Structural Materials for Supersonic Transport

G. A. Fairbairn*
North American Aviation, Inc., Los Angeles, Calif.

The selection of materials for the airframe structure on the supersonic transport (SST) must be established on the basis of requirements considered important to the SST. These criteria are 1) efficiency in design and performance, 2) cost-weight analysis, 3) economical and reproducible fabrication processes, 4) long airframe life in the SST environment with extended life expectancy beyond the design minimum, and 5) minimum maintenance required during the life of the aircraft. The static properties under short-time and long-time exposure, in relation to the density of materials, are a measure of structural efficiency. Other properties, such as fatigue, fracture toughness, and corrosion resistance, are important in ensuring a long-life airframe structure requiring a minimum amount of maintenance. Finally, materials must be evaluated in terms of the various types of structures under consideration. This is necessary in order to obtain an optimum relationship between cost and weight and to insure that economical and reproducible fabrication processes can be used in the manufacture of the airframe structure.

Selection of Material

In making material selections for a supersonic transport, there are certain criteria to be considered which are important in the development of an aircraft profitable for commercial airline operation. These criteria include 1) efficiency in design and performance, 2) long life with minimum maintenance, 3) economical and reproducible fabrication processes, and 4) cost-weight analysis.

The purpose of this paper is to discuss some of the materials being considered for the airframe structure of the SST in consideration of the forementioned criteria. Inasmuch as titanium is being proposed as the primary metal for the airframe, the discussion will be concerned principally with an evaluation of titanium, including some comparisons to 2219 aluminum alloy and PH14-8Mo stainless steel. The 2219 aluminum alloy is included, since it is one of the best high-temperature aluminum alloys. The data presented are for the T81 heat-treat condition. The PH14-8Mo stainless steel is included since it is a relatively new heat treatable and weldable material developed for an optimum combination of corrosion resistance, strength, and fracture toughness. The data given are for the SRH1050 heat-treat condition. The titanium alloys, 6Al-4V and 8Al-1Mo-1V, discussed in this paper, are considered to have good high-temperature strength, fracture toughness, and weldability. The data presented are for the 8Al-1Mo-1V alloy in the duplex annealed condition and the 6Al-4V alloy in the annealed condition. These conditions were selected for the best fatigue strength and fracture toughness properties, which are of utmost importance to the SST.

For purposes of this discussion, the following design requirements have been selected from the evaluation data available from the North American Aviation proposal effort on the SST. These are a cruise speed of Mach 2.65, a cruise temperature (maximum) of 420°F for skin and 475°F for stagnation, and an operating life (minimum) of 36,000 hr (total) and 24,000 hr (cruise temperature); the number of flights (minimum) is 26,000.

Efficiency in Design and Performance

Static strength properties generally are the basis for initial tradeoff studies in comparing material for structural efficiency. The customary method of measuring structural efficiency involves the determination of strength-weight ratios, which are a measure of the load-carrying capability of materials, for a given amount of weight. Figures 1–4 show some comparative strength-weight data on 2219 aluminum, PH14-8Mo stainless steel, and 6Al-4V and 8Al-1Mo-1V titanium alloys over the operating temperature range for the SST structure. Figure 1 includes data relating to the compressive yield strength of materials. This strength parameter is very significant, since the sizing of the airframe structure is controlled more by compression loading than by any other single factor. Stiffness is another important property of structural materials, and this is shown on Fig. 2 as a modulus-density ratio parameter. Both of the forementioned data plots indicate that the stainless-steel and titanium alloys have higher structural efficiency for the SST design requirements than the high-temperature 2219 aluminum alloy. The comparisons also show that the 8Al-1Mo-1V titanium alloy is the optimum material choice. In consideration of the long life requirements for the SST, the data presented have been extrapolated from 2000-, 5000-, and 10,000-hr data to show the compression yield strength and elastic modulus when tested after 24,000-hr exposure to the test temperature.

It is also necessary to look at the properties of the materials after exposure to the combined effects of load and temperature for long periods of time. The properties usually evaluated under these conditions are the creep and stress-rupture strengths. Creep may be defined as the property of materials to undergo continuing plastic deformation under long-time exposure to temperature at a constant stress. Generally, the allowable stress level for creep must be below the yield strength of the material before creep strength is considered

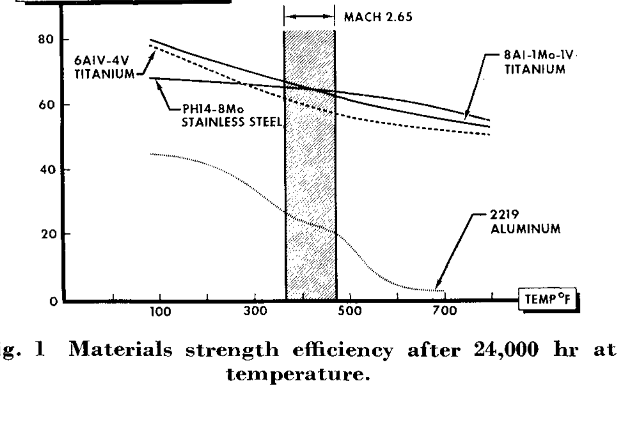


Fig. 1 Materials strength efficiency after 24,000 hr at temperature.

Presented as Preprint 62-628 at the AIAA Transport Aircraft Design and Operations Meeting, Seattle, Wash., August 10–12, 1964; revision received November 20, 1964.

* Project Engineer, Materials and Producibility for Advanced Systems, Los Angeles Division.

