minimized by allowing thermal deformation of the curved elements. Submerged spar caps are provided except at panel closeouts and at fuel tank bulkheads.

Selective reinforcement of the basic metallic structure (figure 4d) was considered as the appropriate level of composite application for the near-term design. The chordwise-stiffened arrangement described above provides the basic approach offering the maximum mass·saving potential and was used for the exploration of composite reinforcing. In addition to the surface panels, multi-element (fail-safe) composite reinforced spar cap designs were investigated.

Fuselage Structure Concepts

The structural design concepts initially considered for fuselage design included both sandwich shell construction and skin-stringer and frame shell construction.

The sandwich shell design was thought to have a potential mass savings over the more conventional skin-stringer and frame design, with specific advantages with regard to sonic-fatigue resistance and reduced sound and heat transmission. Preliminary structural design and analyses were conducted to assess the potential mass savings benefit and manufacturing/design feasibility of a sandwich shell. The manufacturing complexity and the parasitic mass which the sandwich must carry, in terms of core and bonding agents, proved to be a disadvantage, and thus this concept was not included as part of the study.

Hence, the basic structural arrangement considered for the design fuselage was that of a uniaxially stiffened shell with closely spaced supporting frames. The panel configurations with the most potential were the zee-stiffened and the hat-stiffened configurations. In addition, these stiffener concepts all contain flat elements which are amenable to composite reinforcing. Supporting frames that merited consideration were both the fixed and floating type. The joining methods evaluated for fabricating these concepts include mechanical fastening, welding and bonding.

DESIGN METHODOLOGY AND ANALYSES

A systematic multidisciplinary design-analysis process was used for the structural evaluation. The corresponding analytical design cycle is illustrated in figure 5. The evaluation encompassed in-depth studies involving the interactions between airframe strength and stiffness, static and dynamic loads, flutter, fatigue and fail-safe design, thermal loads, and the effects of variations in structural arrangement, concepts and materials on these interactions. Due to the complex nature of these studies, extensive use was made of computerized analysis programs, with the predominant use of Lockheed-California Company's integrated NASTRAN-FAMAS structural analysis system. The system incorporates the Lockheed-California Company-modified version of the NASTRAN finite element analysis program and the Company's FAMAS program system for aeroelastic loads and flutter analysis.

Design Concepts Evaluation

To initiate the structural evaluation, an investigation was conducted using a single finite-element model to obtain a representative set of wing and fuselage load intensities for selective maneuver conditions. These load intensities were used in conjunction with computer sizing programs to obtain representative values of structural stiffness for each general type of wing load-carrying structure, i.e., chordwise-stiffened, spanwise-stiffened, and biaxially-stiffened wing surface panels, and for a representative skin-stringer-frame fuselage shell. Using these stiffnesses, NASTRAN finite-element structural models were established for each general type of structure. To conserve resources during this investigation these models were "two-dimensional", that is, they are generated to be symmetrical about an assumed flat mean camber surface. One half the airplane was represented with 1300 elements and approximately 1050 degrees of freedom.

Internal forces/stresses and deflections were obtained for each general type of structure using the appropriate structural model and the corresponding aeroelastic loads caused by maneuver conditions (based on subsonic and supersonic potential-flow theories), and ground operations (based on company experience and the requirements of FAR 25). These internal forces were supplemented with pressure and temperature data to define the load-temperature environment used for conducting the point design analysis.

Three areas on the wing and four areas on the fuselage were selected for conducting point design analyses of the candidate structural concepts. Each area represented a different general structural requirement and was sized using the aforementioned load-temperature environment derived from the appropriate finite-element model; e.g., the internal loads for the chordwise finite-element model were used to analyze the candidate chordwise wing panel concepts.

The point design analyses were conducted using structural optimization computer programs and resulted in a ranking by mass of each of the structural concepts. The least-mass concept (most promising) of each general arrangement (i.e., chordwise-stiffened, spanwise-stiffened and monocoque) was selected and subjected to further point design analysis for three additional wing regions. Total mass data of these strength-sized concepts were obtained by extrapolation of the unit mass of the point design regions over the remainder of the structure. These strength-sized arrangements were evaluated for damage tolerance, flutter, and the effects of aeroelasticity on stability and control.

Vibration and flutter analyses were performed on each general arrangement using the stiffness matrix derived from the finite-element model condensed by Guyan reduction (ref. 6) to 188 and 178 degrees of freedom, symmetric and antisymmetric, respectively. The inertia matrices were formed for two airplane masses: the operating mass empty and the full fuel and full payload condition. These conditions represent the extremes of minimum and maximum airplane mass. No intermediate mass conditions were examined. Flutter analyses encompassed both symmetric and antisymmetric boundary conditions for selected Mach numbers. Flutter deficiencies were corrected by use of an interactive computer system (ref. 7) where sensitivities to operator selected variables were determined and structural parameters incremented by the operator. New modes and frequencies were calculated for each structural change because of the nonlinear stiffness effect introduced by the Guyan reduction process. This method provided a good estimate, in short time, of the amount and location of required additional structural material. This mass penalty was added to the strength-sized structure mass to obtain a total mass estimate for the airframe.

All of the primary structures, analyzed for consistent load-temperature criteria, are satisfactory from the standpoint of static aeroelasticity, lifting surface flutter, static and dynamic loads, fatigue and fail-safe design, acoustics, thermal stress, and stability and control.

For each of the design concepts, advanced producibility techniques considering the use of welding, brazing or bonding technology were applied. Extensive use of welding and bonding resulted in improved fatigue quality through minimizing fasteners and the number of manufactured joints, and elimination of tank sealing.

Detailed mass breakdowns and comparisons are given in reference 1.

Concept Selection

The wing primary-structure design concepts were ranked (table 1) on the basis of relative mass (constant gross mass airplane). When these primary-structure concepts were applied to constant payload-range aircraft (by interaction evaluation of structural mass, cost, and performance), the ranking of the primary-structure concepts was unchanged, and the relative direct operating cost shown on the table were obtained.

The relative cost increases show the effect that structural efficiency has on overall cost. Small mass inefficiencies evaluated under range-payload constraints can and do raise costs appreciably.

The results of this phase of the design study indicated that a hybrid design using a combination of a chordwise-stiffened and monocoque wing structural arrangement (fig. 6) has least mass and cost and thus provides the most promising arrangement for further detailed evaluation. The stiffness critical wing tip is monocoque construction to make use of the high biaxial stiffness of the aluminum brazed titanium honeycomb sandwich to satisfy the flutter requirements. In the remainder of the wing, low-profile convex beaded surface panels of weld bonded titanium alloy (6Al-4V) are used. The cover panel stiffening is oriented in the chordwise direction with discrete spanwise submerged titanium spar caps reinforced with unidirectional multielement boron-polyimide composites. The fuselage has a hat-stiffener shell design with supporting frames.

Although the Beta alloy showed desirable strength properties and fabrication benefits, it did not exhibit the density compensated elastic properties for minimum mass for the surface panel design.

Engineering Design Analysis

Detailed engineering design analyses of the hybrid design concept were made to define the critical design conditions and requirements, and estimate the structural mass of the near-term technology airplane.

A more detailed 3-dimensional finite-element model was developed and used as the basis for the final structural analyses. The finite-element model (fig. 7) contains approximately 2200 degrees of freedom and 2450 elements. The external loads, internal forces, and displacements for the hybrid design were determined. Strength sizing and one resizing were conducted at six wing regions and four fuselage regions.

The allowable stresses and distribution of the structural material reflected strength requirements, fatigue effects (both load and sonic) and damage tolerance consideration for a commercial airplane. In addition, material distribution was constrained by fabrication and minimum gage design considerations.

