STRUCTURAL DESIGN OF SUPERSONIC CRUISE AIRCRAFT

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SUMMARY

The supersonic cruise aircraft structures efforts have been supported by NASA-contracted studies, reference 1, and McDonnell Douglas-funded research and testing. The major efforts leading to an efficient structural design include (a) the analysis methods used to improve the structural model optimization and compare the structural concepts; (b) the analysis and description of the fail-safe, crack growth, and residual strength studies and tests; (c) baseline structural trade studies to determine optimum structural weights including effects of geometry changes, strength, fail-safety, aeroelastics and flutter; (d) comparison of British, French, and United States aluminum alloys with 6AL-4V annealed titanium in structural efficiency after 70 000 hours at temperature; (e) the study of three structural models for aircraft at 2.0 Mach, 2.2 Mach, and 2.4 Mach cruise speeds; (f) the study of many structural concepts to determine their weight efficiencies; and (g) the determination of the requirements for large-scale structural development testing.

INTRODUCTION

The highlights of the McDonnell Douglas structural study results are presented herein. This includes extensive Company-funded efforts to improve the analytical methodology for use in preliminary design activities. The system studies represent work supported by NASA during the 1973 to 1976 period and Company-funded efforts for research and development for a longer period. An arrow wing has presented a structural design challenge for over a decade. Early studies indicated large weight penalties to solve the aeroelastic and flutter problems. The thin wing, high aspect ratio, thermal stresses and thermal degradation of the materials from the long life at temperature were all important considerations in the design selection process that lead to the use of high percentages of titanium in spite of the higher material and fabrication costs. Direct operating cost studies substantiated the titanium selections. The strength, fail-safe, aeroelastic, and flutter optimization methods developed have enabled McDonnell Douglas to achieve cost effective structure for the 2.2 Mach number selected for the baseline design. With a substantial structural development program in titanium and to a lesser degree in aluminum, the structural integrity can be insured for an early design go-ahead. A longer development period is required for the introduction of composites because minimum time temperature experience is available.

STRUCTURAL ANALYSIS METHODOLOGY

The structural optimization process used by McDonnell Douglas for the supersonic cruise vehicle has a long history of development. In the late 1940's, McDonnell Douglas developed a matrix method of structural analysis (see reference 2). Continued development, with some help from the Air Force Flight Dynamics Laboratory, reference 3, has enabled McDonnell Douglas to create FORMAT, a Fortran Matrix Abstraction Technique. This system has been the foundation for structural analysis development. In the early 1970's at McDonnell Douglas it was formally recognized that the Advanced Design needs are different than Production Design. Drastic reductions in elapsed time for obtaining accurate structural information in the Advanced Design of a supersonic cruise vehicle were required. Improved methods were developed which resulted in the following operational programs and procedures for the supersonic cruise aircraft design activities today:

- A structural optimization program, reference 4, has been developed which uses many options to speed up the resizing process and obtain accurate results. By using combined allowables with the best element representation, the weight for strength is more accurately predicted early in the design phase.

  The various types of structural models used for the supersonic cruise vehicle are shown in table I. It can be noted that a 24.3 percent improvement in strength weights result from the improved methods that more accurately represent the structure. The simple bar and panel elements are shown in figure 1. The more sophisticated upper wing and fuselage membrane elements substituted for some of the shear panels are shown in figure 2. The lower wing panels are also used but do not show in this figure. This membrane analyses can be improved by using interaction formulas that account for the combination of the biaxial, shear, thermal degradation, thermal stresses, and size-dependent allowables. This is now done.

- In addition, a sub-program for sandwich panels is used for the wing surfaces that accounts for the local and general stability of the core and sandwich and optimized facings, core depth, and core ribbon thickness. It also accounts for the panel end fixity in the presence of biaxial, shear, thermal gradients, temperature degradation, pressure, and constraints on panel deflection and panel rigidity.

- By establishing wing deflection constraints with a fully stressed design, the structural analysis can be optimized more rapidly. A good estimate of the wing deflection constraint reduces the convergence time. Roll effectiveness, control effectiveness, and aerodynamic center movement with Mach number and dynamic pressure can also be more rapidly optimized by deflection constraints.

  The above studies are now used in the advanced design process for the advanced supersonic design. As the design progresses towards production, larger structural models are used with many more structural elements. The FAA has approved the FORMAT methodology.

For the DC-10 substantiation, 100,000 internal structural elements were used by joining 77 substructures. The degree of accuracy of the FORMAT method has been demonstrated by correlating 10 to 15 full-scale static test airplanes. Test strain gages and deflection readings showed excellent correlation when compared with analysis predictions.

ANALYSIS METHODS FOR FAIL-SAFETY

It is desirable to account for fail-safety, crack growth, and residual strength in the initial sizing for structural design. This can be done by cutting a structure member and using the resulting required sizes as the minimum initial size for further optimization studies.