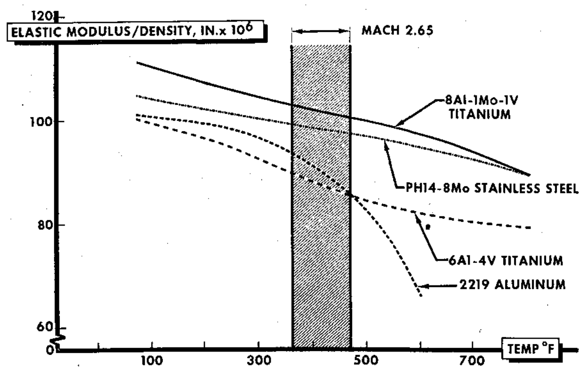


Fig. 2 Materials stiffness efficiency after 24,000 hr at temperature.

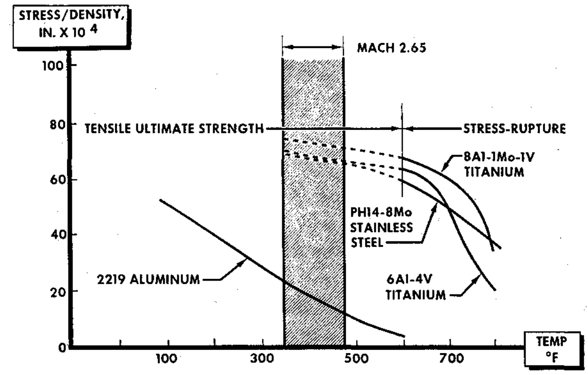


Fig. 4 Stress-rupture efficiency of materials after 24,000 hr at temperature.

limiting in design. Figure 3 shows a creep-strength parameter for the candidate SST materials plotted for comparison on a stress-density basis. The upper left-hand portions of the curves for the titanium and steel alloys are shown as dotted lines to indicate that creep is not significant in this regime, and the static yield strengths of the materials are limiting. Stress-rupture is the property of a material to fail after long-time exposure to temperature and load at stresses less than the tensile ultimate strength of the material. Failure may or may not be preceded by significant amounts of creep, depending upon the stress, temperature, and time-to-failure conditions. The comparative structural efficiencies of the aluminum, stainless-steel, and titanium alloys, from the standpoint of stress-rupture, are presented in Fig. 4. As in Fig. 3, the upper left-hand portions of the curves are shown as dotted lines to indicate that stress rupture is not a significant factor in this regime, and the static ultimate strengths of the materials are limiting. The data from Figs. 3 and 4 again indicate the superiority of the stainless-steel and titanium alloys over aluminum for SST structures, with the 8Al-1Mo-1V titanium alloy showing the best efficiency.

In summary, from the standpoint of structural efficiency, the aluminum alloys should be eliminated from further consideration as the prime structural material for the SST. The titanium alloys, particularly the 8Al-1Mo-1V alloy, appear to be the optimum choice for the majority of the SST airframe structure. Certainly, special requirements or cost factors in specific applications, such as passenger compartment structure, landing gear, trailing edges, and engine nacelle structure, will dictate the use of aluminum, steel, nickel-base alloys, and other materials.

Long Life and Minimum Maintenance

One of the essential considerations in the design of an SST is to provide an airframe structure that will have a long service life with a minimum of structural maintenance and repair. Fatigue strength, fracture toughness, and corrosion are the important material properties to consider in evaluating materials for long-time serviceability.

Fatigue

The fatigue properties, or relative strengths of materials as a function of the number of cycles of alternating stress, are necessary design criteria for structural components on the SST. It is expected that the majority of the tension-critical structure on the SST will be designed to fatigue allowables. The fatigue properties of materials in the presence of a notch are particularly important, since practical design and fabrication considerations do not permit the elimination of all stress concentrations. The effects of service temperatures and extended exposure at elevated temperature upon the fatigue properties are also necessary design criteria. A further requirement is the establishment of the effects produced upon the fatigue properties by fabrication processes such as forming, machining, and welding.

A comparison of the fatigue characteristics of aluminum, stainless-steel, and titanium alloys is shown in Fig. 5. The data are for room temperature and for a notch factor of K_t = 2.4–2.5, which is equivalent to a drilled and deburred open hole. The 2024-T3 aluminum alloy was included on the plot for comparative purposes, since it is a familiar structural material used on current transport aircraft. Examination of the data indicates that the fatigue characteristics of the PH14-8Mo stainless steel and the 8Al-1Mo-1V titanium alloy compare favorably with the characteristics of 2024-T3 aluminum. Although the test data on the titanium and stainless-steel alloys are quite limited in comparison with the aluminum, the steel and titanium appear to have better fatigue properties than 2024 aluminum under conditions of longer cycle life. In addition, the fatigue properties of the 8Al-1Mo-1V alloy are better than the properties of Ph14-8Mo stainless steel for the short cycle life and are essentially the same at the longer cycle life. The data on spotwelded 6Al-4V titanium are included to show the notch effect of the spotweld on fatigue.

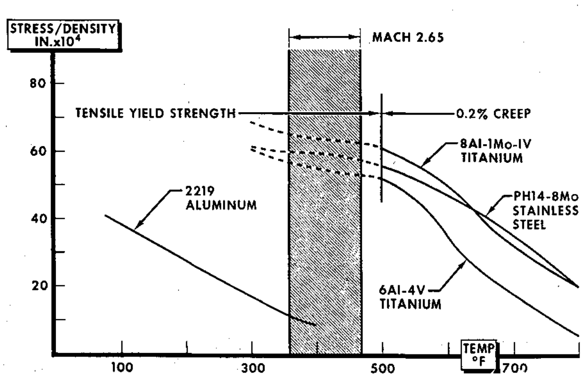


Fig. 3 Creep-strength efficiency of materials after 24,000 hr at temperature.

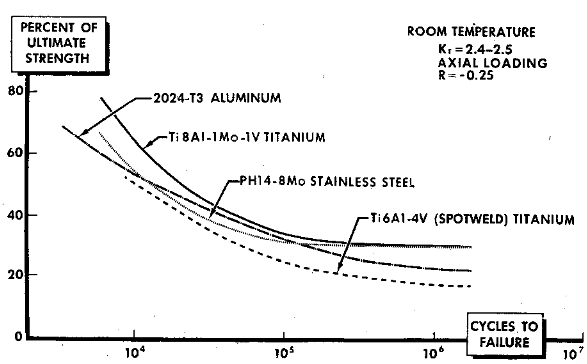


Fig. 5 Materials: fatigue comparison.

