STRUCTURAL DESIGN PHILOSOPHY FOR THE SUPERSONIC TRANSPORT

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INTRODUCTION

The Lockheed Aircraft Corporation is currently participating in the National Supersonic Transport Program under study and development in the United States. In response to the original Request for Proposal from the Federal Aviation Agency, Lockheed proposed in January 1964 a fixed, double-delta-wing airplane designed around the fundamental goal of maximum productivity and optimum airline economics for operation in the post-1970 time period. The general arrangement of this airplane is shown in Fig. 1. Thorough analysis of alternative means of attaining the objectives led to the combination of a large fuselage capable of housing up to 221 passengers, a high cruising speed of Mach 3.0, and the large fixed wing. It is the purpose of this paper to deal in depth with the basic structural design philosophy being applied to this airplane.

The recommendation of a high cruising Mach number was based upon fundamental considerations of optimum range, payload, and direct operating cost together with a high flight altitude for minimum sonic boom. The Mach 3.0 speed is recommended with full understanding of the progressively more hostile environment in the high-Mach-number flight regime, a factor which has been accounted for in these analyses, both with

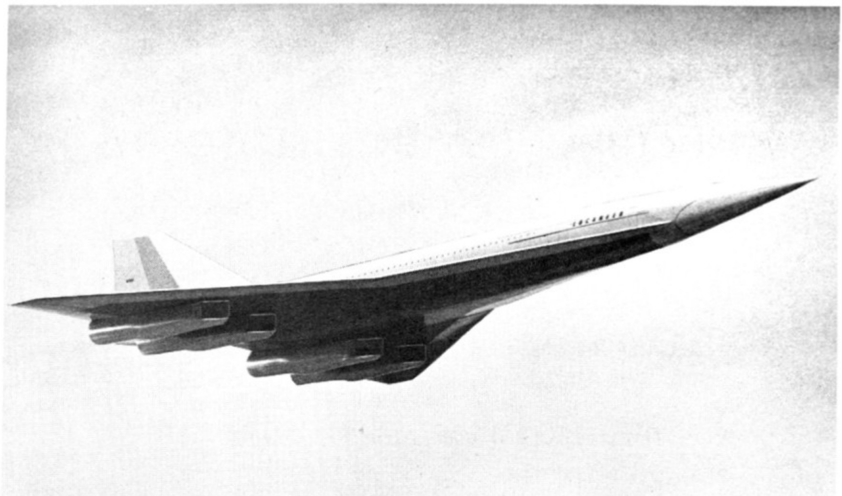

Figure 1. Lockheed Double Data supersonic transport.

respect to development time and cost. Materials of construction are being selected in detail to meet the requirements of this environment while maintaining or improving the levels of safety and reliability which have been developed in modern transport operation. It is not the purpose of this paper to discuss detail design nor the detail selection of materials in individual areas, but it should be stated at the outset that the principal materials of construction will be of the titanium family in a number of alloys as suited to individual applications.

To attain the economic, reliability, and safety objectives of the supersonic transport, a well defined and proven structural design philosophy is being applied. This philosophy has evolved from more than 30 years of design, production, and airline operation of transport aircraft throughout the world. Although the environment is new and new materials will be employed, the fundamental approach is the same one which has contributed to successful airline operating aircraft over this time period.

The five principal elements of the supersonic transport structural design philosophy are (1) the use of rational design load criteria, (2) the attainment of unlimited service life, (3) the attainment of an airframe having fail-safe characteristics with respect to both strength and stiffness, (4) recognition from the outset of the role of inspection and maintenance in structural design, and finally (5) an extensive ground and flight structural proving effort. The remainder of this paper describes the approach being taken in each of these five areas.

