Design Study of Structural Concepts for an Arrow-Wing Supersonic Cruise Aircraft

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An analytical study was performed to determine the best structural approach for design of primary wing and fuselage structure of a Mach number 2.7 arrow-wing supersonic cruise aircraft. Concepts were evaluated considering near-term start-of-design. Emphasis was placed on the complex interactions between thermal stress, static aeroelasticity, flutter, fatigue and fail-safe design, static and dynamic loads, and the effects of variations in structural arrangements, concepts, and materials on these interactions. Results indicate that a hybrid wing structure incorporating low-profile convex-beaded and honeycomb sandwich surface panels of titanium alloy 6A1-4V were the most efficient. The substructure includes titanium alloy spar caps reinforced with boron-polyimide composites. The fuselage shell consists of hat-stiffened skin and frame construction of titanium alloy 6A1-4V. This paper presents an executive summary of the study effort, and includes a discussion of the overall study logic, design philosophy, and interaction between the analytical methods for supersonic cruise aircraft design.

Introduction

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

The design of an economically viable advanced supersonic cruise aircraft requires reduced structural mass fractions attainable through application of new materials, advanced concepts, and design tools. 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 fraction.

The investigation now being reported was conducted to subject promising structural concepts to in-depth analyses, including the more important enviromental structural considerations that could affect the selection of the best structural approach for design of primary wing and fuselage structure of a given Mach 2.7 arrow-wing supersonic cruise aircraft, assuming a near-term start-of-design.

This paper presents the results of the study made by the Lockheed-California Company and discusses the design methodology and results of an in-depth structural design study of an supersonic cruise aircraft. (Results of a similar study conducted by the Boeing Commercial Airplane Company are presented in Ref. 1.) Extensive use of computer programs and their associated math models was required. Finite-element structural models (NASTRAN) were used to determine internal forces, displacements, and aeroelastic effects. Interactive computer graphic programs were used for flutter optimization and low-speed handling quality time-history studies on a relatively detailed analytical model of a supersonic cruise aircraft.

The results of this study have provided insight into future research requirements in the areas of advanced lightweight structural design concepts, analysis and design methodology, materials characterization, and aeroelastic evaluations of low-aspect-ratio, highly flexible aircraft. Furthermore, opportunities for structural mass reduction and needed research and technology to achieve the objectives of reduced structural mass fractions were identified. More detailed results of this study including recommended supersonic airframe technology items are given in Ref. 2.

Configuration

Reference Configuration

The reference configuration shown in Fig. 1 is a discrete wing-body airplane with a low wing that is continuous under the fuselage. The external shape of the airplane was defined at the design cruise lift coefficient by a computer card deck supplied by NASA. Refinements to the NASA concept in regards to 1) fuselage cross-section requirements to provide suitable passenger accommodations in terms of comfort, baggage stowage, cargo, and passenger services; and external contour deviations of the wing to permit landing gear stowage, 2) low-speed longitudinal characteristics of the arrow-wing, and 3) low-speed usable lift capabilities of the arrow-wing were examined and appropriate changes incorporated into the design.

Design Changes

The efficient use of all volume within the airplane was a primary objective in the configuration refinement studies performed. Related to this objective was the interior layout of the fuselage in the passenger accommodations area, for cruise drag places heavy emphasis on minimizing the fuselage cross-sectional area. From a passenger comfort standpoint, however, it was necessary to provide head room and have a cabin width which would allow for wide seats and sufficient aisle widths. Below-the-floor-volume was needed for cargo and baggage. These objectives were met by increasing the

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fuselage depth. The pressure shell radius remained essentially unchanged from the NASA concept. Appropriate decrement in airplane lift-to-drag (L/D = -0.10) was determined.

Also adopted was a landing gear concept which avoided deviations from the NASA supplied external contour. This main landing gear (MLG) is wing-stowed, forward retracting and has 12 tires per strut. The concept does not require a hump in the upper surface, thus avoiding a drag penalty and minimizing the complexity and weight of the wing structure.

Low-Speed Longitudinal Characteristics

The low-speed pitch-up characteristics of the arrow-wing were examined using a computer graphic arrangement that has the capability of displaying in real-time on a cathode ray tube, the longitudinal behavior of the airplane response to control disturbances which are applied and monitored by the operator.

Feasibility of using the horizontal tail as a pitch limiter to provide satisfactory longitudinal control while operating into the pitch-up region was investigated. Findings showed that if adequate control authority was provided, it was possible to provide automatic pitch limiting capability and good handling qualities. However, two requirements must be met; 1) there is a definite tail size and center-of-gravity relationship, and 2) the pitch limiter system must be fail-operative. On the basis of these considerations, a tail volume coefficient of 0.07 is the minimum that would yield an acceptable center-of-gravity range. The airplane balance should be set so that the center-of-gravity is at 55% MAC at the maximum landing weight.