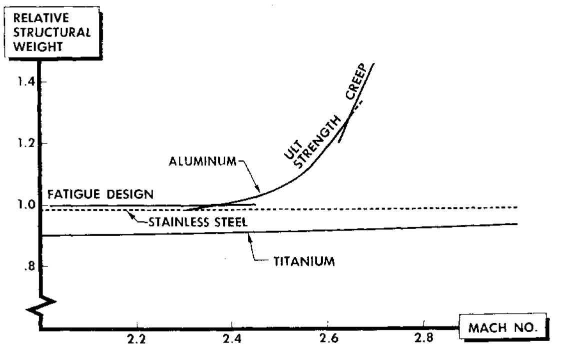


Fig. 6 Relative weights: tension critical structure.

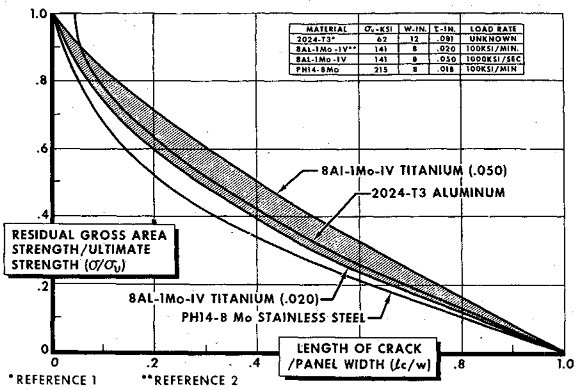


Fig. 8 Relative residual strength of material, 80°F.

The S-N curve shown for 6Al-4V titanium is not applicable to a spot-welded joint, but is representative of the fatigue properties of a skin containing spot welds for the attachment of stiffeners. As such, the data show the notch effect (geometric and metallurgical) because of the discontinuity existing in the form of a spot weld. Referring to Fig. 5, the 6Al-4V alloy spot-weld data can be compared to the 8Al-1Mo-1V alloy parent metal S-N curve, since they are very similar alloys from the standpoint of fatigue in the annealed condition. The comparison shows a significant effect resulting from the spotwelding operation which might indicate a K_t in the order of 3.5 as a measure of the notch effect from spotwelds. The fatigue properties of welded joints in titanium will be discussed further in a later the section of this paper dealing with welding.

A comparison of the efficiency of using aluminum, stainless steel, or titanium for a fatigue critical structure is shown in Fig. 6. The relative weights of such tension critical structure as the lower wing skin are plotted as a function of Mach number, using the weight of aluminum as unity. Examination of the data shows that the fatigue allowable used to size the aluminum structure for a typical SST ground-air cycle loading spectrum remains the limiting factor for speeds up to about Mach 2.35. Beyond this point, the structure must be made heavier as the hot strength (tensile ultimate and later creep) of aluminum becomes more limiting than fatigue. The weight of the stainless-steel and titanium structure, designed for the same fatigue conditions as the aluminum, is practically unaffected by increasing speeds up to at least Mach 3.0.

Before leaving the subject of fatigue, I would like to briefly review the fatigue characteristics of high-strength steel that will probably be used on the SST for major structural fittings such as landing gear and the vertical stabilizer support. Many designers tend to use as high a heat-treat strength as possible on such fittings in order to save weight and space. In the case of structure designed to fatigue allowables, such as on the SST, there may not always be an advantage in heat treating to the highest strengths. This is particularly important in view of the desirability of providing a long-life, low-maintenance airframe. The higher-strength levels are generally attained with a sacrifice of toughness, attendant with increasing difficulties in cleaning, heat treating, and finishing operations. With the foregoing problems in mind, fatigue data were accumulated on 4340 steel heat-treated to two strength levels, 220,000 and 260,000 psi. The 4340 alloy was chosen for this study because of the large amount of fatigue data available on the alloy. Figure 7 shows a comparison of the fatigue strengths of 4340 at the two strength levels on the basis of the actual fatigue allowable strength for any given cycle life. As can be seen from the data, there is no advantage in using the higher static strength material for structure which is sized by fatigue for a life of 1000 cycles or more. The lower-strength level will provide greater toughness and a structure more tolerant of the nicks and scratches that occur in processing and service.

Fracture Toughness

The structural materials used in the SST design must be resistant to notch effects and possess good toughness in order to provide a long life and safe aircraft structure. The skin structures, in particular, will require materials with a high resistance to rapid (unstable) crack propagation under static or alternating loads. This requirement is in recognition of the fact that large structures completely free from small cracks are seldom realized even when the best fabrication and maintenance techniques are used. Fracture toughness data, including the residual strength of a part containing a fatigue crack, are also necessary material criteria. For purposes of this paper, the discussion will be limited to data pertaining to the rate of crack propagation and the residual strength of sheet metal panels containing center cracks through the material. Figure 8 shows a comparison of the static load-carrying ability of aluminum, titanium, and stainless-steel panels containing center cracks of various lengths. The criterion used for comparison is the ratio parameter of the strength of the

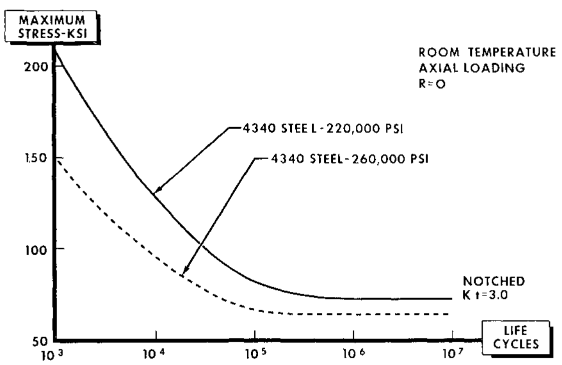


Fig. 7 High-strength steel: fatigue strength.