STRUCTURAL CRITERIA

The basic structural design philosophy requires that criteria be responsive to the operating environment and novel characteristics of the supersonic transport and, at the same time, provide a level of safety at least equivalent to that of current transports. The structural airworthiness requirements of the U. S. Civil Regulations (CAR-4*b*) are known to have provided satisfactory levels of strength for current civil transports. However, in those flight regimes where the operating environment or the response characteristics of the supersonic transport differ markedly from earlier transports, it is important that the adequacy of past criteria be reviewed. Some of the areas requiring such consideration are discussed below.

FLIGHT PROFILE

The first element of the rational approach to structural criteria for design of the supersonic transport involves the actual flight profile. Significant variations in flight profile are anticipated for different operating ranges, routes, and because of air traffic control requirements. Nevertheless, the flight profile shown in Fig. 2 for a range of 3,500 nautical miles is representative and can be used to demonstrate critical aspects of the planned operation.

The supersonic transport is characterized by large weight changes due to fuel burnoff during flight. For example, the gross weight may reduce as much as 70,000 lb from takeoff to the start of high-Mach-number cruise. During the cruising portion of the flight, over 100,000 lb of fuel may be burned off. These large variations in gross weight combined with the large changes in Mach number, altitude, and environmental temperature suggest

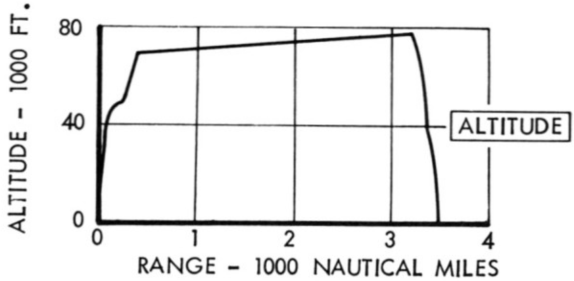

Figure 2. Typical design flight profile.

that rational combinations of weight, speed, and altitude should be considered rather than arbitrary ones. Accordingly, structural design weights are selected for specific altitude and flight conditions with due consideration being given to possible variations from these values for a variety of operating conditions.

DESIGN SPEEDS

The selection of design cruising, V_C, and dive, V_D, speeds is made with consideration of such factors as structural weight, flutter, sonic boom overpressure, airplane performance, and inadvertent overspeed. For the supersonic transport, the selection of these design speed values is a particularly sensitive matter, and the values shown in Fig. 3 are but representative of the factors involved.

At subsonic speeds, V_C must be high enough to permit economic climb and cruise performance, but low enough to avoid excessive weight penalties

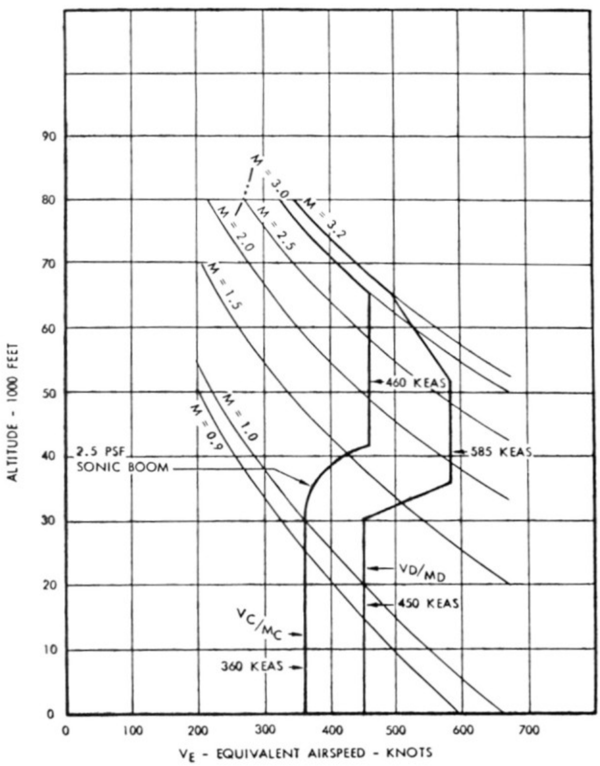

Figure 3. Design speed—altitude relationship.

due to gust loads. At intermediate altitudes, some increase in V_C leads to decreased climb fuel, but must be balanced against the weights involved in demonstrating aeroelastic and flutter characteristics with an adequate dive speed margin. In addition, adequate margin must be provided for climb and acceleration along different sonic boom paths so as not to compromise the eventual operational flexibility of the airplane. Finally, at high altitude, the V_C becomes related to the design cruising Mach number and must adequately cover the needs of the airplane to realize optimum long range cruise capability.