Low-Speed Lift Capabilities

Configuration development studies explored application of leading and trailing edge devices with auxiliary trimming surfaces (canards and horizontal tail) to provide schemes for supplementing low-speed lift characteristics of the arrow-wing planform. 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 64.6° as defined by the NASA supplied data to a 60° tip sweep was made. This change reduced the demands of the longitudinal stability augmentation system and permitted a more aft center-of-gravity with the existing horizontal tail power.

Final Configuration

The general arrangement incorporating the design changes on the NASA concept is shown in Fig. 2 with the geometric characteristics defined in Table 1. The airplane has a design gross mass of 750,000 lb (340,000 kg). The overall length is 296.9 ft (90.5 m), and the wing span is 132.6 ft (40.4 m).

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 in Fig. 2. An all-moving horizontal stabilizer with a geared elevator is used for pitch control. For yaw control, a fuselage mounted all-moving vertical tail with a geared rudder is provided. The tail volumes for the horizontal stabilizer ($\bar{V}_H$) and vertical tail ($\bar{V}_V$) 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.

The wing-mounted main landing gear retracts into a well just outboard of the body. Four duct-burning turbofan engines, each within 89,600 lb (398,600 N) of uninstalled thrust, are mounted in under-wing pods having axisymmetric inlets and thrust reversers aft of the wing trailing edge. The engines are sized to provide a total thrust-to-airplane weight ratio of 0.36 at takeoff. An inlet fence was provided to prevent unstart because of mutual interference.

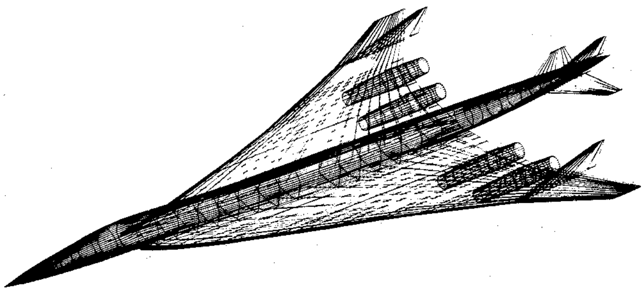

Fig. 1 Reference configuration.

The major portion of the lower fuselage is used for fuel and baggage stowage, with baggage and other requirements establishing the forward limit of fuel stowage. Forward of the fuel stowage area, the wing does not extend through the fuselage.

The tank arrangement shown in Fig. 2 provides for a fuel storage capacity of 393,600 lb (178,500 kg). Based on previous studies relating to fuel containment and management requirements for supersonic cruise aircraft, it was elected to stow a significant portion of the total fuel within the wing center section. The 16-tank system was designed to take advantage of the "protected-volume" of approximately 43% of the total storage capacity. In this location, the upper surface was exposed to the cooled and controlled environment of the fuselage cabin; whereas the wing lower surface was shielded from the outside airstream by a fairing extending below and separated from the lower surface.

Fuel management scheduling for airplane center-of-gravity control was specifically planned to maximize the available heat sink capacity of the fuel by emptying the exposed outboard tanks as early as possible in the flight. Additional considerations included fuel usage to permit the aircraft to cruise with a minimum trim drag penalty. The landing and reserve fuel was located in the protected fuselage area.

Structural Design Concepts

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

Design and manufacturing concepts studies 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 concept to provide consistency between manufacturing/design studies and analyses.

These studies examined the fabrication down to the smallest subcomponent level, and involved the design of structural concepts that represented both structural efficiency and applicability to advanced fabrication techniques. Candidate materials included both metallic and composite material systems. Alpha-Beta (Ti-6A1-4V) and Beta (Beta C) titaninum 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 (graphite-polyimide, boron-polyimide) and metallic (boron-aluminum) matrix 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