The results of these analyses defined a strength-level design. Flutter characteristics for this airplane were then determined at the Mach numbers of 0.60, 0.90, and 1.85 to assess the additional stiffness requirements to correct flutter deficiencies.

The math model for the near-term airplane incorporated the additional stiffness dictated by aeroelastic requirements as well as design/manufacturing considerations to provide a realistic structural design. Structural influence coefficients, internal loads, and aeroelastic displacements were calculated for this airplane.

Relative to the flutter-speed requirements defined from the operating envelope all Mach numbers investigated have adequate flutter margins of safety. Roll-control reversal speeds and FAR requirements were compared for both the normal scheduled surface cominations and for selected fail-safe conditions which involve loss of a surface which has the most adverse effect on roll-control reversal speed. In all cases, the airplane met or exceeded the specified requirements.

NEAR-TERM TECHNOLOGY AIRPLANE

The near-term technology aircraft has a takeoff gross mass of 340,000 kg (750,000 lbm), and a wing loading of 334 kg/m^2 (68.7 lbm/ft^2). A fail-operative ("hardened") three-axis stability augmentation system (SAS), integral to the primary control system concept, is provided. Active controls such as flutter mode control and load alleviation were not included. The zero-fuel mass for the aircraft is 164,600 kg (362,800 lbm). The near-term aircraft reflects a 1980 start-of-design technology employing titanium alloy 6Al-4V as the primary construction material with composite materials accounting for approximately 7 percent of the airframe mass.

Wing Structure

The hybrid wing design shown in figure 6 employs a combination of chordwise-stiffened and monocoque structural arrangements. Structurally efficient convex-beaded surface panels are used in both the forward and aft box. Submerged titanium alloy spar caps, reinforced with multi-element unidirectional boron-polyimide composites, are found in the strength-critical aft box structure. Design details of the chordwise-stiffened beaded surface panel and structure are shown in figure 8. With the beaded skin design, wing bending material is concentrated in the spar caps and the surface panels primarily transmit the chordwise and shear inplane loads and out-of-plane pressure loads.

Weldbonding is the basic method proposed for joining the inner and outer skins of the surface assembly. Surface panel size was held to 4.6 x 10.7 m (15 x 35 ft). The length limit was based on tooling considerations for hot vacuum forming of the skin while the width limit was based on postulated size of spot-welding equipment.

In locating wing spars in the chordwise-stiffened wing area, a minimum spacing of 0.53 m (21 in) was maintained between constraints such as fuel tank boundaries. Wing rib spacing was a nominal 1.52 m (60 in) but was modified as required to suit geometrical design constraints. These dimensions define minimum mass conditions which were determined through studies involving various spar and rib spacing. In the chordwise-stiffened and transition areas, welded truss spars were used except where a spar serves as a fuel tank well. At such locations, spars have welded circular arc webs with stiffened "I" caps. To facilitate fuel sealing, surface beads do not extend across tank boundaries. Wing spars in the aft wing box were fabricated as continuous subassemblies between BL 470 L and R. Boron-polyimide was selected for the spar cap reinforcement for its structural efficiency. The multielement approach results in damage tolerance capability.

Aluminum brazed titanium alloy honeycomb-core sandwich panels are used in the stiffness critical wing tip region.

The sandwich surfaces were brazed together using 3003 aluminum alloy as the brazing material (the "Aeronca" process). Welded circular-arc spars and ribs were used since the minimum need for web penetrations allows the realization of their inherent minimum mass and design simplicity feature. Composite reinforcement was not used in the brazed surfaces or the welded circular arc spars and ribs. A size limit of 1.73 x 12.19 m (5.66 x 40 ft) for brazed surfaces was postulated as a guide after consultation with Aeronca. The panel configurations were based on the design philosophy that all or some panels of the upper surfaces are attached with screws and are removable for inspection and maintenance purposes.

The flexibility of the aluminum braze process was exploited by incorporating crack stoppers and panel edge doublers in the surface panel bracements. Also, the capability of tapering the panel thickness was utilized in the joint between the chordwise and monocoque surface areas. In this joint area, where transition in arrangement was made, the outboard sandwich surfaces were extended inboard so that spanwise components of the outboard sandwich surfaces were extended inboard so that spanwise components of the outboard surface loads due to wing bending loads are transferred directly to the chordwise-stiffened structure at the interface rib.

Fuselage Structure

The fuselage shell incorporates machined extrusion stringers, crack-stoppers between frames, and floating zee frames with shear clips. Closed hat-section extruded stringers which provide structural efficiency were proposed to be machined to provide for crack stoppers and to vary stringer thickness. Extruded stringers also were found to be well suited to effective installation of composite reinforcement. The floating zee frames with shear clips were considered preferable, from a fatigue standpoint, to full depth frames having notches for stringers. Also, zee frames avoid the offset shear center associated with channel section frames.

Weldbonding was proposed to be used for attaching frames, stringers and crack-stoppers to the skin because of economy, minimum mass, good fatigue characteristics, and the avoidance of sealing problems. Satisfactory weldbonding of three thicknesses, as encountered at some locations, may require development. Weldbrazing was considered as a possible backup to weldbonding. Where fasteners were used at shear clips and frame/stringer attachments, fastener-bonding was utilized in lieu of fasteners alone to obtain enhanced properties. The size of fuselage skin panel assemblies has been limited to 4.57 x 15.25 m (15 x 50 ft); the former is based on the postulated size of spotwelding equipment, the latter on the postulated length of the adhesive curing ovens.

Longitudinal skin-panel splices were located only at the top centerline of the fuselage and at the floor/shell intersections fore and aft of the wing carry-through area. These longitudinal splices utilize external and internal splice plates in conjunction with fastener-bonding to achieve a double shear splice having damage tolerance capabilities and good fatigue properties. Suitable combinations of fastener size and external splice-plate thickness were utilized to avoid feather edges at countersinks for flush fasteners. At circumferential panel splices and other locations as required, feather edges were avoided by incorporating thickened pads in the external skin in a manner similar to that for wing skins. Chemical milling was used to vary fuselage skin thickness in accordance with load requirements.

Critical Design Conditions and Requirements

The design conditions and requirements that sized various portions of the near-term design are shown in figures 9 and 10. In figure 9, the upper and lower surface of the wing are divided into three distinct zones according to the design requirements that dictated structural sizes.

The tip structure was stiffness critical and sized to meet the flutter requirements. The highly loaded aft box and some portions of the forward box structures were strength-designed to transmit the wing spanwise and chordwise bending moments and shears. The forward box structural sizing resulted in surface panels and substructure components with active minimum gage constraints. Foreign object damage was the governing criterion for selection of minimum gage.

The conditions which displayed the maximum surface panel design loads are presented in figure 10. An exception was the tip structure which was stiffness critical for the Mach 1.85 condition. The supersonic cruise aircraft displayed critical loads at transonic and low supersonic Mach numbers wherein the structural temperatures did not influence the design appreciably. Although major areas of the wing lower surface were impacted by the thermal environment, analysis of surface panels and substructure using the applicable load-temperature environment resulted in the symmetric maneuver condition at Mach 1.25 as the critical design condition. The upper surface in the forward box was constrained by the minimum gage criterion. The forebody shell region was loaded principally by fuselage pressurization, and therefore critical for the operational environment at Mach 2.7. The constant amplitude loading imposed upon this structure requires reduced allowable stresses to achieve the life requirements.

The fuselage design was influenced by the high temperature environment for the major portion of the upper shell and the pressure critical forebody shell. As indicated on figure 10 a major part of the shell structure was bending critical; the lower shell being critical for the dynamic landing conditions. The forebody and aftbody conditions display critical downbending which occur at varying time from main landing gear impact.