Of particular importance and interest are the speed margins between the design cruising speed and/or Mach number values and the dive limits. Margins which are felt to be adequate for a variety of rational conditions are shown in Fig. 3. Typical inadvertent overspeed conditions which have been investigated include a 7.5-degree nose-down upset from cruise conditions held for 20 sec, autopilot "hard-overs," trim system malfunctions, passenger movement, level off from climb with delay in power reduction, and level off at low altitudes while crossing airways with delay in power reduction. For all of the conditions considered, it has been found that a dive speed margin of 0.2 Mach number above the cruising speed is adequate for cruise operation. At lower altitudes for various conditions during both ascent and descent, the minimum overspeed margin requirements are of the order of 100 knots equivalent airspeed.

TEMPERATURE CRITERIA

Steady-state and transient temperature conditions and the aerodynamic characteristics of the airplane appropriate to each particular flight condition are considered for all loading conditions. Both normal and emergency conditions are considered in the flight profile analysis which determines the appropriate structural temperatures and loads. To protect the airplane from hotter-than-standard day conditions, a total temperature placard is established. For example, this placard, T_{MO}, might be defined as 650°F, corresponding to Mach 3.0 under U. S. Standard atmosphere conditions at 80,000 ft. This placard is shown in Fig. 4.

The structure is designed for fatigue, fail-safe, and ultimate conditions for the equilibrium and transient temperatures along the V_C/M_C versus altitude profile. The climb segment is conducted at maximum climb power for all gross weights from minimum to maximum in order to produce the maximum heating rates and greatest thermally induced stresses. Loads due to conditions of limit severity (pitch, roll, and yaw maneuvers, gusts, engine failure, etc.) are combined with thermally induced stresses and temperatures at any point on the flight profile. For maximum rates of cooling,

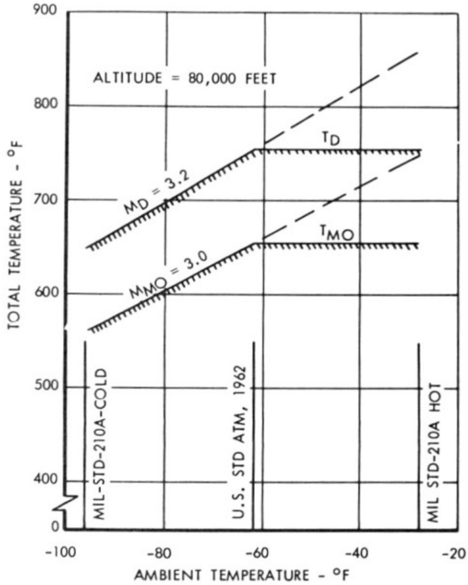

Figure 4. Design temperature limits.

emergency descent along the V_C/M_C boundary is assumed from any point along the design flight profile.

In addition, the design provides for the transient temperatures and loads resulting from inadvertent overspeeds to V_D/M_D. The initial condition is assumed to be flight along the V_C/M_C boundary with initial attitudes, power settings, weights, and temperatures appropriate to the particular point in the flight profile where the overspeed is assumed to occur. Times to reach dive speed and to return to the V_C/M_C placard boundary are a function of the type of upset condition. Shortest times to accelerate to $M = 3.2$ have been found to occur due to leveling-off from climb with lag in power adjustment. The time at V_D/M_D is assumed to be 10 sec and loads of limit severity are assumed to occur during the overspeed.