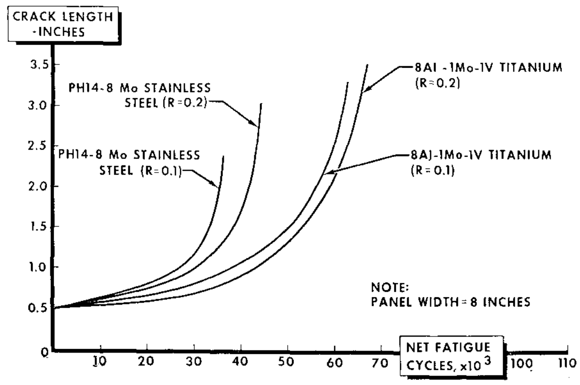


Fig. 9 Fatigue crack growth in materials.

panel at failure (based on the original gross cross-sectional area including crack area) and the ultimate strength of the material. When this parameter is plotted against the ratio parameter of the crack length to the panel width, it becomes a measure of the sensitivity of the material to the presence of the crack. The plot would be a straight line joining a value of 1.0 on the ordinate with the corresponding value on the abscissa in the case of a material completely insensitive to the crack, since the two ratio parameters would be equal to one another. Examination of the data on Fig. 8 shows that the 8Al-1Mo-1V titanium alloy[2] would be expected to behave like 2024-T3 aluminum alloy[1] in the presence of a crack and would be less sensitive to cracks than PH14-8Mo stainless steel.[2] It is interesting to note that, the 0.050 gage 8Al-1Mo-1V titanium sheet, which was tested at a higher loading rate, exhibited considerably better toughness than the 0.020 gage material. This is probably partly due to the effect of the increased gage thickness, but also the higher loading rate (faster strain rate) undoubtedly had a pronounced effect. In addition, the two gages were from different heats of material, which resulted in an effect from compositional differences in the alloy. The preceding effects, when coupled with the effect of normal testing variables, can cause significant variations in the results which make it difficult to generalize on the relative toughness of materials. Certainly, when attempting to make such comparisons, it is most important that the material, the material processing, and the test parameters be clearly defined. Figure 9 presents a comparison of the crack propagation resistance of the 8Al-1Mo-1V titanium and PH14-8Mo stainless-steel alloys. Both of the materials appear to have good resistance to crack growth for crack lengths up to approximately 1.5 in. The titanium alloy exhibited a slower crack growth for the same number of cycles than the stainless steel and tolerated a longer crack before failure. The influence of the R factor (ratio of minimum to maximum load) is also shown, with the lower R factor producing a faster rate of crack propagation as would be expected.

Stress Corrosion

Another long-life property that is particularly important for SST structural materials is the resistance to stress-corrosion cracking. The structural materials selected should be resistant to cracking within the anticipated service combinations of tensile stress (operational plus residual from fabrication) and corrosive environments (marine atmospheres together with aerodynamic heating).

Titanium alloys generally display an excellent resistance to corrosion. At elevated temperatures, however, stress-

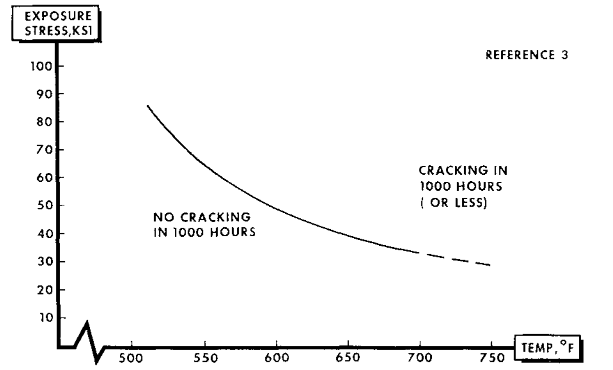


Fig. 10 Hot stress-corrosion cracking in 6Al-4V titanium (annealed).

corrosion cracking has been observed in some of these alloys when they were highly stressed in tension and were in contact with various compounds containing chlorine. As early as 1955, surface cracking was noted on titanium alloy creep test specimens in areas where residual salt (sodium chloride) remained from fingerprints. The susceptibility of some of the titanium alloys to hot salt stress-corrosion cracking has since been confirmed by numerous investigators; however, the fundamental corrosion mechanism remains to be identified.

The stress-corrosion cracking susceptibility of titanium alloys in hot salt is of particular concern in the selection of structural materials for the SST. During operational service from coastal airports, salt deposits could accumulate both on the exterior surfaces and on certain interior surfaces as well, and the subsequent aerodynamic heating from supersonic flight would result in a hot-salt environment.

Although the kinetics of the stress-corrosion cracking of titanium alloys while in contact with hot salt have not been established, it appears that the susceptibility to cracking for a given alloy exists when certain threshold stress-temperature conditions are exceeded under stagnant air conditions. Figure 10[3] shows the exposure stress and temperature conditions that are required to produce hot-salt stress-corrosion cracking in the 6Al-4V alloy within 1000 hr. It can be seen that, as the exposure temperature is reduced, the threshold stress is raised; and, as the temperature approaches 500°F, the threshold stress appears to become asymptotic.