Studies shown in table II optimized spar cap areas for fail-safe design. This increased the panel sizes, reduced spar cap areas, did not appreciably increase the wing weight, but substantially increased fail-safety. It was also found that by using a 400 finite-element model around a crack on a face sheet of a honeycomb panel, the analysis was able to duplicate the test results of the residual strength for honeycomb panels. Notice that titanium honeycomb panels have high residual strength compared with unstiffened sheet (see figure 3). This residual strength, when corrected for actual panel widths, can be used to help determine the initial allowables, when correlated with the crack growth rate for the actual 2.2 Mach cruise design spectra.

TRADE STUDIES

The ability to trade structural weight against specific complex geometry parameters has been developed. A computer graphics program has been developed which enables the analyst to input critical geometry points and quickly create a structural model. This final detailed model can automatically have all elements listed for further analysis. The aerodynamic box loads, inertia fuel loads, and concentrated loads can quickly be transferred to the structural load vectors by special automated programs. Figure 4 and table III are examples of the complex geometry changes analyzed. Figure 4 shows the geometry changes, specifically of thickness ratio variations with span, and of variations in location of maximum thickness ratio. The results of the trade study depicted in table III have been used to assist in the design of the baseline.

A flutter optimization program that uses as an input, derivatives of the strength optimization program's structural influence coefficients as functions of weight, has been developed. Good results have been obtained for variation of the wing thickness ratio both spanwise and chordwise, and the fuel distribution. The aerodynamics section has created a drag equivalent weight for combining with the strength, fatigue, fail-safety, roll and control effectiveness, and flutter penalties for a variation in wing thickness ratio. (See figure 5). Estimates of the structural total weight and the equivalent wing weight are shown in table III. To ease the space and the manufacturing problems, the 3 percent thickness ratio wing has been selected for the baseline design. It is recognized that a slightly reduced thickness ratio wing has improved aerodynamic performance and a small range increase, but such a refinement is beyond the scope of present analysis requirements.

ALUMINUM TRADE STUDIES

Aluminums have been investigated for high temperature long time application. See reference 5. British and French data have been compared with NASA, ALCOA, and other data sources. British test data of percent creep strain against hours is from reference 5 and is compared with NASA data in figure 6. The McDonnell Douglas design life requirements of 100,000 hours (two lifetimes) with 70,000 hours at temperature may be achievable with some of the British/French alloys of aluminum that show a creep strain of approximately 0.1% at 120°C (248°F) at a maximum continuous stress of 17,650 N/cm^2 (25,600 lb/in^2). Creep-fatigue and rupture for aluminum alloys for the 2.2 Mach supersonic cruise vehicle also have been investigated. With the best aluminum alloys, a one g stress of approximately 5516 N/cm^2 (8000 lb/in^2) is recommended to account for the long time creep, rupture strength, thermal stresses, and creep-fatigue effects for long-time temperature exposure for 2.2 Mach cruise vehicles. This is approximately a 40% reduction in the one g stress as compared with subsonic wide-body transports. Titanium does not appreciably deteriorate in allowables due to thermal effects for the 2.2 Mach supersonic cruise vehicle (table IV). The comparable one g stress for titanium is 15,223 N/cm^2 (22,080 lb/in^2). The specific one g stress ratio of aluminum is 5516 N/cm^2 (8000 lb/in^2) divided by a density of 0.1 compared with titanium with 15,223 N/cm^2 (22,080 lb/in^2) divided by 0.16 density. The best aluminums are, therefore, only 58% of the structural efficiency of current annealed 6AL-4V titaniums when used for strength design parts for 70,000 hours at temperature at 2.2 Mach number.

CRUISE SPEED TRADE STUDIES

An in-depth trade study of the structural weights of a 2.0 Mach, 2.2 Mach, and 2.4 Mach aircraft with a common payload and range has been accomplished. Table IV shows the results from reference 1. The most important structure variables are the thermal stress differences from the thermal gradients, the allowables in compression, and the allowables in tension that account for fatigue and fail-safety. Aeroelastic and flutter weight penalties for each of the three Mach number aircraft are included. The relative direct operating costs (DOC) are shown. The DOC for the 2.4 Mach design is appreciably higher (9.6%) than that for the 2.2 Mach design.

COMPOSITE TRADE STUDIES

From what has been learned of composites from the McDonnell Douglas system studies (see reference 1), further range improvements are possible.

These composite studies have been shown to increase the range significantly if substantial development of composites occur. Table V shows that the baseline airplane has a range of 8093 km (4730 n. mi.) with 1977 go-ahead and near-term mini-bypass engine. For such a go-ahead in 1977, an all-metal aircraft of 70% titanium and 30% aluminum is recommended. If the go-ahead date is 1980, sufficient time seems to be available to develop an all-composite graphite-epoxy secondary structure for use on floor beams, flaps, elevators, and other nontemperature-critical areas. In addition, it may be possible to utilize some limited applications of composite-reinforced titanium to reduce the thickness ratio of the outboard wing panel, for example. These improvements could increase the range to 9153 km (4942 n. mi.). By post-1985 the variable-stream control engine (VSCE) could probably be available for airline service, and the base airplane could have a range of 9354 km (5051 n. mi.). By 1985 composite reinforcements of the titanium main wing spars and fuselage longerons and frames with unidirectional boron-epoxy could be sufficiently analyzed and tested to provide confidence in this concept. An additional 602 km (325 n. mi.) range can be realized, yielding a total range of 10,605 km (5726 n. mi.).