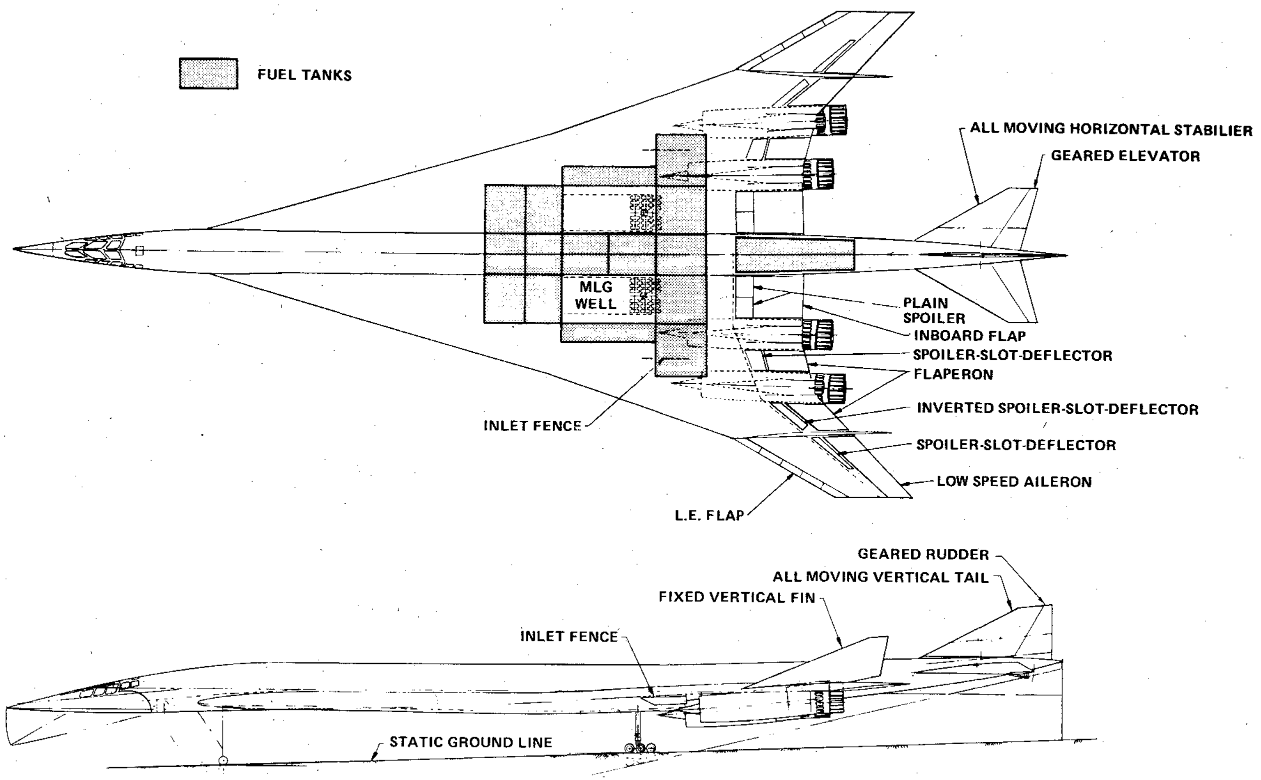

Fig. 2 Final arrangement.

unidirectional reinforcing of members carrying primary axial loads, such as stringers, spar caps, rib caps, and stiffeners of wing panel design.

Wing Structural Concepts

The primary load carrying structural concepts for the wing are shown in Fig. 3.

Monocoque

Monocoque contruction (Fig. 3a) consists of biaxially stiffened panels which support the principal load in both the span and chord direction. The substructure arrangement consists of both multirib and multispar designs. The monocoque construction has a smooth skin that results in minimum aerodynamic drag; however, thermal stresses are absorbed by the primary structural elements with minimal relief. Biaxial loading results in reduced fatigue allowables, yet criticality of other design parameters often controls minimum mass structural designs.

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

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

Semimonocoque

The two types of semimonocoque concepts are 1) panels supporting loads in the spanwise direction (Fig. 3b), and 2) panels supporting loads in the chordwise direction (Fig. 3c). Both have 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 panels cannot support biaxial loads. Either the spar cap or rib cap must have sufficient area to support 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. Smooth skins are required for aerodynamic performance.

The chordwise-stiffened arrangement is essentially a multispar structure with widely spaced ribs. Submerged spar caps are provided except at panel closeouts and at fuel tank bulkheads. The submerged caps afford reduced temperatures and increased allowable stresses (strength and fatigue). The surface panel concepts for this arrangement have stiffening elements oriented in the chordwise direction. Structurally efficient beaded-skin designs were explored. 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. These 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 expansion (deformation) in the spanwise direction.

Selective reinforcement of the basic metallic structure (Fig. 3d) is 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 adopted for the application of composite reinforcing. The many unique design features of the chordwise-stiffened arrangement are retained. In addition, structurally efficient, multielement (fail-safe) composite reinforced spar cap designs are employed to transmit the spanwise bending moments as concentrated axial loads with minimum mass.

Fuselage Structural Concepts

The primary load carrying structural concepts for fuselage design are categorized as 1) sandwich shell and 2) skin-stringer and frame shell.