At the present time, only limited hot-salt stress-corrosion data are available for the 8Al-1Mo-1V alloy, and these have been obtained from programs with widely varying test conditions. Table 1[4,5] summarizes the test results currently available for this alloy. There is considerable scatter in the test

Table 1 Hot-salt/stress-corrosion test results for 8Al-1Mo-1V titanium[a]

Exposure temperature, °F	Exposure stress, psi	Exposure time, hr	Remarks	Reference
450	30,000	2,000	No failure	NAA
	75,000	2,500	No failure	NAA
500	40,000	2,625	No failure	4
550	30,000	2,000	No failure	NAA
	40,000	<1,000	Failed; less than 1000 hr	4
	75,000	2,500	No failure	NAA
600	40,000	<1,000	Failed; less than 1000 hr	4
	80,000	100	No failure	5
650	28,000	2,640	Mill annealed; failed	NAA
	28,000	3,890	Mill annealed; failed	
	28,000	18,000	Mill annealed; no failure	NAA
	60,000[b]	18,000	Mill annealed; no failure	NAA
	60,000[b]	18,000	Mill annealed; no failure	
700	22,000	100	No failure	5
	40,000	<100	Failed; less than 100 hr	5

[a] Duplex annealed unless otherwise noted.
[b] Maximum stress at root of 60° notch ($K_t \simeq 2.0$).

data which indicates the sensitivity of the hot-salt stress-corrosion phenomenon to actual test parameters, such as material composition and condition, thickness and moisture content of the salt coating, and air-flow conditions. The results also emphasize the urgent need for additional test data obtained under conditions that simulate realistic operational service and with time intervals extended to 30,000 hr. It is significant, however, that no evidence of stress-corrosion cracking has been reported by any investigators, up to the present time, at temperatures of 500°F or lower.

Considerable research is underway for the SST to develop data and solutions to the hot-salt corrosion problem on titanium. The programs include the evaluation of protective coating systems for titanium. This work is summarized in a paper presented by Richard H. Raring of NASA at the 1st AIAA Annual Meeting in Washington D. C.[6]

The precipitation-hardening stainless steels are not subject to hot-salt stress-corrosion cracking within the anticipated SST temperature regime. These alloys, however, are not completely resistant to corrosion in marine atmospheres, and some compositions have been proven to be susceptible to stress-corrosion cracking at ambient temperature in these environments. Tests conducted under these conditions on PH14-8Mo stainless steel showed the alloy to be highly resistant to stress-corrosion cracking in marine environments, although not completely resistant to rusting. Table 2 is a summary of results obtained from stressed PH14-8Mo test specimens exposed to 1) 5% salt spray, and 2) natural marine beach environments. It can be seen that the only specimens that failed had been heated in air for 1000 hr at 650°F prior to testing. The reason for the increased stress-corrosion cracking susceptibility after the 650°F exposure has not been determined, but it probably is interrelated with the metallurgical instability of PH14-8Mo at this temperature. The instability may result in a decreased corrosion resistance, particularly in localized areas of high residual tensile stresses, and a decreased fracture toughness. In any case, the effect of long-time exposure at temperatures of 450° to 550°F upon the subsequent stress-corrosion susceptibility of this alloy warrants further investigation.

From the foregoing discussion, it can be concluded that the titanium and stainless-steel materials will be suitable for the SST structure from the standpoint of long-life serviceability. In the matter of general corrosion resistance and durability against mechanical damage, the steel and titanium structures are expected to be superior to aluminum. There is a need for additional data in the area of stress corrosion and fracture toughness in order to better define potential problem areas and necessary corrective action.

Fabrication

Manufacturing processes for all phases of fabrication (forming, machining, and joining) of titanium and stainless-steel materials have been established and have been utilized to control the manufacture of past and current-day aircraft. North American Aviation (NAA) established forming and machining practices for titanium and precipitation-hardening stainless-steel alloys for the early F-86 and F-100 aircraft. Although this work was done over 10 years ago, and many innovations have been added, such as high-energy forming and the use of cryogenic temperatures for forming and machining, the basic processes are still in use. The intent of this paper is not to describe the fabrication processes, but rather to point out that accepted processes are available for fabricating of the SST. Certainly the fabrication of titanium and steel airframe structures is expected to be more expensive than comparable aluminum structure. In addition, the availability of special product forms, such as extruded or rolled shapes, castings, and tubing, is likely to be more limited in titanium and stainless steel than in aluminum. The forementioned considerations indicate the necessity of carefully reviewing not only the material selection, but also the structural component concepts and available processes for fabricating components. This review becomes a tradeoff study on cost effectiveness which will be discussed later in this paper.

Table 2 Stress-corrosion test results for PH14-8Mo stainless steel in marine environments

EXPOSURE ENVIRONMENT	HEAT TREAT CONDITION	EXPOSURE STRESS (% OF FTY)	GRAIN ORIENTATION	EXPOSURE TIME (1000 HOURS) 1 2 3 4 5 6 7 8 9 10
SALT SPRAY (5%)	SRH 1050	80	T	NO FAILURE- TEST DISCONTINUED
		60	T	NO FAILURE- TEST DISCONTINUED
	BCHT 1050*	80	T	NO FAILURE- TEST DISCONTINUED
		60	T	NO FAILURE- TEST DISCONTINUED
MARINE (KURE BEACH)	SRH 1050	80	L	NF→
			T	NF→
		50	L	NF→
			T	NF→
	BCHT 1050*	80	L	NF→
			T	NF→
		50	L	NF→
			T	NF→
	SRH 1050 +1000 HRS AT 650°F	80	L	NF→
			T	2 OF 5 SPECIMENS FAILED
		50	L	NF→
			T	NF→
	BCHT 1050* +1000 HRS AT 650°F	80	L	NF→
			T	5 OF 5 SPECIMENS FAILED
		50	L	NF→
			T	NF→

*BRAZE CYCLE HEAT TREAT
1700°F FOR ONE HOUR COOL TO 1000°F
IN 45 MIN. AIR COOL TO ROOM TEMPERATURE
-100°F FOR EIGHT HOURS 1050°F FOR ONE HOUR

5 SPECIMENS PER TEST CONDITION
NF→ = NO FAILURES - TESTS CONTINUING

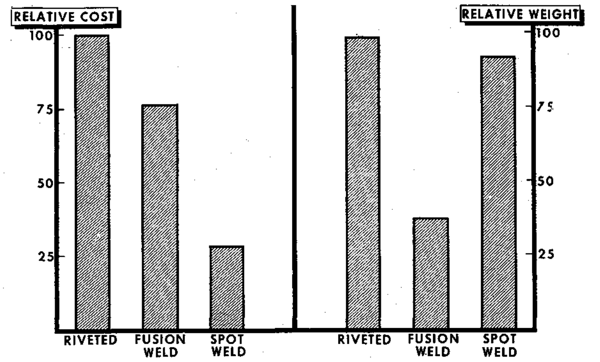


Fig. 11 Cost and weight comparison of spot-welded, fusion-welded and riveted joints.