STRUCTURAL CONCEPTS TRADE STUDIES

Eight structural concepts are being investigated and have been compared with the baseline aluminum-brazed titanium honeycomb sandwich concept for the wing and weld-brazed skin stiffener concept for the fuselage. Results show that the selected baseline concepts are the best for overall efficiency.* Nonetheless, one of the more interesting concepts, the superplastically formed diffusion-bonded concept (SPF/DB), figure 7, has been compared with the baseline aluminum-brazed titanium honeycomb sandwich, figure 7. For a uniaxial loading, the superplastically formed concept can use some of its core weight to relieve the face skin stresses if it were optimized so that the web core would not fail from local or general stability. However, in the honeycomb sandwich concept, the core only serves to stabilize the face sheets and does not sustain any axial loading. Therefore, some small advantage can be obtained for a uniaxial loading for the SPF/DB concept. However, when transverse loads and shear are added, the McDonnell Douglas SPF/DB concept seems to rapidly lose its advantage, see figure 8. The SPF/DB concept face sheet is stabilized by the pitch of the truss core corrugations rather than the small cell size for the honeycomb concept.

Therefore, the skin can buckle from transverse and shear loads if the pitch is too large. However, making the pitch small seems to increase the

*1976 NASA study for Langley Research Center under preparation by the Douglas Aircraft Co., McDonnell Douglas Corp.

weight. The use of SPF/DB would seem to save fabrication costs compared with the baseline honeycomb concept. Further studies are planned to fabricate and test SPF/DB concepts, and determine overall system cost effectiveness. Notice in figure 7 that special inserts are required at the splicing bolt attachments to prevent crushing the truss core. With honeycomb sandwich, figure 7, this is readily accomplished by increasing the core density near the joint.

Another interesting concept considered is integrally stiffened titanium panels. Figure 9 shows two arrangements compared with aluminum-brazed titanium sandwich with the same weight per square foot. When optimized for uniaxial loads only, the integral panel approaches the honeycomb values in the Nx direction but cannot sustain desirable transverse loads. The test values were obtained in the 1976 study and the analysis optimization results are from a five-mode stability analysis developed at McDonnell Douglas.

For high confidence in the candidate selections, a large-scale test and development program is necessary. Additional analyses to determine the required coupon, small test panels, large test panels and components are necessary. Tests are then required to obtain the allowables, weight, and cost values with the realistic operating requirements for a supersonic cruise vehicle. Cutouts for inspection, lightning strikes, foreign object damage, splices, corrosion, crack growth under realistic loadings, are a few of the requirements. The testing should start with small coupons, sheet, and panels and culminate in large panels and components to yield confidence in the structural concept.

CONCLUDING REMARKS

The advanced design studies of the supersonic cruise vehicle show that the near-term structural efficiency of titanium using the structural concepts that can sustain high biaxial and shear loads along with pressure, thermal degradation, and long-time fatigue, is high enough to obtain the desired range with an arrow wing for a 2.2 Mach cruise design. To insure the weight, cost, and structural integrity of the most promising titanium concepts, an early structural development program must be initiated. This program should have reached the large component test phase in the near term to insure low risk for an early go-ahead.

The use of composites shows high payoffs for the intermediate and far term aircraft. Because of the elevated temperature problems, an intensive analysis, ground test, and flight test program is necessary to insure the structural reliability of the composites for secondary structure and for composite-reinforced titanium primary structure for the intermediate term. Composites are sensitive to the long-time cruise temperature. Therefore, an intensive testing program using the 2.2 Mach cruise spectra for 70,000 hours at temperature and for a 100,000 hour design life (two lifetimes) is desired for graphite-epoxy and boron-epoxy composites. This development program must start with the analysis and testing of small composite coupons, built-up composite-reinforced titanium joints, and built-up composite-reinforced titanium panels. Later phases should include large panels and components that would be static and fatigue tested to the 2.2 Mach cruise vehicles' spectra. This large development test program should run concurrently with a flight test program using various structural concepts of composites and composites reinforcing titanium. The NASA flight test composite programs already completed and those planned are important contributions to test the subsonic environment. However, if a partial composite supersonic cruise vehicle is to be considered for the intermediate term, and it is to have a considerable percentage of composites, the ground structural development test should be supplemented by supersonic cruise flight testing using the composite concepts. Perhaps the near-term all-metal (70% titanium and 30% aluminum) aircraft is the best test bed for the intermediate and far-term partial composite supersonic cruise aircraft since no other vehicle will have the right environment and design life at temperature.