Airplane Mass Estimates

Detailed mass descriptions of the wing and fuselage are presented in tables 2 and 3, respectively. The wing mass description includes fail-safe provisions, allowance for flutter prevention, and panel-thickness changes for manufacturing/design constraints. The fixed mass consists of those items invariant with box structural concept, such as engine-support beams, and leading and trailing edge structure. The fuselage mass was also divided into two major categories: shell mass and fixed mass. Here again the shell mass was dependent upon structural concept while the fixed mass such as doors, windows, flight station, and fairings were invariant.

The mass properties for the near-term technology airplane were determined as shown in table 4 as an Estimated Group Mass Statement. The data reflects a fixed size aircraft with a takeoff gross mass of 340,000 kg (750,000 lbm) and payload of 22,000 kg (49,000 lbm).

ADVANCED TECHNOLOGY ASSESSMENT

Starting with the near-term design, projections for airplane structural mass were determined for an aircraft employing technologies beyond the 1980 time period. It was postulated that by the 1990's advanced composites using polyimide resin systems for long-time application in the Mach 2.7 environment would be sufficiently matured to be aggressively used for both primary and secondary structural application. Similarly, advances in the titanium technology would be in-hand to apply to specific regions of the airframe for reduced mass and cost. Furthermore, advanced controls concepts would be employed in reducing structural mass as well as reduce normal accelerations to provide satisfactory ride quality and fatigue damage control.

Advanced Composite Technology

Projected composite development trends postulated the availability of improved stable high temperature resin systems such as thermoplastic polyimides or high temperature polyaromatics, large numerically controlled tape laying equipment, filament winding and pultrusion equipment, and larger autoclaves. Reference 1 studies indicated that, with aggressive application of composite materials and fabrication technology, the payload-range characteristics could be improved by 12 percent for a constant gross mass aircraft or the takeoff mass reduced 14 percent for a given design payload-range goal.

To obtain the mass for the wing and fuselage primary structure, the results of the design concepts evaluation study were employed to size specific point design regions in graphite-polyimide and boron-polyimide The sizing data included the internal loads and stiffness requirements for the appropriate airframe arrangement (i.e. chordwise-stiffened and monocoque designs). A comparison was then made between the near-term design and similar designs in graphite and boron composites. For secondary structure and other structural components (i.e. landing gear, nacelle, etc.), reduction factors were used based on Lockheed experience.

Evaluation of the wing box mass data for the near-term design and the composite material system design (ref.1) indicated the mass advantage of the minimum gage titanium alloy beaded panels of the forward box, as compared to an equivalent stiffness composite design of either graphite-polyimide or boron-polyimide. For the strength critical aft box and stiffness critical wing tip structure, an all-composite design indicated a 5-percent savings in total wing box mass. Composite application to the fuselage shell reflected a decrease in shell unit mass at all point design regions; the magnitude varied from 4 percent to 21 percent. A mass saving for the total shell when employing composites was 13.7 percent. A mass reduction factor applied to the secondary and other structural components (i.e. tail, nacelle, etc.) resulted in 9,500 kg (21,000 lbm) savings. These items alone offered a significant mass payoff and potentially improves the aircraft performance by approximately 650 km (350 nmi).

These data, although preliminary in nature, show that advanced composites application to the far-term structural approach offers significant improvement in the fuel fraction for the constant gross mass airplane. The 12,200-kg (26,800 lbm) total mass saving relates to a range increase of 830-km (450-nmi) or a total range potential of 8,630 km (4,650 nmi).

Advanced Controls Technology

The near-term technology airplane postulated the use of a 3-axis stability augmentation system (SAS) that was fail-operative. The mass benefits of reduced tail size were incorporated into the design, as well as the additional mass required for automatic sensors which detect motions (yaw, pitch, roll) of the aircraft and results in the actuation of the normal flight control to provide artificial stability.

Two other potential sources of structural mass reduction related to the application of active controls were identified on the near-term design airplane. They include: load alleviation and flutter mode control. The design conditions and requirements of the near-term airplane showed that the aft box structure was strength-critical. The wing mass data also indicated that the spars which transmit the bending moments and shears had a mass of 3,890 kg (8,570 lbm). By the application of an active load alleviation concept, it was postulated that the span load distribution could be sufficiently altered by deflection of the trailing edge devices so as to appreciably reduce the bending moments during maneuver. An overall 25-percent reduction in bending requirements could potentially reduce the structural mass by 950 kg (2,100 lbm). To suppress the critical flutter modes envisioned the use of the trailing edge surfaces which were automatically actuated to increase aerodynamic damping. It was postulated that sufficient structural stiffness was required to meet the dive speed (V_D) boundary at all Mach numbers, and that the stiffness increment required to achieve 1.2 V_D could be potentially eliminated. Thus, using this premise, the results of flutter optimization studies (ref. 3) were reviewed and a possible mass savings of 720 kg (1,600 lbm) was forecasted for the wing tip structure. Collectively these two advanced control concepts have a potential for reducing 1,670 kg (3,700 lbm) from the structural mass affording an increase in range of 110 km (60 nmi).

Advanced Technology Trends

A major potential source for structural mass reduction identified on the near-term aircraft and through the advanced technology assessment was the increased use of advanced composite materials. This was particularly true when the cascading effects on the aircraft size and cost were considered. The application, however, must be consistent with the projected start-of-design date and the availability date of composite materials and manufacturing technology.

A sequential application of new technology to the near-term airplane is displayed in figure 11. These trends postulate the availability of the technology because of the requirement of minimum development and risk, and/or as a direct fallout of technology currently pursued by government and industry.

(1) The near-term design (1980 technology) is shown as an aircraft with a takeoff gross mass of 340,000 kg (750,000 lbm) and a range of 7800 km (4200 nmi). The aircraft has a fail-operative three-axis stability augmentation system concept and approximately 7-percent composite material application.

(2) Design changes were made to the near-term design by applying composite materials to the secondary structure of the wing and fuselage and to the empennage structure and wing verticals. These structures were related to the requirement for minimum development and risk, thus identified as 1985 technology. The secondary components are categorized as those that are noncritical to flight safety components, inspectable components where damage would be apparent in routine maintenance operations, and repairable or replaceable components. For this assessment, control surfaces and leading edge structures were included in this category, recognizing that proper allowance for temperature must be made in selecting materials and allowables. The composite empennage postulates technology transfer of current development of an advanced composite vertical fin for the L-1011 aircraft being pursued by Lockheed-California Company under the auspices of NASA Langley (ref. 8). The aforementioned level of application provides a mass savings of 5,670 kg (12,500 lbm) and a range increment of 390 km (210 nmi) for the 340,000 kg (750,000 lbm) aircraft. This composite material usage accounts for approximately 23 percent of the structural mass.

(3) Further range improvement of 110 km (60 nmi) was realized by the application of load alleviation and flutter mode control. Mass saving in the flutter critical wing tip was realized by providing structure to meet the V_D requirement at the critical Mach number for the bending and torsion mode flutter. Appropriate reduction in the bending material requirement of the aft box structure was also made. The composite materials application and technology level remains as determined for the design change (2). Early introduction of active controls technology (ACT) is based on a current program which includes the demonstration of near-term feasibility of ACT for commercial application as part of NASA's Aircraft Energy Efficiency/Energy Efficient Transport Program. The program looks to the application of active controls to the Lockheed L-1011 for increased energy efficiency, with application to today's fleet as early as 1981.

(4) The aggressive application of advanced composite materials, active controls concepts, and advanced production technologies were the basis for defining the far-term (1990 technology) airplane trends.

Composite application encompassed the previously identified secondary structure and empennage plus the wing aft box and tip, and the fuselage shell. The composite material mass represents 51 percent of the total structural mass of the aircraft. A range increment of 430 km (230 nmi) was realized for the 340,000 kg (750,000 lbm) aircraft.