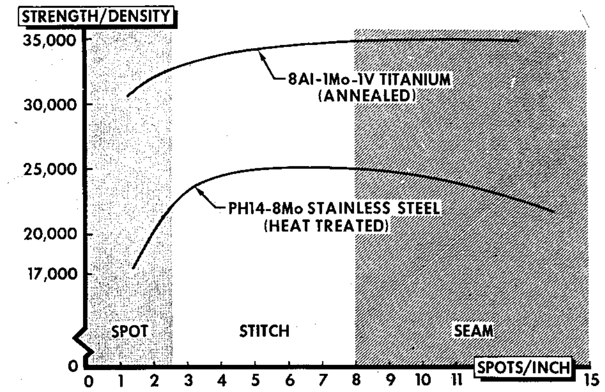


Fig. 13 Strength vs spot spacing for resistance welds.

It is expected that welding will be used extensively on the SST airframe structure for two principal reasons. The first reason is related to the use of titanium and stainless steel rather than aluminum as the primary structural material. This change means that skins will be of thinner gage material because of the higher inherent strengths of titanium and steel over aluminum. The thinner skins are less suitable for mechanically fastened joints and yet require more closely spaced supporting structure fastened to the skin for local stability. The economy and weight savings to be realized by using welding instead of mechanical joints for such structure are considerable and result in a definite preference by designers to use welding. Figure 11 shows a general comparison of the relative weight and cost of spot-welded and fusion-welded joints vs riveted joints. The second reason for expecting an extensive use of welding on the SST is related to fuel containment. The tendency toward the use of integral tank structure for maximum fuel volume and the degradation of organic sealant materials at SST temperatures are considerations that influence designers to favor welding for the attachment of stiffeners and the subassembly joining of stiffened panel structures in fuel-containment areas.

The welding of titanium structure on the SST can be readily accomplished for the most part with conventional welding processes and equipment. Resistance welding, using the spot, seam, or stitch welding processes for different spot spacings, in some respects is less a problem with titanium than with aluminum because of the higher electrical resistance of titanium and less stringent cleaning requirements prior to welding. Spot welding of titanium aircraft structure is not new and was used in applications such as the aspirator section on the F-100 rear fuselage shown in Fig. 12. There is a significant difference between the properties of resistance-welded, alpha-type titanium alloys, such as annealed 8Al-1Mo-1V, when compared to the heat-treated stainless-steel and aluminum alloys. This difference, which shows up primarily as higher strength in the weld-nugget area in the case of the titanium welds, has an effect on the strength of joints with varying spot spacings that are obtained with spot, stitch, and seam welding. For annealed titanium, the lap shear strength continues to increase with closer nugget spacing, and the highest joint strengths are obtained with seam welding. In the case of the steel and aluminum, there is an optimum spacing, since the annealing effect of the weld nugget becomes more significant because of closer nugget spacing. This effect, which compares the influence of weld-nugget spacing on the joint efficiency of 8Al-1MO-1V titanium and PH14-8Mo stainless steel, is shown in Fig. 13. Another way of showing the effect of the higher nugget area strength of spot-welded titanium is given in Fig. 14, where it can be seen that higher joint strengths are obtained in both lap-shear and cross-tension loading by increasing the weld-nugget diameters.

The fatigue strength of resistance-welded joints in most metals is relatively low. This is related primarily to the notch effect of the weld, which is discussed briefly in the earlier discussion on fatigue. The fatigue strengths of spot-welded joints in the 8Al-1Mo-1V and 6Al-4V titanium alloys are comparable to the fatigue strengths of such joints in stainless steel and aluminum as measured on the basis of a percent of the tensile ultimate strength of the parent metal. Figure 15 shows *S-N* curves for simple spot-, stitch-, and seam-welded joints in the 8Al-1Mo-1V alloy. It is interesting to note that the stitch- and seam-welded joints showed better fatigue strength than spot-welded joints. This could be expected on the basis of the higher static strength obtained with closer nugget spacing, as discussed previously. Attempts have been made to raise the fatigue strength of resistance-welded joints by using various spotweld pattern geometries and reinforcement in the weld area. These variations have resulted in some improvement, but the gain is not enough to make resistance welding competitive with fusion

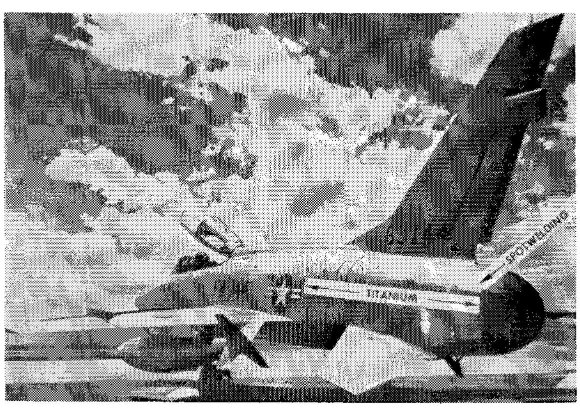


Fig. 12 Titanium structure, F-100.