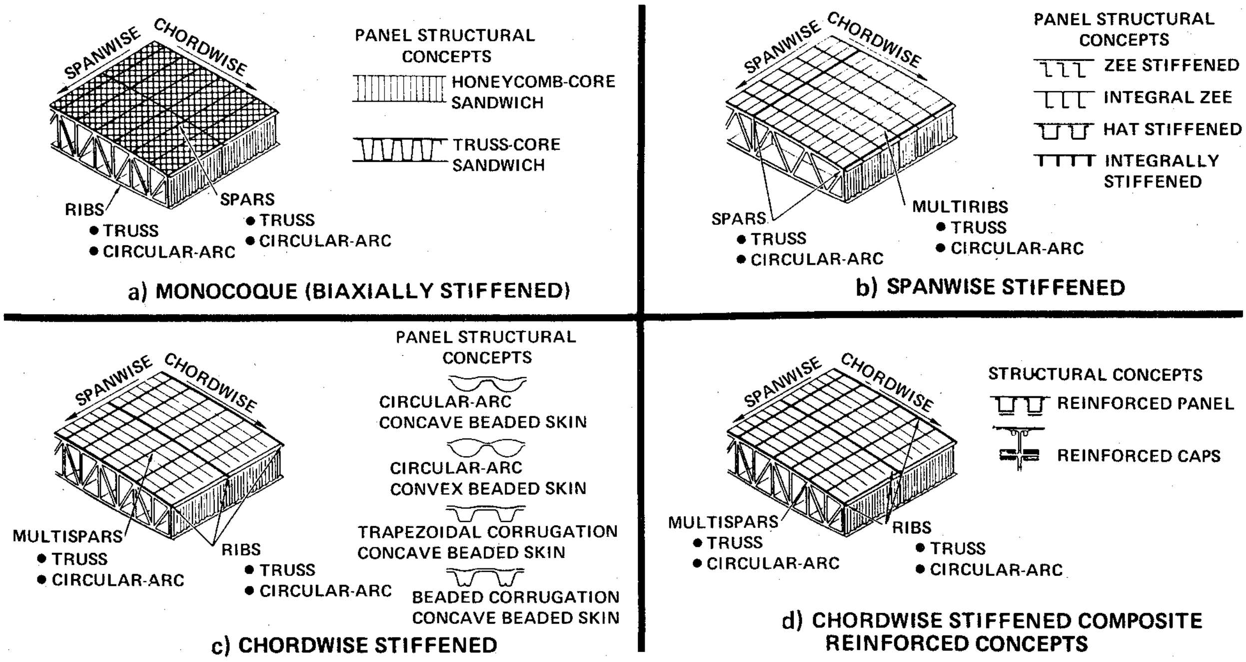

Fig. 3 Wing structural concepts.

Sandwich Shell

The sandwich shell design was thought to have a potential for 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 weight 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.

Skin-Stringer and Frame Shell

The basic structural arrangement for this design is a uniaxial stiffened structure of skin and stringers with closely spaced supporting frames. The stringer configurations with the potential of achieving minimum mass are the zee-stiffened and the open- and closed-hat sections. All of these stiffener concepts contain flat elements which are amenable to composite reinforcing. Supporting frames that merit consideration were both the fixed and floating type. The joining methods evaluated for this arrangement include mechanical fastening, welding, and bonding.

Structural Criteria

Evaluation of the wing and fuselage primary structure concepts was based on an aircraft with an economic service life of 15 yr and a service life of 50,000 flight hours, with the environment determined from a design flight profile for an international mission. The international mission is approximately 3.4 hr in duration; three-quarters of that time (2.5 hr) is at Mach 2.62 (Hot Day) cruise (Fig. 4).

Design Speeds

Design equivalent airspeeds (Fig. 5) are selected to provide an operational envelope compatible with the desired flight profiles (Fig. 4). The structural design cruise speed (V_C) was selected as the planned operating speed in climb, cruise, and descent. The design dive speed (V_D) was selected to provide a margin of safety for the inadvertent large excursions in excess of operating speed.

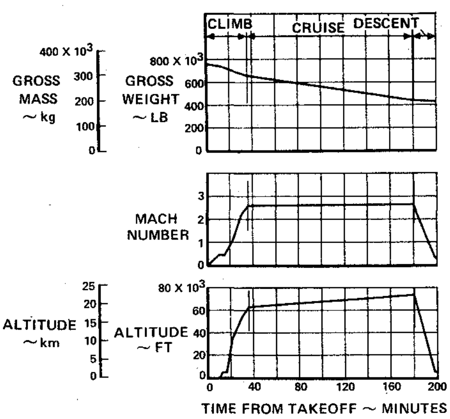

Fig. 4 Design flight profile – international mission (Mach 2.62 cruise-hot day).

Design Mass

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

Maneuvering Flight Criteria

Maneuver loads analyses are based on solution of the airplane equations of motion for pilot-induced maneuvers, considering altitudes between sea level and 70,000 ft (21.3 km), all speeds, gross mass, and center-of-gravity limits. Except where limited by a maximum usable normal force coefficient or by available longitudinal controls deflections, the limit load factors (n_z) are as follows: 1) positive maneuvers: $n_z = 2.5$ at all design speeds; 2) negative maneuvers: $n_z = 1.0$ up to V_C and varies linearly to zero at V_D; 3) rolling maneuver entry load factors: a) upper limit: $n_z = 1.67$ at all designs speeds; b) lower limit: $n_z = 0$ up to V_C and varies linearly up to $+1.0$ at V_D.