These various "state-of-the-art" aircraft were constrained to a constant takeoff gross mass of 340,000 kg (750,000 lbm) and all equal or exceed the design range goal of 7800 km (4200 nmi). Resizing these aircraft for varying degrees of advanced technology are noted by the decreasing mass and range trends.

FAR-TERM TECHNOLOGY AIRPLANE

The far-term airplane employs those technologies available for application beyond the frame work of the near-term (1980 technology) approach. Potential areas include advanced composite materials, advanced controls concepts, and cost reducing production approaches. It is postulated that the various potential technologies will be pursued in a timely manner through appropriate research and development and be available for application by the year 1990.

The hybrid wing structural approach shown in figure 12 employs both advanced metallic and composite materials, and a combination of the chordwise-stiffened and monocoque arrangements. The convex-beaded surface panel concept of titanium alloy 6Al-4V with the submerged spar caps resulted in minimum mass for the forward box structure. Advanced titanium manufacturing concepts, such as the Lockheed-California Company's low cost/no draft precision titanium forging technology and Rockwell-International's superplastic forming-diffusion bonding technology may find application for reduced mass and cost. Both concepts eliminate machining requirements and, in particular, the low cost/no draft precision titanium forging technology has been successfully applied to a rather complicated structural component used on the Lockheed L-1011 aircraft. A mass savings in raw material of 91 percent over the current method of machining from a plate stock was demonstrated with a total cost reduction (per part) of 77 percent realized.

A graphite-polyimide honeycomb sandwich panel concept is used in the strength and stiffness critical aft- and tip-box structure, respectively. For environmental protection of the composite material system, aluminum wire fabric is also employed. Composite materials account for approximately 58 percent of the wing structure mass.

The fuselage shell is a graphite-polyimide skin-stringer-frame design. In the pressure critical forebody a T-stiffened skin is employed; the closed-hat stiffened skin is used in the more highly loaded centerbody and aftbody structure. Fuselage frames are spaced at 50.8 cm (20 in) and the frame height constrained at 7.6 cm(3.0 in). The composite shell and secondary structure represents 77 percent of the total body mass.

Composite materials application to the other structural components (i.e. landing gear, tail, nacelle, air induction system) varies between 12 to 40 percent of the respective component mass.

The aircraft employs advanced controls concepts that are related to reducing structural mass (i.e. load alleviation, flutter mode control) as well as those concepts that are fundamental ingredients for viability of a slender, highly flexible arrow-wing configuration (i.e. ride quality control, elastic mode suppression, etc.).

The aircraft was resized (smaller) for a constant design payload-range and has a takeoff gross mass of 291,000 kg (642,000 lbm). The zero-fuel mass of the aircraft was reduced 15 percent from the near-term design with a commensurate reduction in flyaway cost. Composite materials account for approximately 51 percent of the total structural mass.

CONCLUDING REMARKS

The best structural approach for design of wing and fuselage primary structure of a Mach 2.7 arrow-wing configuration aircraft was determined considering near-term start-of-design technology. To accomplish this goal, a systematic multidisciplinary design-analysis process was used to assess the effects of the more important environmental considerations (e.g., thermal, airload, flutter) on the selection of the structural arrangement for a flexible arrow-wing configuration. Detail studies defined a near-term design airplane and its characteristics, and showed that the airplane was viable in terms of structural mass and flexibility. Supplemental studies provided airplane mass and performance trends as impacted by the application of structures technology postulated to be available beyond the near-term design time period. Significant improvement in fuel fraction for the constant mass airplane, with varying degree of advanced structures technology application, displayed performance improvements between 6 to 14 percent over the near-term design. Resizing the aircraft to the design payload-range goal resulted in a 15-percent reduction in empty mass and a commensurate reduction in flyaway cost.

A design methodology to cope with the various interactive parameters was established and provides guidance for future studies of this type. The study illustrated that the design analysis of large, flexible aircraft requires realistic aeroelastic evaluation based on finite-element analyses, and steady and unsteady aerodynamic loading determination. Static aeroelastic and flutter characteristics are important design considerations, and should be investigated early in the design cycle. Significant additional structure, over and above that required for strength, was required in the wing tip and the engine support rails to eliminate initial flutter deficiencies. Innovative application of computer graphics in the design process was demonstrated in the flutter optimization and low-speed handling quality time-history studies. These graphic systems were conducted using a relatively detailed analytical model of a supersonic cruise aircraft and showed the feasibility and cost-effectiveness in terms of decreasing manpower expenditures and desgin calendar time.

The study developed a realistic flexible model of an advanced arrow-wing supersonic cruise aircraft and has shown that the application of advanced structural panel concepts and unaxial reinforcement of the titanium spar caps with composite materials is a promising approach for a 1980 technology design. Although the proposed design concepts for the near-term design airplane satisfy the mission requirements, a considerable amount of research effort is required in (1) aerodynamics and configuration refinements, (2) experimental validation of the promising concepts, (3) advanced materials and fabrication development, including composites, and (4) continued development of advanced design-analysis methods to accelerate the design process. Included in the latter are automated data generation, integration of the design-analysis system and associated data management system, and interactive design analysis.

Also, as a part of the aircraft stability and control and performance investigations, the use of active controls was postulated. Further studies are needed concerning their use for stability augmentation and handling quality investigations particularly from the structural loads standpoint. Methodology for application of advanced control concepts to the airplane design must be pursued, addressing those concepts not only related to reducing structural mass (i.e. flutter mode control, load alleviation) and life enhancement (fatigue reduction) but also those concepts that are fundamental ingredients for viability of slender, highly flexible airframe configurations (i.e. ride quality, elastic mode suppression). In addressing these control concepts, focus must also be placed on improving the analytical representation of the transonic, nonlinear unsteady aerodynamic flow characteristics and the interaction of control surface and structural deformations under aerodynamic loading.

Of most concern and challenge for commercial application of advanced technology is the achievement of a systems reliability sufficiently high so that no failure would cause catastrophic loss of aircraft control or structural failure during the complete life of the aircraft. The results of the NASA development and flight evaluation program for both advanced composite primary structure and active controls application to the L-1011 will greatly enhance the advanced technology challenge and contribute immeasureably towards the development of a viable supersonic cruise aircraft.

APPENDIX

SYMBOLS AND ABBREVIATIONS

t	thickness
V_C	design cruise speed
V_D	design dive speed
Λ	sweep angle
B/AL	boron/aluminum
B/PI	boron/polyimide
EMS	elastic mode suppression
FMC	flutter mode control
GA	gust alleviation
MLC	maneuver load control
MLG	main landing gear
RQ	ride quality
RSS	relaxed static stability

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TABLE 1. CONCEPT EVALUATION SUMMARY

WING PRIMARY STRUCTURE CONCEPT[d]	MASS COMPARISON FOR BASELINE-SIZE AIRCRAFT[a]			COST[b] COMPARISON FOR OPTIMUM-SIZE AIRCRAFT[c]
	WING MASS		RELATIVE MASS	RELATIVE COST
	kg/m^2	lbm/ft^2		
(1) Chordwise stiffened - convex-beaded panels; B/PI reinforced spars; and aluminum brazed honeycomb core tip panels	39.99	8.19	1.00	1.00
(2) Chordwise stiffened - convex-beaded panels; B/PI reinforced spars	40.28	8.25	1.01	1.00
(3) Monocoque - aluminum brazed honeycomb core panels (mech. fastened)	41.70	8.54	1.04	1.07
(4) Monocoque - aluminum brazed honeycomb core panels (welded)	43.21	8.85	1.08	1.10
(5) Spanwise stiffened - hat-stiffened panels	47.26	9.68	1.18	1.09
(6) Chordwise stiffened - convex-beaded panels	47.85	9.80	1.20	1.11