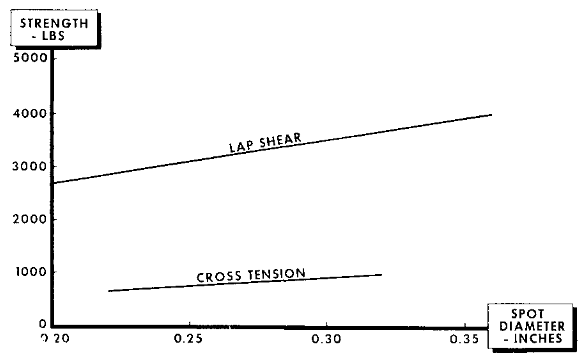


Fig. 14 Effect of nugget diameter on strength of 8Al-1 Mo-1V titanium resistance welds.

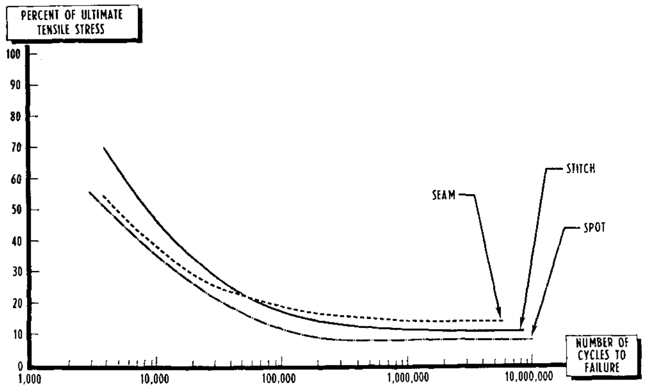


Fig. 15 Fatigue strength of spot, seam, and stitch welds in 8Al-1 Mo-1V titanium.

welding or mechanical joints for fatigue critical structure such as the lower wing panels.

Titanium can be readily fusion welded, provided attention is given to two potential problem areas: 1) contamination of the weld area by absorption of interstitial elements such as oxygen, nitrogen, and hydrogen and 2) material phase transformations resulting in embrittlement, due to thermal cycles induced during welding. The latter problem is controlled primarily by the proper selection of the titanium alloys to be welded. The 8Al-1Mo-1V and 6Al-4V alloys being considered for the SST are high aluminum bearing, or alpha-rich-type alloys, which are suitable for fusion welding on the basis of not being subject to significant transformation hardening or embrittlement during the thermal cycle from welding. The structural efficiency of as-welded joints in the annealed 8Al-1Mo-1V titanium alloy compare very favorably with the strength of PH14-8Mo stainless-steel joints in either the as-welded or postweld heat-treated condition. This comparison is shown in Fig. 16, using the tensile ultimate strength of the joint divided by its density as a measure of efficiency. Examination of the data also indicates the advantage of using an annealed material such as the 8Al-1Mo-1V titanium alloy, as compared to heat-treated stainless steel, from the standpoint of not losing parent metal strength in the weld-joint area. This eliminates the necessity of heat treating after welding or increasing the thickness of material in the weld area. The fatigue properties of fusion-welded joints in the 8Al-1Mo-1V titanium alloy are shown in Fig. 17. The data indicate adequate fatigue strengths for the welded titanium alloy as compared to the parent metal unnotched fatigue strength. It is interesting to note that the reduction in fatigue strength due to welding is approximately equivalent to the reduction for a notched condition in the parent metal where $K_t = 2.5$. This means that the fatigue strength of a fusion-welded joint in 8Al-1Mo-1V titanium is roughly equivalent to that of a riveted joint.

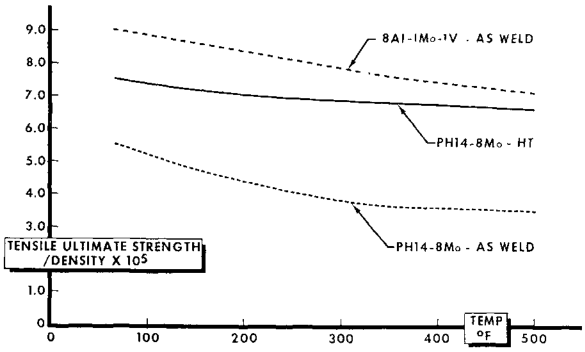


Fig. 16 Fusion weld joint efficiency of 8Al-1Mo-1V titanium alloy vs PH14-8Mo stainless steel.

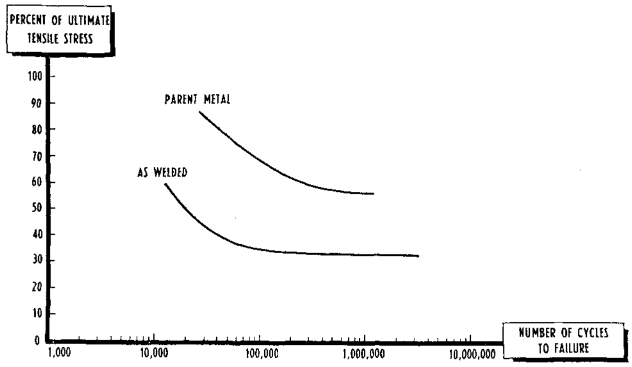


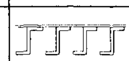
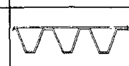
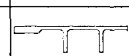
Fig. 17 Fatigue strength of 8Al-1 Mo-1V titanium fusion welds.

Cost-Weight Analysis

The selection of materials for the SST airframe structure cannot be divorced from the selection of the structural configurations for which the material is being considered. The choice of material and the structural concept should be evaluated concurrently in order to arrive at the best selections from the standpoint of producibility and cost-weight analysis. The environmental conditions imposed on the SST airframe resulted in the necessity to consider several materials, structural concepts, and fabrication techniques. This consideration includes many combinations of materials and concepts, which appear to offer structural advantages but which must be also cost-evaluated before their true worth can be known. It is necessary to utilize a method of equating the advantages, usually measured in terms of weight, with the cost of achieving them in terms of fabrication costs.