Fatigue and Fail-Safe Loads Criteria Fatigue Spectrum

Fatigue analysis is based upon a representative loading spectrum[2] and provides a moderately conservative represen-

Table 1 Airplane dimensional data

WING		
TOTAL AREA (S_W)	10923 ft^2	1014.69 m^2
REFERENCE AREA (S_{REF})	10500 ft^2	975.45 m^2
ASPECT RATIO (AR)	1.607	
TAPER RATIO (λ)	0.1135	
SPAN (b)	1590.0 in	40.386 m
ROOT CHORD (C_r)	2195.5 in	55.766 m
TIP CHORD (C_t)	249.2 in	6.330 m
MEAN AERODYNAMIC CHORD ($\bar{C}$)	1351.1 in	34.317 m
L.E. SWEEP (Λ_{LE})		
(TO BL 391.2)	74 deg	1.292 rad
(TO BL 600)	70.84 deg	1.236 rad
(TO TIP)	60 deg	1.047 rad
FUSELAGE		
LENGTH	3444.0 in	87.5 m
WIDTH	135.0 in	3.4 m
DEPTH	166.0 in	4.2 m

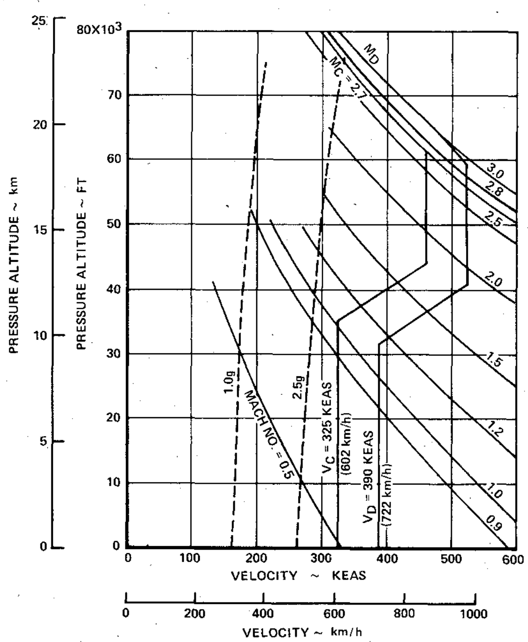

Fig. 5 Structural design speeds.

tation of a loading history for supersonic cruise aircraft. The reference stress level and oscillatory flight loads were defined on the spectrum and included representative tensile thermal stress increment and simulation of ground loadings.

Fatigue Criteria

The basic fatigue criterion is to provide a structure with a service life of 50,000 flight hours. Appropriate multiplying factors are applied to the design life for use in establishing allowable design stresses. For structure subjected to a spectrum loading, the allowable stresses are selected using a factor of twice the life of 50,000 hr. For areas of the fuselage subjected to constant amplitude loading the allowable stresses are selected for 200,000 flight hours of service (50,000 × 4).

Fail-Safe Criteria

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

Minimum Gage Criteria

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

Material Allowable

The allowable material stress was compared to stresses based on thermal and airloads, predicted structural temperatures, and appropriate fatigue spectrum. Limit thermal and airload stresses were not allowed to exceed the material yield strength or two-thirds of the material ultimate strength (the lower for the appropriate temperature). For compression members, the ultimate allowable stress was considered to be the critical buckling stress of the members supporting the primary loads.

Design Methodology and Analyses

The structural evaluation was conducted using a systematic multidisciplinary analysis procedure considering the effects of the complete environment on the structural integrity of the aircraft. The flow diagram of the design cycle from initial definition of the airplane configuration to the establishment of the final design is presented in Fig. 6. Because of the complex nature of this design cycle, extensive use of computer programs and their associated math models was required. These calculations were accomplished using Lockheed's Structural Design Analysis System, which has the combined program capabilities of the NASTRAN System and the Lockheed FAMAS System.

The Lockheed FAMAS System contains a very extensive matrix algebra and manipulation system, and a large family of functional modules for aerodynamic loads, structural response, and flutter analysis. In addition, this system has completely compatible matrix input/output capability within all its programs. The NASTRAN system includes its finite-element analysis capabilities in statics, dynamics and structural stability. These two systems are connected with an interface module which permits transit from one system to the other. In this fashion all of the design load analysis capabilities of the FAMAS system are joined with the finite-element analysis capabilities of NASTRAN. Similarly, NASTRAN stiffness or structural flexibility matrices, vibration mode vectors, etc., can be used in direct link with the flutter analysis system in FAMAS, as well as the aeroelastic loads calculations.