(a) Gross takeoff mass = 340,000 kg (750,000 lbm)

(b) Direct operating cost for 25 x 10^9 ton-mile fleet mission

(c) Gross takeoff mass varies

(d) Each with a skin-stringer/frame fuselage structure

(e) Wing mass per unit planform area

TABLE 2. MASS ESTIMATES FOR NEAR-TERM DESIGN WING

ITEM	PLANFORM AREA		MASS	
	ft^2	m^2	lbm	kg
Variable Mass			50,432 (A)	22,876 (A)
Forward Box	4136.6	384.3	(20,580)	(9,335)
• Surfaces - convex beaded, chordwise stiffened			9,452	4,287
• Spars - including 522 lb (227 kg) composites			8,558	3,882
• Ribs			2,570	1,166
Aft Box	2132.4	198.1	(17,384)	(7,885)
• Surfaces - convex beaded, chordwise stiffened			7,302	3,312
• Spars - including 3,762 lb (1706 kg) composites			8,568	3,886
• Ribs			1,514	687
Transition - Aft Box to Tip Box			(1,380)	(626)
Tip Box	947	88.0	(11,088)	(5,030)
• Surfaces - brazed honeycomb sand., mech. fast.			9,435	4,280
• Spars			1,336	606
• Ribs			317	144

(A) Includes fail-safe penalty of 373 kg (822 lbm)

TABLE 2. MASS ESTIMATES FOR NEAR-TERM DESIGN WING (Continued)

ITEM	PLANFORM AREA		MASS	
	ft^2	m^2	lbm	kg
Fixed Mass			40,152	18,213
Leading Edge	1047	97.3	5,235	2,375
Trailing Edge	1941	180.3	4,888	2,217
Wing/Body Fairing	800	74.3	1,600	726
Leading Edge Flaps/Slats	133	12.4	1,130	512
Trailing Edge Flaps/ Flaperons	553	51.4	5,890	2,672
Ailerons	250	23.2	1,250	567
Spoilers	225	20.9	1,380	617
Main Landing Gear - Doors	484	45.0	2,904	1,317
Sup't Structure			3,750	1,701
B.L. 62 Ribs			1,430	649
B.L. 470 Ribs			700	318
Fin Attach Ribs (B.L. 602)			435	197
Rear Spar			3,400	1,542
Engine Support Structure			2,380	1,080
Fuel Bulkheads			3,800	1,724
Total Wing Mass			90,584	41,088

TABLE 3. MASS ESTIMATES FOR NEAR-TERM DESIGN FUSELAGE

ITEM	MASS	
	lbm	kg
<u>Shell Structure</u>	22,582 (A)	10,243 (A)
Skin	11,144	5,055
Stiffeners	7,921	3,593
Frames	3,517	1,595
<u>Fixed Mass (B)</u>	19,540 (B)	8,863 (B)
Nose and Flight Station	2,500	1,134
Nose Landing Gear Well	900	408
Windshield and Windows	1,680	762
Flooring and Supports	3,820	1,733
Doors and Mechanism	4,170	1,891
Underwing Fairing	1,870	848
Cargo Compartment Prov.	1,060	481
Wing to Body Frames and Fittings	1,500	680
Tail to Body Frames and Fittings	600	272
Prov. for Systems	740	336
Finish and Sealant	700	318
Total Fuselage Mass	42,122	19,106

(A) Includes fail-safe penalty of 650 kg (1,432 lbm)

(B) Includes composite material mass of 508 kg (1,120 lbm)

TABLE 4. ESTIMATED GROUP MASS STATEMENT
NEAR-TERM DESIGN AIRPLANE

ITEM	MASS	
	lbm	kg
Wing	90,584	41,088
Tail - Fin on Wing	2,800	1,270
Tail - Fin on Body	2,600	1,179
Tail - Horizontal	7,950	3,606
Body	42,122	19,106
Landing Gear - Nose	3,000	1,361
Landing Gear - Main	27,400	12,428
Air Induction	19,760	8,963
Nacelles	5,137	2,330
Propulsion - Turbofan Engine Inbd.	25,562	11,595
Propulsion - Turbofan Engine Outbd.	25,562	11,595
Propulsion - Systems	7,007	3,178
Surface Controls	8,500	3,856
Instruments	1,230	558
Hydraulics	5,700	2,585
Electrical	4,550	2,064
Avionics	1,900	862
Furnishing & Equipment	11,500	5,216
Environmental Control System	8,300	3,765
Tolerance & Equipment	1,980	898
Manufacturer Empty Mass	303,144	137,504
Std & Oper. Eq.	10,700	4,853
Operating Empty Mass (OEM)	313,844	142,357
Payload	49,000	22,226
Zero-Fuel Mass	362,844	164,583
Fuel	387,156	175,611
Taxi Mass	750,000	340,194

LEMAC = FS 1548.2 in. (39.32 m) MAC = 1351.06 in. (34.32 m)

X ARM = Distance from F.S. 0.

Fus. Nose at F.S. 279 in. (7.09 m)

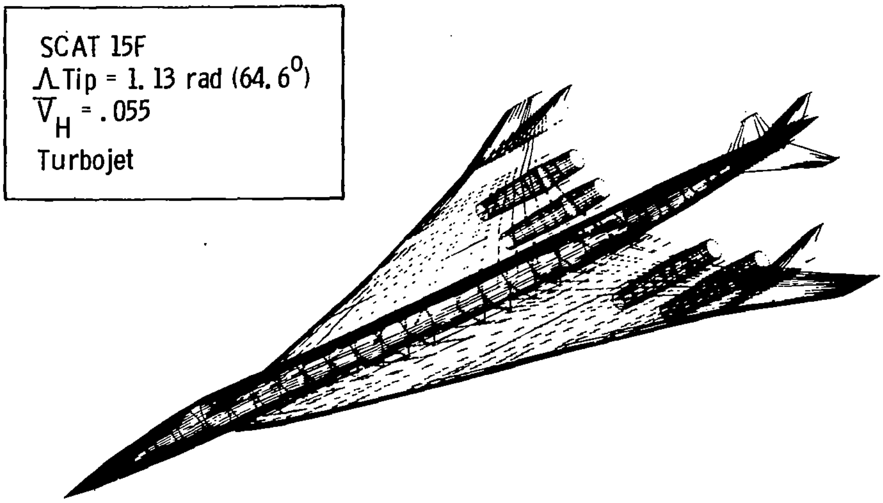


Figure 1.- Reference configuration.

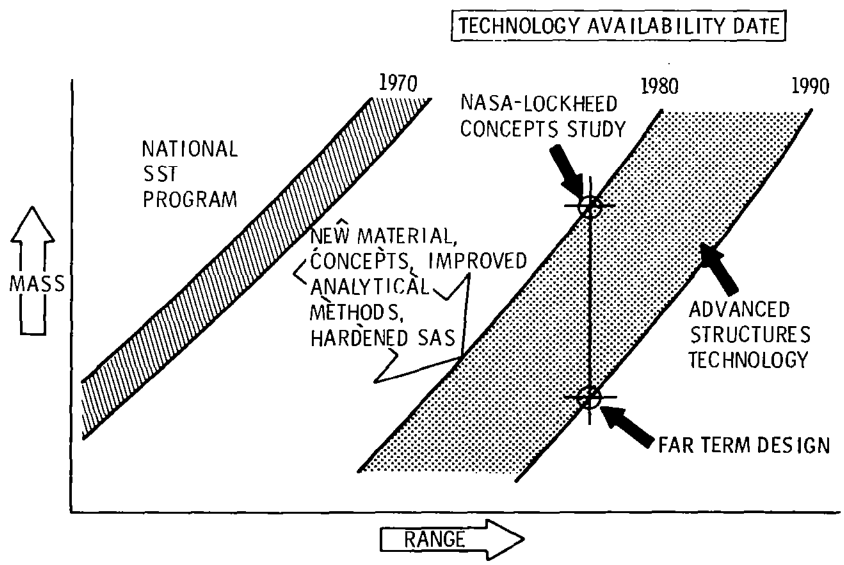


Figure 2.- Advanced technology trends.