A method of making a cost-weight analysis has been developed at NAA and has been applied to studies on SST structures. The first step in the procedure is to establish the dollar value of saving a pound of weight in the design of a proposed SST configuration. This determination is based upon a modification of the standard Air Transport Association direct operating cost formula, and also includes such variables as the airframe weight-growth factor, engine sizing, and cost of financing the purchase of the new aircraft. The results of such a study on a particular SST configuration indicated that weight saving could be bought at a rate up to \$300/lb with a resulting lower total operating cost of the aircraft over its lifetime, including depreciation. Using \$300/lb as a common denominator, it is possible to evaluate the various concepts and materials in terms of cost-weight tradeoffs.

Examples of some initial tradeoff studies on SST structural concepts are presented in Figs. 18 and 19. Estimated fabrication costs and unit weights of the illustrated wing panel and wing spar concepts, designed for the same load, are

		MATERIAL		
CONCEPT NO.	CONCEPT	STEEL	TI	AL
1		\$133/FT² 3.03 LB/FT²	———	\$10/FT² 4.57 LB/FT²
2		\$120 FS/FT² 3.56 LB/FT²	\$170/FT² 2.67/LB FT²	\$60/FT² 3.32 LB/FT²
3		\$91/FT² 3.6 LB/FT²	\$130/FT² 2.76 LB FT²	\$55/FT² 3.97 LB/FT²
4		———	\$190 FT² 2.97 LB/FT²	\$67/FT² 3.68 LB/FT²

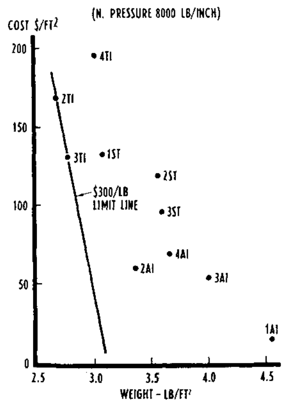


Fig. 18 Comparison: wing panel structure.

CONCEPT NO.	CONCEPT	MATERIAL		
		AL	TI	STEEL
1 BURN THROUGH SINE			\$100/FT 5.0 LB/FT	\$55/FT 6.7 LB/FT
2 NODE ATTACH SINE		\$14/FT 6.1 LB/FT	\$118/FT 7.0 LB/FT	\$34/FT 9.0 LB/FT
3 BEADED WEB		\$15/FT 6.1 LB/FT	\$91/FT 5.9 LB/FT	\$40/FT 8.5 LB/FT
4 FLAT WEB STIFFENED		\$15/FT 9.0 LB/FT	\$133/FT 8.1 LB/FT	\$56/FT 11.8 LB/FT
5 SANDWICH		\$25/FT 6.6 LB/FT	\$220/FT 5.6 LB/FT	\$375/FT 7.3 LB/FT

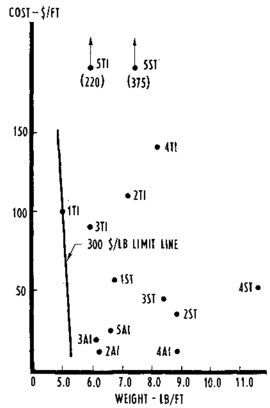


Fig. 19 Comparison: 30-in. spars.

shown. The panel concepts were designed for an axial compressive force of 8000 lb/in. and the spars were designed for a shear flow of 3000 lb/in., which plotted on the graphs, are points representing the cost in dollars-per-square-foot and weight in pounds-per-square-foot for the concepts studied. A sloping line may be drawn anywhere on this chart representing a \$300/lb gradient. All points falling on such a line are equivalent economically, trading off cost and weight at \$300/lb, whereas points falling above and to the right of the line indicate less desirable concepts. Points falling to the left and below the tradeoff line are, of course, more desirable as they come closer to the origin where the cost and weight are the lowest. In the figures shown, the tradeoff line has been drawn through the most desirable construction in each case, with all other points lying above and to the right of the line.

The examples discussed previously were initial studies and, as such, were limited to simple, noncontoured sections or panels without attachments. Later studies would be conducted to include added costs for additional complexities necessary, since the concepts selected become more defined as actual structure. By the use of techniques such as these, structural concepts and material selections can be analyzed on a continuing basis, and the results become an important factor in structural design decisions.

Summary

In summary, it can be said that suitable materials exist for making a highly efficient airframe structure for the SST. These materials can be expected to provide a long-life structure requiring a minimum amount of maintenance. Suitable fabrication processes are available for the building of an airplane from the materials most likely to be selected for the SST. Finally, techniques are available to assist the designer in making an optimum selection of materials and structural concepts in order to obtain an economical and reproducible airframe structure.

References

[1] Bochrath, G. E. and Glassco, J. B., "Fracture toughness of high strength sheet metal," Douglas Missile and Space Systems Div. Engineering Paper 1607 (March 1963).

[2] Posner, D. L., "Fracture toughness and tear tests," Boeing-North American Joint Rept. ML-TDR-64-238, Contract AF33-(657)-11461 (October 1964).

[3] Minkler, W. W., "A brief discussion of saline stress corrosion of titanium alloys in relation to trisonic transport materials requirement," Titanium Metals Corp. (December 4, 1961).

[4] Boeing Airplane Co. Data Reported to Battelle Memorial Institute for NASA Special Committee on SST (January 22, 1964).

[5] Avery, C. H., Turley, R. V., Schmid, F. R., and Simpson, R. A., "Screening test program for evaluation of the stress corrosion susceptibility of alloys under consideration for application as skin material," Douglas Aircraft Co. Rept. 31421, Contract AF33-(657)-8543 (April 1, 1963).

[6] Raring, R. H., "SST materials," Astronaut. Aeronaut. 2, 54–59 (September 1964).