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 were 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 weight-strength 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 aforemen-

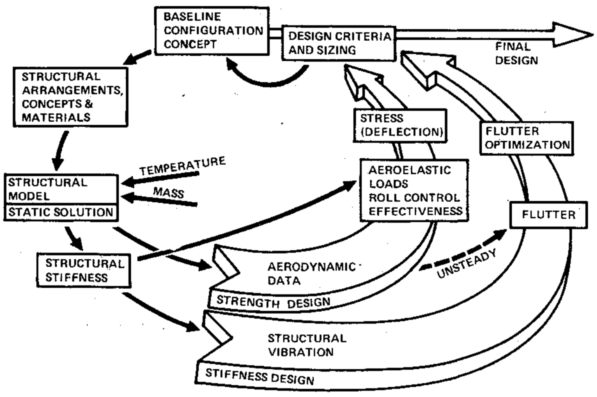

Fig. 6 Analytical design cycle.

tioned 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 weight-strength 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 arrangement 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[3] to 188 and 178 degrees of freedom, symmetric and antisymmetric, respectively. The inertia matrices were formed for two airplane mass conditions: the operating weight empty (OWE), and the full-fuel and full-payload (FFFP). These conditions represent the extremes of minimum and maximum airplane mass. No intermediate mass conditions were examined. Flutter analyses were conducted at selected Mach numbers for both symmetric and antisymmetric boundary conditions. An interactive computer graphics system, Graphics Flutter Analysis Methods (GFAM)[4] was used to optimize the placement of mass and/or stiffness to correct any flutter deficiencies. 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 a short time, of the amount and location of required additional structural material considering practical design/manufacturing constraints. 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 and 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 Ref. 2.

Concept Selection

The wing primary-structure design concepts were ranked (Table 2) on the basis of relative mass (constant mass airplane). When these primary-stucture 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 rise 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. 7) 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 is of closed hat-stiffener design with supporting frames.

Engineering Design/Analyses

Using the hybrid design, a more detailed three-dimensional finite-element model was developed and used as the basis for the final structural analyses. The finite-element model (Fig. 8) 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 then were determined at the Mach numbers of 0.60, 0.90, and 1.85 to assess the additional stiffness requirements to correct flutter deficiencies. Relative to the flutter-speed requirements defined from the operating envelope (Fig. 5), all Mach numbers investigated have adequate flutter margins of safety.

Reversal speeds and FAR requirements were compared for both the normal scheduled surface combinations and for selected fail-safe conditions which involved loss of a surface which has the most adverse effect on roll-control reversal speed. In all cases, the final design airplane exceeds the specified requirements.

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

To assess the results of the structural modeling techniques employed in the study and to provide insight for future research studies, an investigation was conducted to compare the accuracy of the two-dimensional and three-dimensional models. One of the two-dimensional models was updated to reflect the flexibilities of the final design three-dimensional model and a structural influence coefficients run was conducted. In general, the results of this run indicate a more flexible wing and fuselage for the two-dimensional model with the wing rear beam influence coefficients being approximately 10% higher in the wing tip region than the corresponding values of the three-dimensional model.

Table 2 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	lb/ft^2		
CHORDWISE STIFFENED - CONVEX-BEADED PANELS; B/PI REINFORCED SPARS; AND ALUMINUM BRAZED HONEYCOMB CORE TIP PANELS	39.99	8.19	1.00	1.00
CHORDWISE STIFFENED - CONVEX-BEADED PANELS; B/PI REINFORCED SPARS	40.28	8.25	1.01	1.00
MONOCOQUE - ALUMINUM BRAZED HONEYCOMB CORE PANELS (MECH. FASTENED)	41.70	8.54	1.04	1.07
MONOCOQUE - ALUMINUM BRAZED HONEYCOMB CORE PANELS (WELDED)	43.21	8.85	1.08	1.10
SPANWISE STIFFENED - HAT - STIFFENED PANELS	47.26	9.68	1.18	1.09
CHORDWISE STIFFENED - CONVEX-BEADED PANELS	47.85	9.80	1.20	1.11

(a) GROSS TAKEOFF MASS = 750,000 LB (340,000-kg)
(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

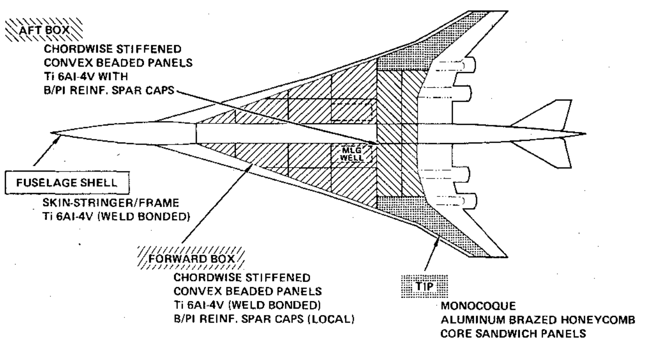

Fig. 7 Hybrid structral concepts for engineering design/analysis.

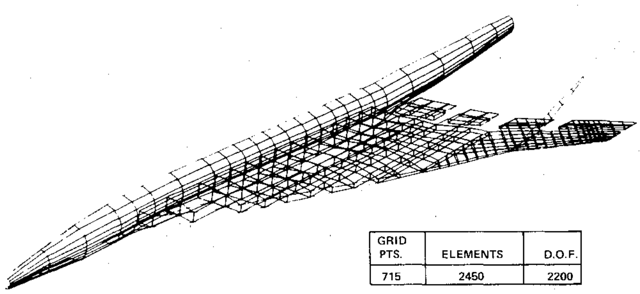

Fig. 8 Three-dimensional finite-element model.