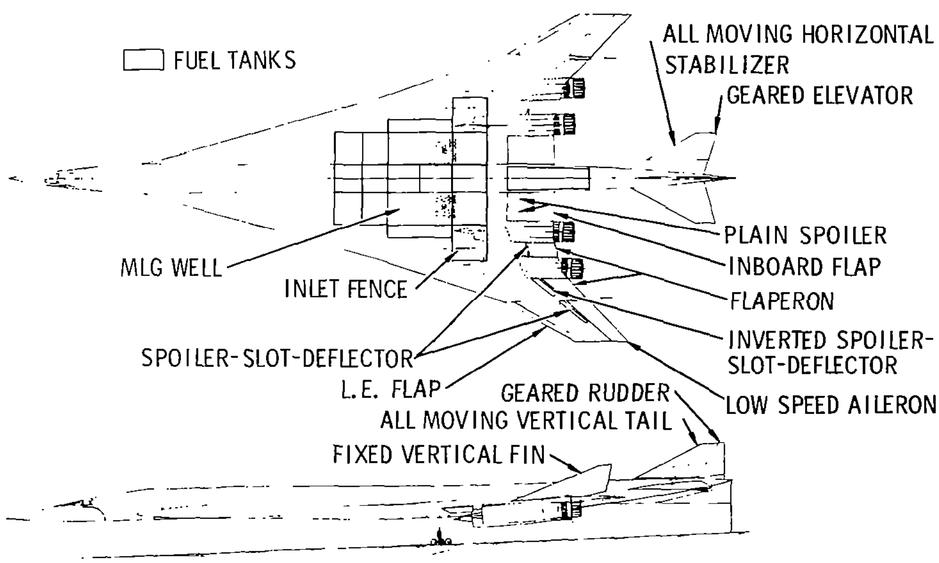
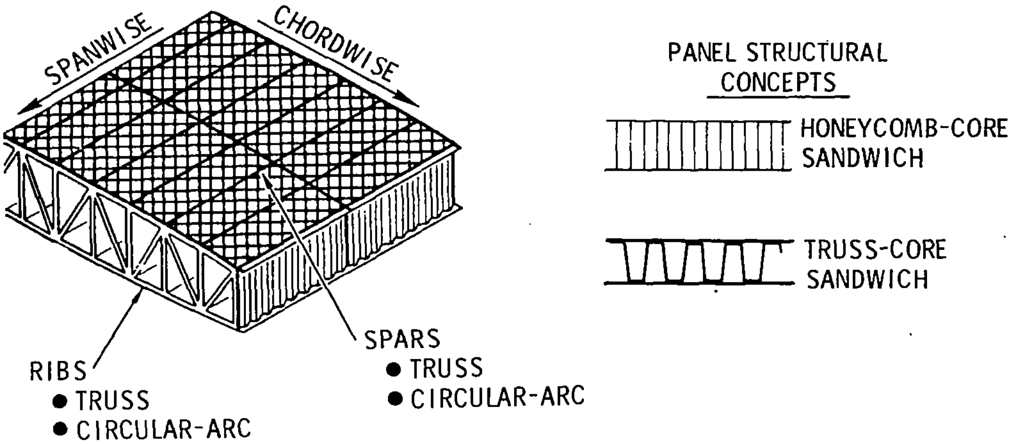
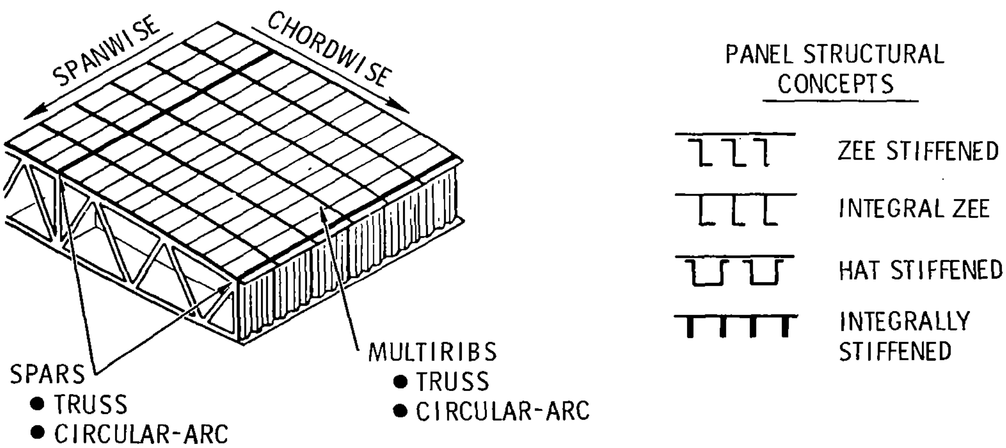


Figure 3.- Final arrangement.

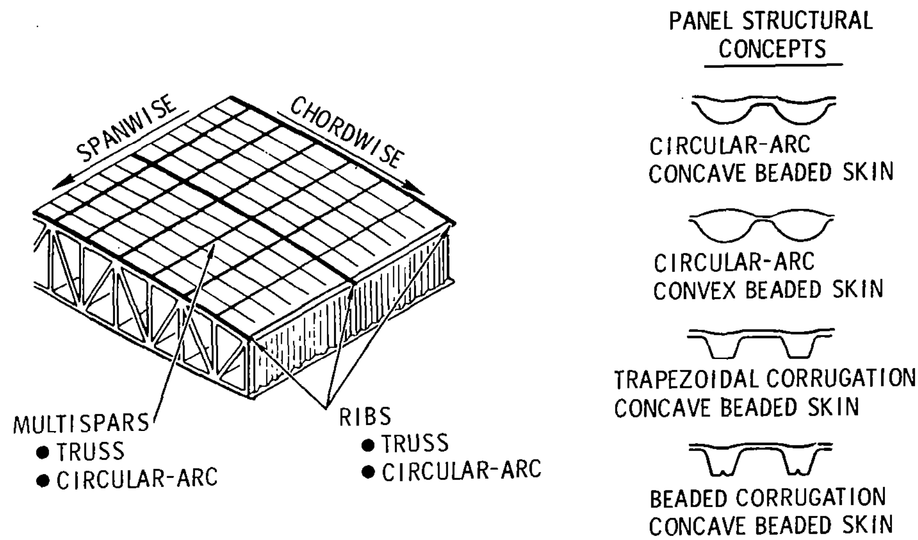


(a) Monocoque (biaxially stiffened).

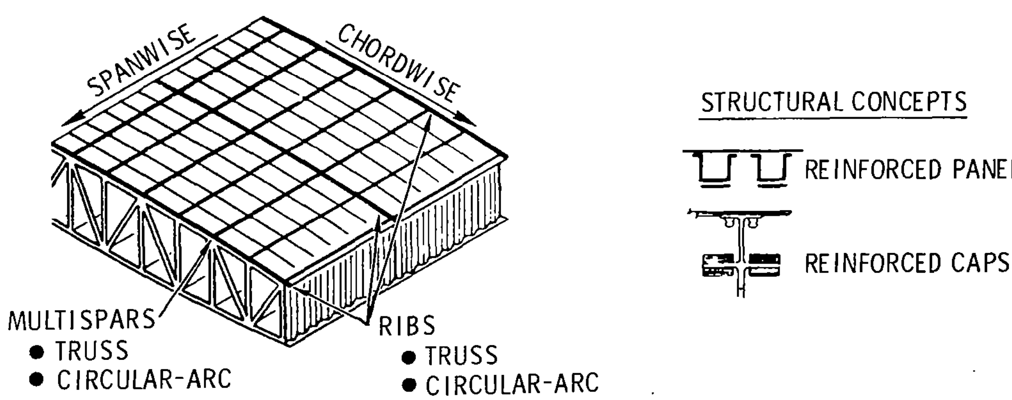


(b) Spanwise stiffened.