REFERENCES

1. NASA CR-144925: Technology Application Studies for Supersonic Cruise Aircraft. Volume 1, prepared under Contract No. NAS1-13612 for Langley Research Center by the Douglas Aircraft Company, McDonnell Douglas Corporation, Douglas Report MDC J4532, November 1975.

2. Denke, P. H.: A Matric Method of Structural Analysis. Proceedings of the 2nd U.S. Congress of Applied Mechanics, June 14-18, 1954.

3. Pickard, J., et al: FORMAT - Fortran Matrix Abstraction Technique. AFFDL-TR-66-207, December 1968.

4. Dodd, A. J.: Specification for a Static Structural Optimization Capability. McDonnell Douglas Report MDC J5442, March 1972.

5. Harpur, N. F.: Concorde Structural Development. British Aircraft Corporation, Filton Division, Bristol, England.

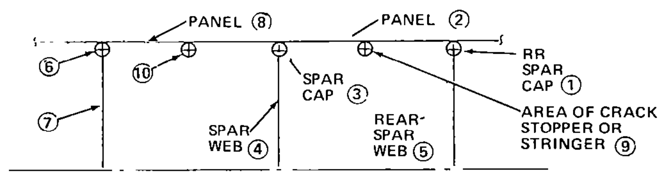
TABLE I.- EXAMPLES OF STRUCTURAL ELEMENTS ANALYSIS FOR STRENGTH, FATIGUE, AND FAIL SAFETY

DESIGN PHASES		ADVANCED DESIGN		DESIGN	DESIGN SUBSTANTIATION
	MODEL TYPE	NUMBER ELEMENTS	MODEL WT kg (LB)	NUMBER ELEMENTS	NUMBER ELEMENTS
DC-10	BARS AND PANELS	6000	CONSTANT VALUE	60,000	100,000
2.2M SCAR	BARS AND PANELS	3938	17,892 (39,445)	–	–
	MEMBRANES	4555	17,232 (37,989)	–	–
	IMPROVED MEMBRANES(1)	4555	14,163 (31,223)	–	–
	SANDWICH PANELS(2)	6365	13,544 (29,859)	–	–

(1) SIZE OPTIMIZATION – INCLUDES INTERACTION EQUATIONS FOR BIAXIAL AND SHEAR LOADINGS, THERMAL EFFECTS, ETC.

(2) SANDWICH OPTIMIZATION – INCLUDES FACING AND CORE THICKNESSES, CORE DEPTH, THERMAL EFFECTS, PRESSURE, ETC.

TABLE II.- FAIL-SAFE CONDITIONS



FAILURE MODE	STRUCTURAL MODEL EQUIVALENT
OUTER OR INNER PANEL FACING FAILURE OF HONEYCOMB SANDWICH (2)	REDUCE AREA OF PANEL (2) BY 50 PERCENT
OUTER OR INNER PANEL FACING FAILURE OF HONEYCOMB PANEL (8)	REDUCE AREA OF PANEL (8) BY 50 PERCENT
OUTER AND INNER PANEL FACINGS FAILURE OF PANEL (2)	CUT PANEL (2) TO ZERO THICKNESS
OUTER AND INNER PANEL FACINGS FAILURE OF PANEL (8)	CUT PANEL (8) TO ZERO THICKNESS
REAR SPAR CAP FAILURE (1)	CUT AREA OF REAR SPAR CAP (1) TO ZERO
INTERMEDIATE SPAR CAP FAILURE CAP (3)	CUT AREA OF SPAR CAP (3) TO ZERO
FAILURE OF PANEL (2) WITH A FAILURE OF CAP (3)	COMBINE III AND VI
FAILURE OF PANEL (2), PANEL (8), AND INTERMEDIATE CAP (3)	COMBINE CASES III, IV, AND VI

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TABLE III.- WING STRUCTURAL BOX WEIGHT OPTIMIZATION

DESIGN		5	-5 MOD	-5A	-5X
$t/_c$ (ROOT, T.E., L.E., TIP)		2.25; 3; 2.5; 2	2.25; 3; 2; 2	2.25; 3; 3; 3	2.25; 3; 4; 4
MAX $t/_c$ LOCATION – % C		40 TO 60	60 TO 75	60 TO 75	60 TO 75
STRENGTH + FAIL-SAFE + AEROELASTIC	kg (LB)	28,123 (62,000)	22,226 (49,000)	21,772 (48,000)	21,319 (47,000)
FLUTTER	kg (LB)	2,268 (5,000*)	907 (2,000)	390 (860)	390 (860)
ROLL EFFECTIVENESS	kg (LB)	2,722 (6,000**)	1,814 (4,000**)	907 (2,000**)	680 (1,500**)
ESTIMATED WING BOX	kg (LB)	33,113 (73,000)	24,947 (55,000)	23,069 (50,860)	22,389 (49,360)
DRAG EQUIVALENT	kg (LB)	0 0	2,268 (5,000)	· 5,216 (11,500)	9,253 (20,400)
EQUIVALENT TOTAL	kg (LB)	33,113 (73,000)	27,215 (60,000)	28,285 (62,360)	31,642 (69,760)