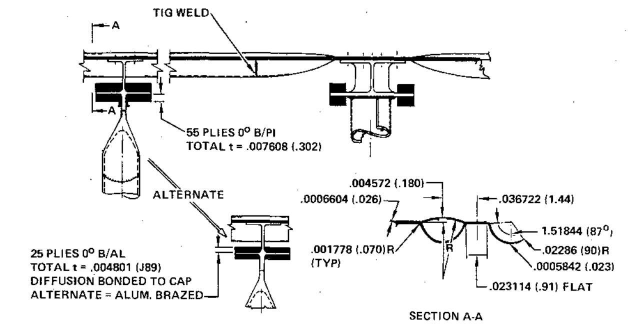

Fig. 9 Structural details for hybrid structural concepts.

Wing Structure Design

The design details for a specific chordwise-stiffened surface panel and substructure are shown in Fig. 9. 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. Also, the surface design alleviates thermal stresses and reduces heat transfer to the fuel, in comparison with a flat skin, since only a portion of the fuel is in direct contact with the wing external skin.

Weldbonding is the basic method proposed for joining the inner and outer skins of the surface assembly. The manufacturing limits for the surface panels were held to 15×35 ft (4.57×10.67 m). The length limit was based on tooling considerations for hot vacuum forming of the skins; whereas the width limit was based on postulated size of spotwelding equipment.

In locating wing spars in the chordwise-stiffened wing area, a minimum spacing of 21 in. (0.533 m) was maintained between constraints such as fuel tank boundaries. Wing rib spacing was a nominal 60 in. (1.524 m) 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 wall. 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.

Monocoque surfaces were used in the stiffness critical wing tip box. 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 lightweight 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 68 in. $\times$ 40ft (1.73×12.19 m) for brazed surfaces was postulated as a guide after consulation with Aeronoca. The panel configurations were based on the design philosophy that all or some panels of the upper surface are attached with screws and are removable for inspection and maintenance purpose.

The flexibility of the aluminum braze process was exploited by incorporating crack stoppers and panel edge doublers in the surface panel brazements. Also, the capability of tapering the panel thickness was utilized in the joint between the chordwise and monocoque surface areas. In the joint area, where transition in arrangement was made, 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 Design

The fuselage structural arrangement includes 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.

Welding 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 weld-bonding 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 obtained enhanced fatigue properties. The size of fuselage skin panel assemblies has been limited to 15×50ft (4.57×15.24 m); 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 and centerlines 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 final design structure are shown in Figs. 10 and 11. In Fig. 10, the upper and lower surface of the wing are divided into three distinct zones according to the three design requirements that dictated structural sizes. The tip structure was stiffness critical and sized to meet the flutter requirements. The aft box and selected regions of the forward box were strength-designed to transmit the wing spanwise and chordwise bending moments and shears. In general, 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 design conditions which displayed the maximum inplane surface panel loads are presented in Fig. 11, an exception being the tip structure which was stiffness critical for the Mach 1.85 condition. Although the start-of-cruise condition (Mach 2.7) has the highest value of inplane loading, combination with the appropriate pressure loads results in the symmetric maneuver condition at Mach 1.25 designing the wing in the forward and aft box region.

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 in the figure, 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 down-bending which occurs at varying times from main landing gear impact.

The supersonic cruise airfcraft displayed critical loads at Mach numbers wherein the structural temperatures do not influence the design appreciably. Although a major area of the wing lower surface is impacted by the thermal environment, analysis of surface panels and substructure using the applicable load-temperature environment results in the symmetric maneuver condition at Mach 1.25 as the critical design condition. The resulting designs, however, were very similar in geometry and structural mass. 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 centerbody and aftbody shell were critical for the imposed thermal environment combined with the appropriate airload effect.

Final Design Airplane Mass Estimates

The mass properties for the final design airplane were determined as shown in Table 3 as an estimated group mass statement. The data reflect a fixed size aircraft with a takeoff gross mass of 750,000 lb (340,000 kg) and payload of 49,000 lb (22,000 kg).

The study focused on the two largest structural mass items, the wing and body, which amounts to 90,584 lb (41,088 kg) and 42,122 lb (19,106 kg), respectively. These mass items represent 66% of the total structural mass and about 18% of the aircraft taxi mass. A more detailed look at the mass distribution of the largest component, the wing, indicates that 50,432 lb (22,876 kg) is attributed to the primary structural box (i.e. forward, aft, tip, and transition structure). The major ribs, rear spar, fuel bulkheads, and engine support structure accounts for 12,145 lb (5,510 kg). The leading-edge and trailing-edge structure, spoilers, wing/body fairing, and main landing gear doors and support structure accounts for the remaining 28,007 lb (12,702 kg).