Figure 4.- Wing structural arrangement.



(c) Chordwise stiffened.



(d) Chordwise stiffened composite reinforced concepts.

Figure 4.- Concluded.

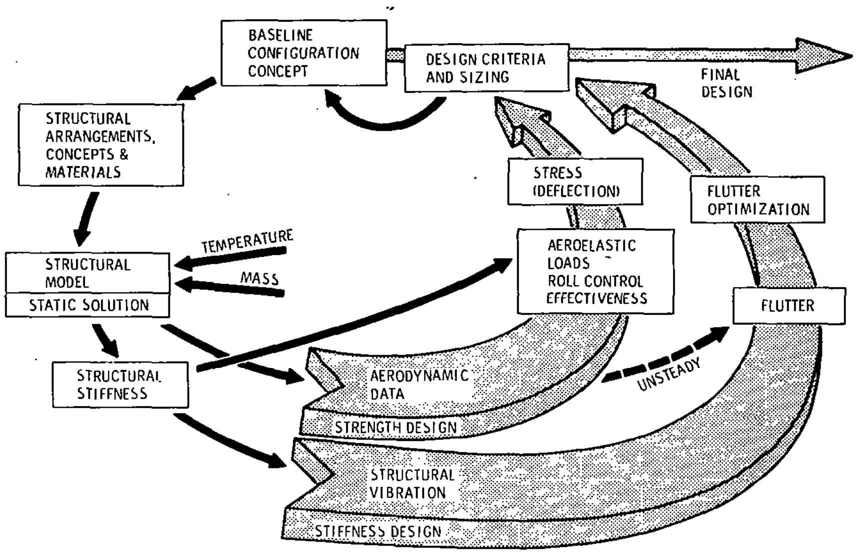


Figure 5.- Analytical design cycle.

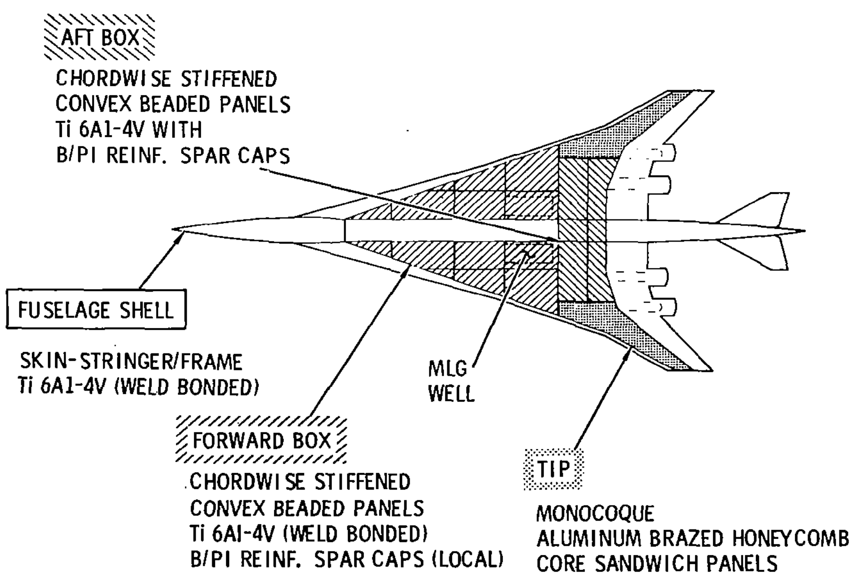


Figure 6.- Hybrid structural concept for near-term design.

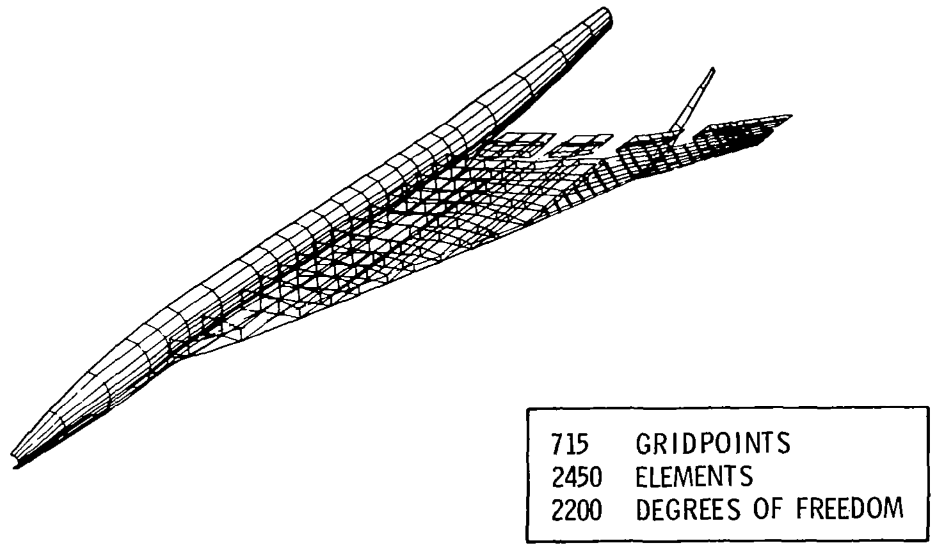


Figure 7.- Finite-element structural model.

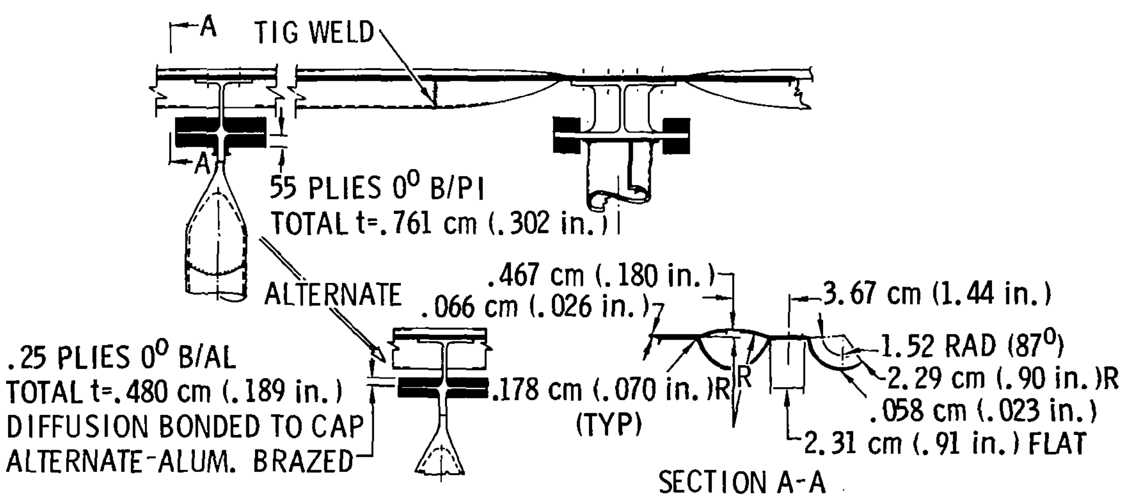


Figure 8.- Structural details for hybrid structural concepts.