*4,536 kg (10,000 LB) FUEL PER SIDE MOVED INBOARD
**PRELIMINARY ESTIMATE

BASELINE (-5A)

TABLE IV.- MACH TRADE STUDY SUMMARY

273 PASSENGERS — RANGE 7400 km (4000 N MI)

DESIGN MACH NUMBER		2.0	2.2	2.4
TAKEOFF GROSS WEIGHT	kg (LB)	311,255 (686,200)	320,962 (707,600)	372,388 (820,974)
ENGINE AIRFLOW	kg/SEC (LB/SEC)	299 (660)	320 (705)	397 (875)
CRUISE SFC (INST)	kg/HR/N (LB/HR/LB)	0.1342 (1.316)	0.1403 (1.376)	0.1516 (1.487)
CRUISE L/D		9.73	9.33	8.86
OPERATING EMPTY WEIGHT	kg (LB)	129,410 (285,300)	134,799 (297,182)	153,228 (337,871)
ΔT = THERMAL GRADIENT	^{o}C (^{o}F)	49 (120)	71 (160)	93 (200)
ALLOWABLES* IN COMPRESSION	N/cm^2 (LB/IN.^{2})	–74,939 (–108,690)	–68,644 (–99,560)	–62,039 (–89,980)
ALLOWABLES* IN TENSION FATIGUE PLUS FAIL-SAFE	N/cm^2 (LB/IN.^{2})	75,842 (110,000)	68,948 (100,000)	63,432 (92,000)
RELATIVE DOC		0.988	1.00	1.096

*INCLUDES INTERACTION EQUATIONS, SIZE DEPENDENT ALLOWABLES, THERMAL STRESSES, AND DEFORMATION CONSTRAINTS (NASA CR-144925)

TABLE V.- EFFECT ON AIRCRAFT RANGE BY ADDING COMPOSITES

		1977	1980	POST-1985
BASELINE AIRPLANE RANGE	km (N MI)	8093 (4370)	8504 (4592)	9,354 (5,051)
ENGINE TYPE		NEAR TERM MINI-BYPASS	ADV TECH MINI-BYPASS	ADV TECH VSCE-502BD
COMPOSITE SECONDARY STRUCTURE	km (N MI)	–	232* (125*)	232* (125*)
NEW OUTBOARD WING PANEL WITH LOWER $t/_c$'S AND COMPOSITES	km (N MI)	–	417* (225*)	417* (225*)
COMPOSITE REINFORCED METAL WING SPARS AND RIBS	km (N MI)	–	–	380* (205*)
COMPOSITE REINFORCED METAL FUSELAGE LONGERONS AND FRAMES	km (N MI)	–	–	185* (100*)
COMPOSITE LANDING GEAR BRACES	km (N MI)	–	–	37* (20*)
TOTAL RANGE	km (N MI)	8093 (4370)	9153 (4942)	10,605 (5,726)

*RANGE INCREASE

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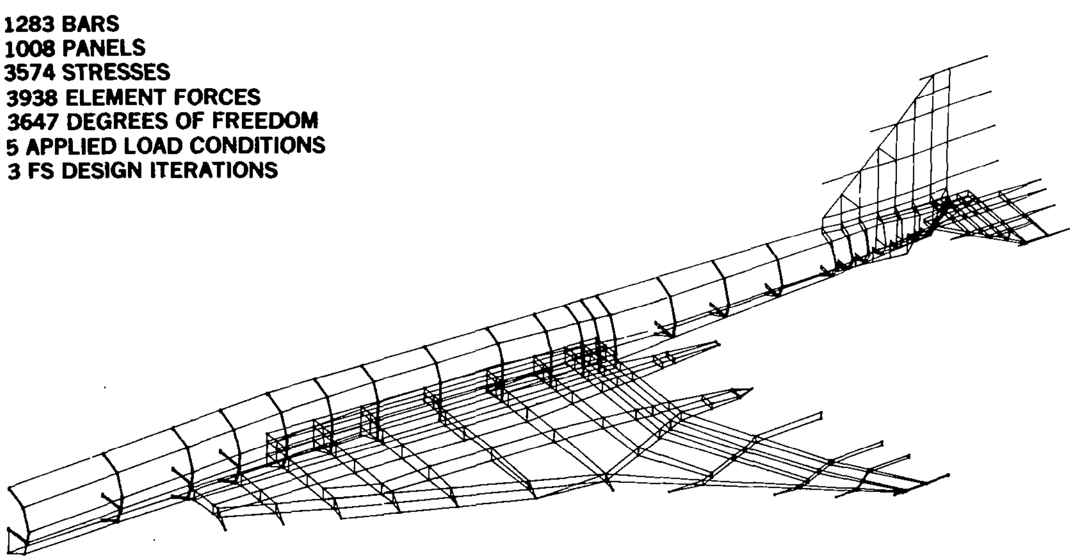


Figure 1.- Typical structural analysis.