Conclusions

An analytical study was performed to determined the best structural approach for design of primary wing and fuselage

Table 3 Estimated group mass statement final design airplane

ITEM	MASS (lbs)	(kg)
WING	90,584	41,088
TAIL – FINS 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 – T/F ENGINE INBD.	25,562	11,595
PROPULSION – T/F 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
ECS	8,300	3,765
TOLERANCE & OPTIONS	1,980	898
MEW	303,144	137,504
STD & OPER. EQ.	10,700	4,853
OEW	313,844	142,357
PAYLOAD	49,000	22,226
ZFW	362,844	164,583
FUEL	387,156	175,611
TAXI MASS	750,000	340,194

LEMAC = FS 1548.2 MAC = 1351.06 IN. (34.32M)

X ARM = DISTANCE FROM FUSELAGE STATION (F.S.) 0

FUS. NOSE AT F.S. 279

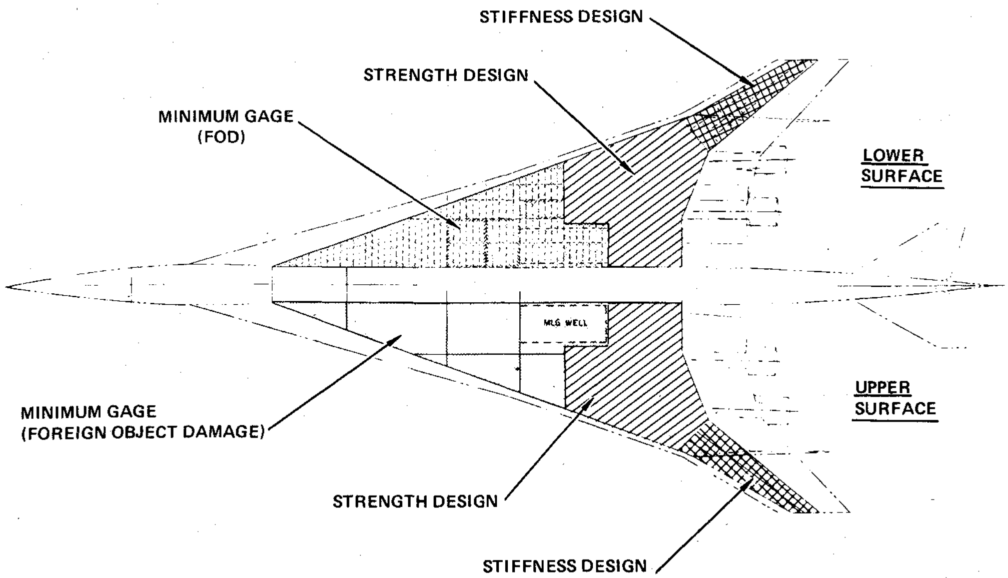

Fig. 10 Critical design requirements for final design wing.

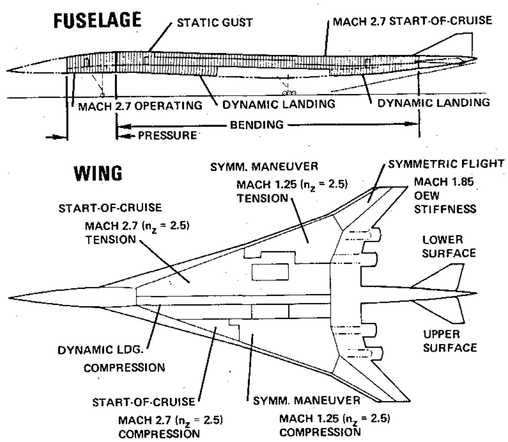

Fig. 11 Critical wing design conditions for final design airplane.

structure of a Mach 2.7 arrow-wing supersonic cruise aircraft considering near-term start-of-design technology. A systematic multidisciplinary analysis was conducted to assess the effects of the complete enviroment on the structural integrity of the aircraft. Verification studies defined the final design airplane and its characteristics and showed that the airplane was viable in terms of structural mass and flexibility.

A design methodology to cope with the various interactive parameters was established and provides guidance for future studies of this type. Extensive use of computer programs and their associated math models were required. These calculations were accomplished using the combined program capabilities of the NASTRAN and the Lockheed FAMAS Systems. The innovative application of interactive computer graphics systems to the design process was demonstrated. The use of the graphics systems as a computing tool for structural optimization to satisfy flutter requirements and low-speed handling quality time-history studies on a relatively detailed analytical model of a supersonic cruise aircraft was feasible and cost-effective.

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 uniaxial refinforcement of the primary structure is a promising approach for a 1980 technology design. Although the proposed design concepts for the final design airplane satisfy the mission requirements, a considerable amount of research effort is required in 1) aerodynamic and configuration refinements, 2) experimental validation of the promising design concepts, 3) advanced materials and fabrication technology development (including composites), and 4) continued development of analytical methods to accelerate the design process to attain a commercially successful supersonic cruise aircraft.

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