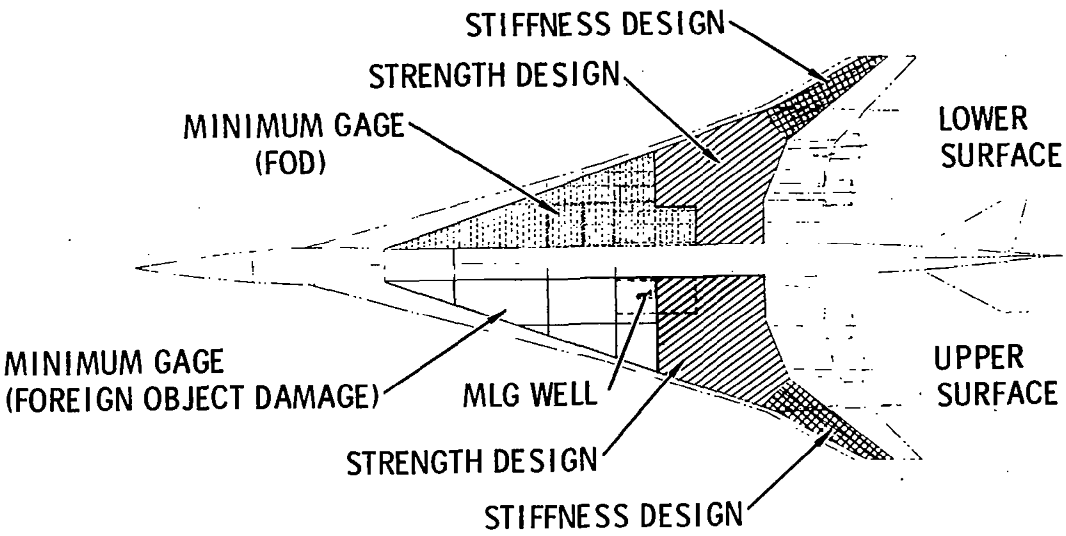


Figure 9.- Critical design requirements for wing.

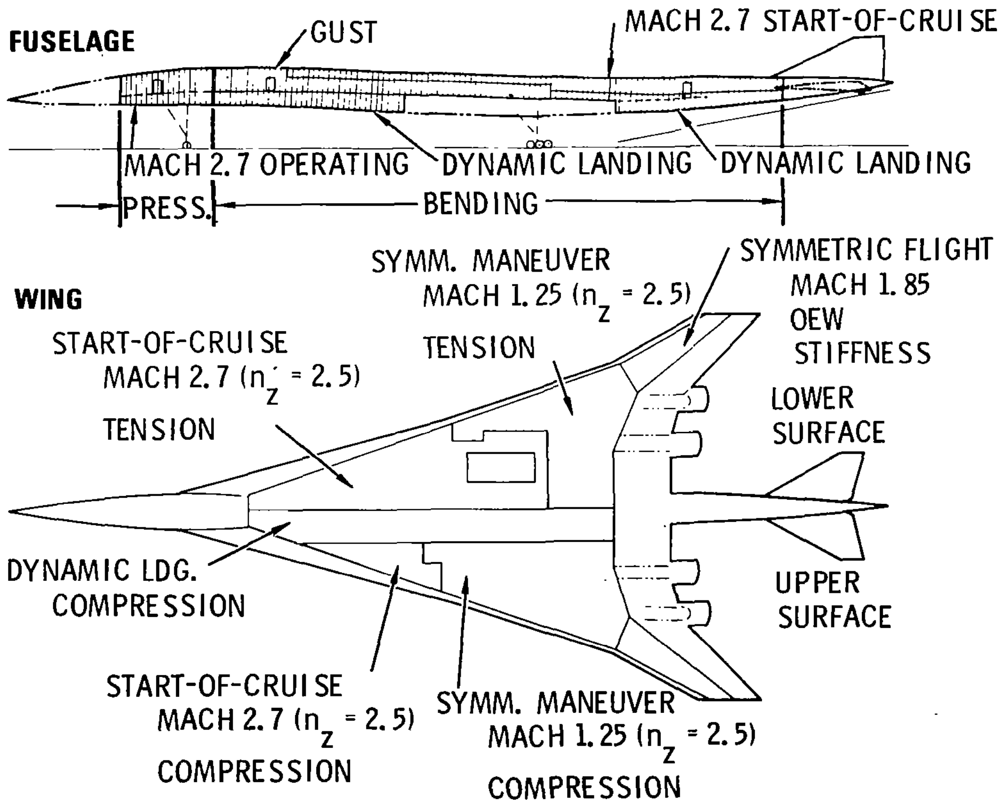


Figure 10.- Critical design conditions.

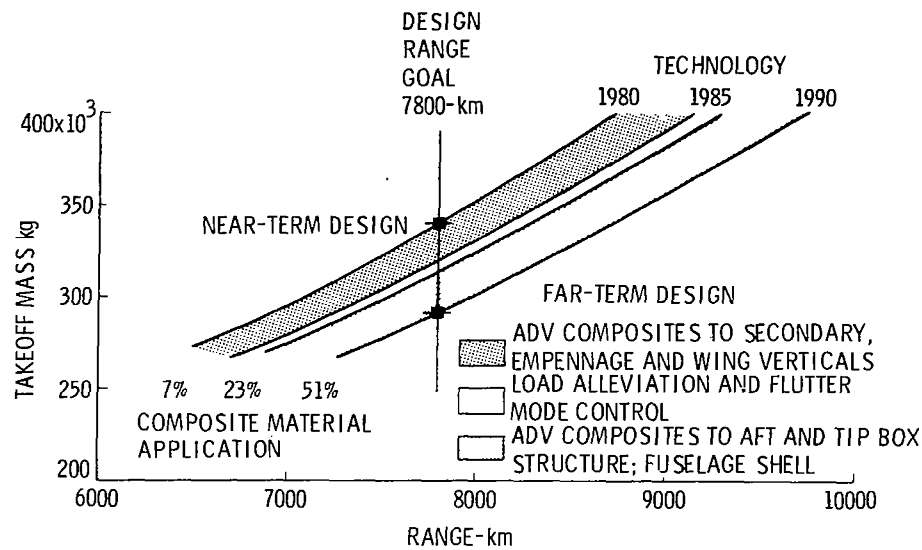


Figure 11.- Advanced structures technology trends.

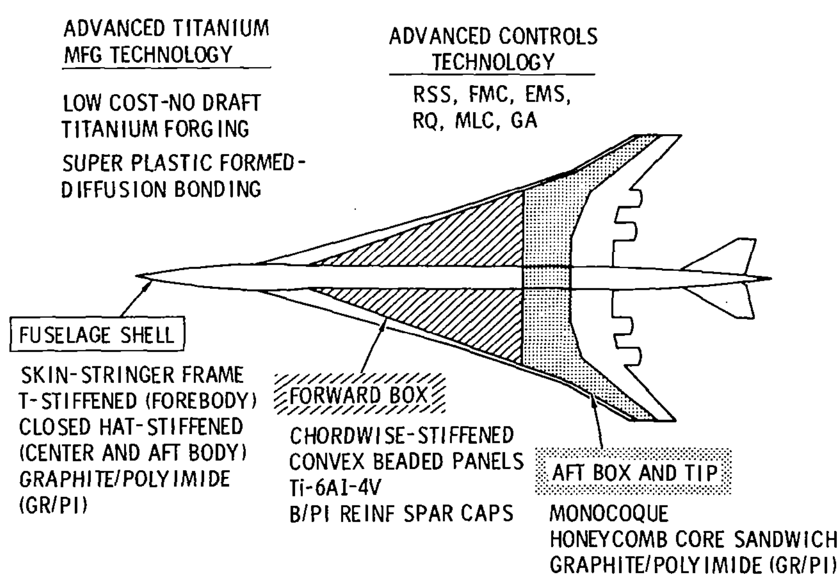


Figure 12.- Advanced hybrid structural approach - far term (1990 start of design).