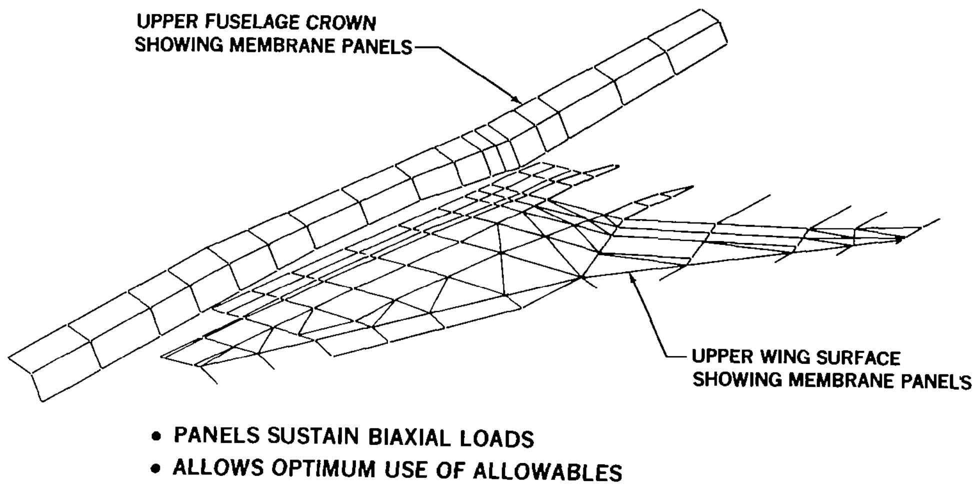


Figure 2.- -5A structural model.

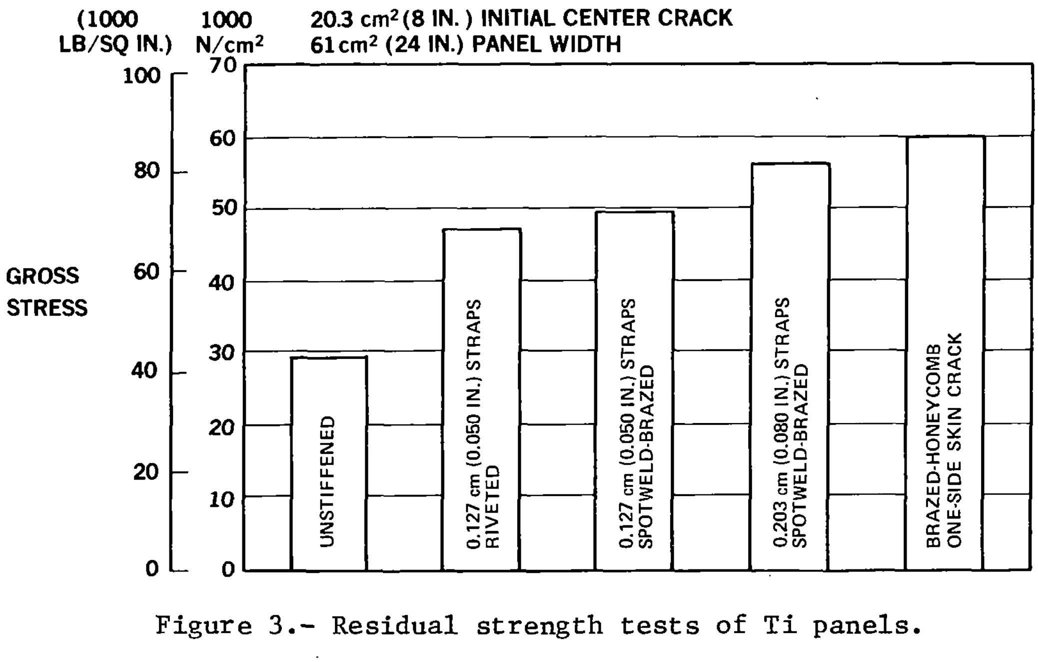


Figure 3.- Residual strength tests of Ti panels.

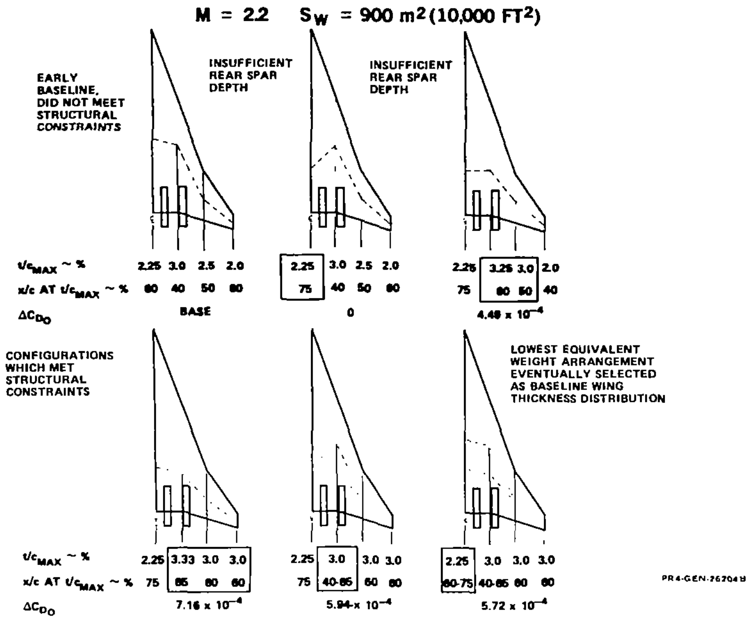


Figure 4.- Inboard panels thickness distribution study.

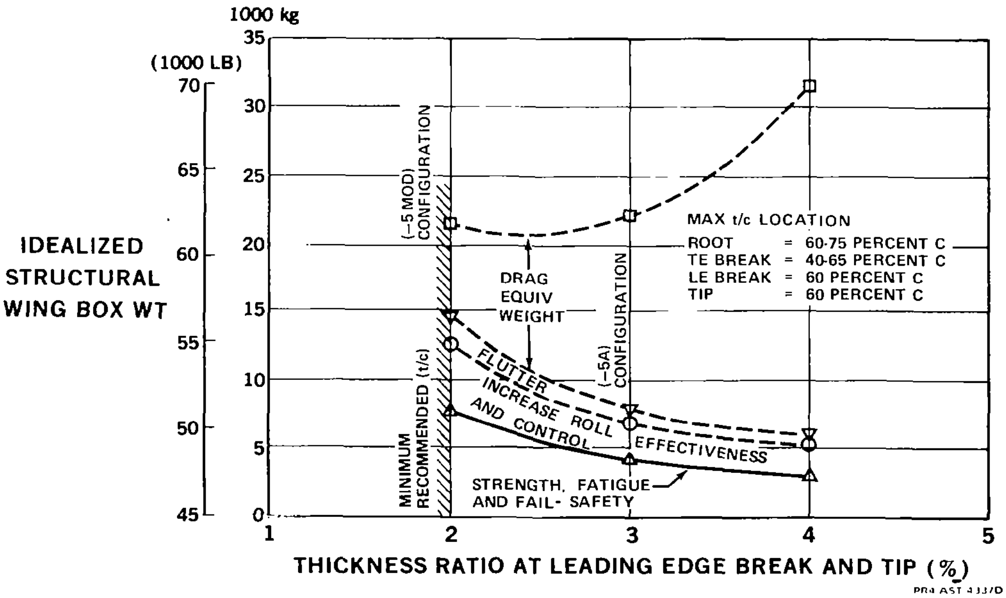


Figure 5.- Structural box weight optimization for AST wing.

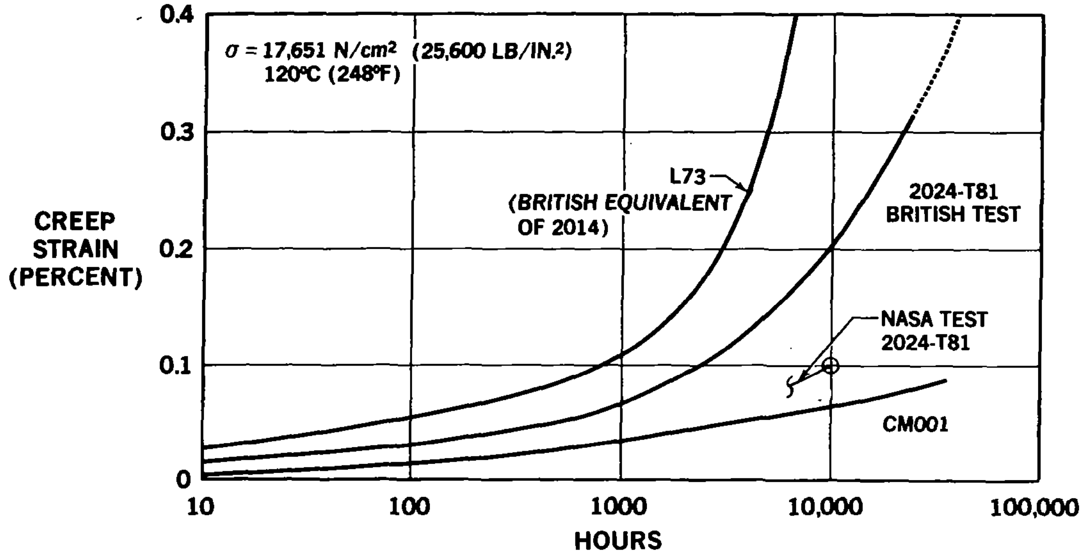


Figure 6.- Creep strain of aluminum materials.

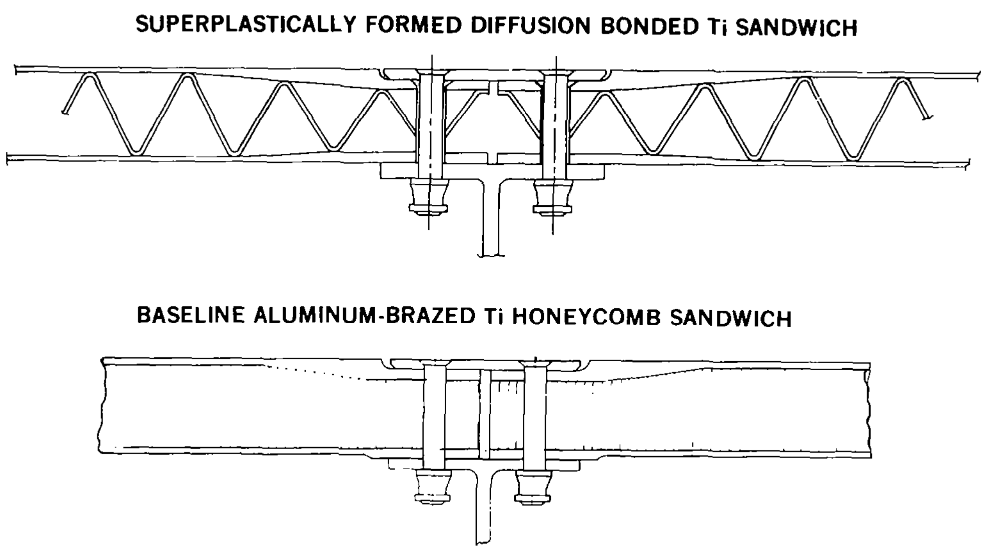
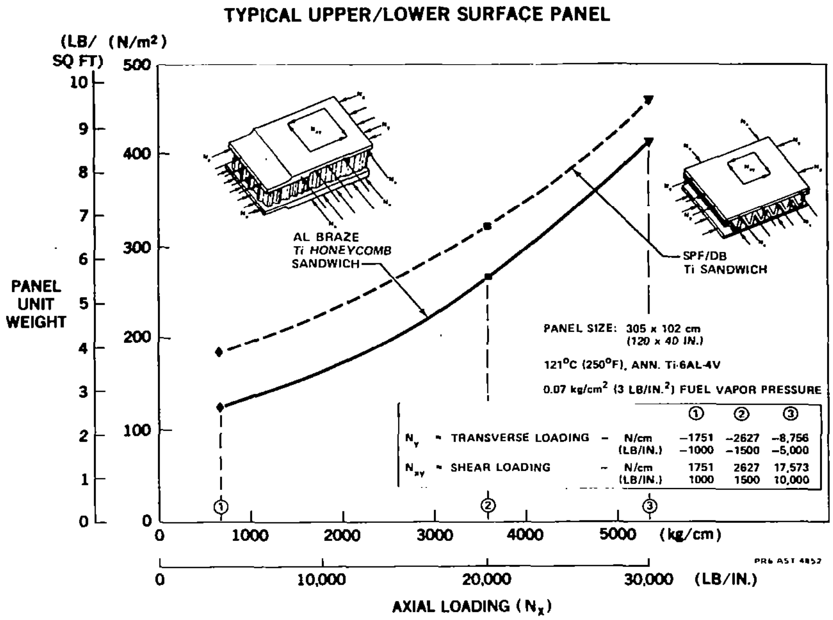


Figure 7.- Candidate structural concepts.



			①	②	③
N_y = TRANSVERSE LOADING	–	N/cm	–1751	–2627	–8,756
		(LB/IN.)	–1000	–1500	–5,000
N_{xy} = SHEAR LOADING	–	N/cm	1751	2627	17,573
		(LB/IN.)	1000	1500	10,000

Figure 8.- Panel weights for candidate concepts.

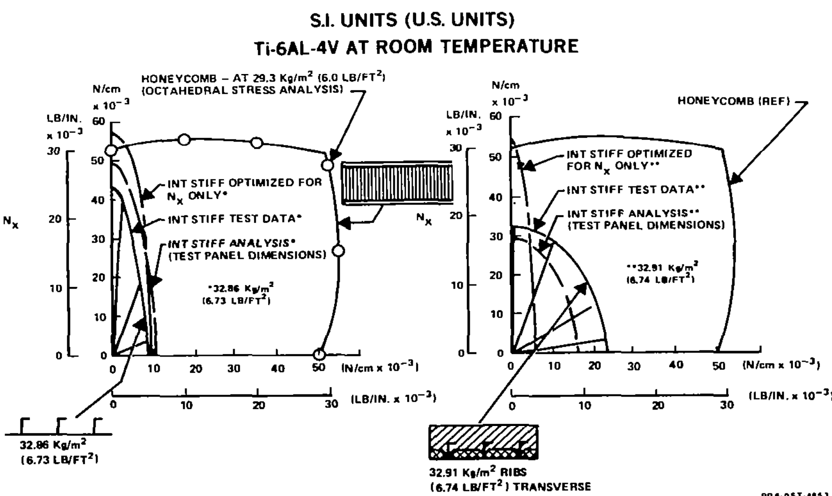


Figure 9.- Comparison of ultimate loads of aluminum brazed Ti honeycomb and integrally stiffened Ti panels.