The combination of load stress and thermal stress due to temperature differentials is fundamentally simple. Thermal expansions in a statically determinate structure will create deformations without corresponding resisting loads. Only a redundant or restrained structure will develop resisting loads. The total thermal expansion of elements is relaxed and divided among surrounding elements according to the flexibility of the total structure. Thermal stresses are, therefore, the result of partially relaxed strains in the redundant structure. There is no resultant external load; the internal force systems at any cut section are entirely self-equilibrating. On the other hand, external load conditions provide the initiating stresses and result in strains in the elements of the structure. The distinction is important for the prediction of the ultimate strength of the structure where plasticity or yielding of the material must be accounted for rationally.

The approach currently in use for the supersonic transport is that the combination of effects from external loads and temperature differentials be based on the sum of strains rather than stresses. The design stress is obtained from the stress-strain curve for the appropriate temperature, corresponding to the sum of these strains. The approach is graphically illustrated in Fig. 5. The specific criteria is as follows:

1. For ultimate tension or compression conditions, a factor of safety of 1.25 is applied to calculated thermal strains when load and thermal strains are additive. No factor is applied to thermal strains when they are relieving the load strain.
2. For fatigue-limited tension conditions and for fail-safe conditions, no factor is applied to thermal strains.

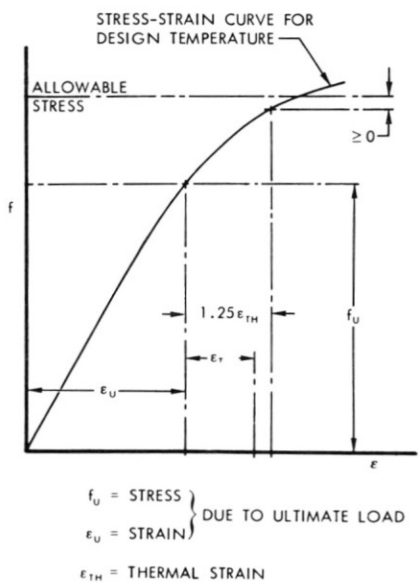

Figure 5. Thermal stress accountability—ultimate conditions.

GUST LOADS

Existing discrete gust criteria are known to provide satisfactory structure for aircraft similar in configuration, flight profile, and design envelope utilization to current aircraft. For an advanced vehicle such as the supersonic transport, however, it is important that the adequacy of these criteria be reviewed, utilizing the more realistic continuous random description of atmospheric turbulence and analysis techniques now available, to assure that a level of strength consistent with that of past aircraft is maintained. Experience has shown it to be particularly important to utilize a continuous random turbulence description for gust-loads work if the damping in either the airplane short-period modes, or in any of the elastic modes, is significantly different from that of airplanes whose operational experience was used to establish current gust criteria.

To accomplish this purpose, design load levels for the wing, fuselage, and vertical tail are defined statistically by means of a power-spectral flight profile analysis, using the general procedure and atmospheric model of NACA TN 4332. The basic philosophy in this work is to provide a level of strength such that, per flight hour, the probability of exceeding limit strength is no greater for the supersonic transport than for present commercial transports having proven adequacy for gust-induced loads. The Model 749 Constellation is representative of such existing aircraft. The supersonic transport power-spectral flight profile analysis provides curves of frequency of exceedance versus load level for any desired load quantity. Entering these supersonic transport curves at the limit load frequency of exceedance established for the Model 749 then establishes the limit design strength level required for supersonic transport loads.

Wing root-bending results for the supersonic transport are shown in Fig. 6. Statistical curves are shown separately for various portions of the flight profile. These separate curves, when added vertically, give the total frequency of exceedance. The most striking result shown by this figure is that the supersonic transport wing will be exposed to far higher gust loads during subsonic climb than during the other portions of the flight. As noted previously, however, the subsonic climb speed has been selected low enough to prevent the gust loads from exceeding the 2.5-g wing maneuver loads. To place in perspective the gust criteria resulting from the flight-profile analysis, it is of interest to note the discrete gust velocity (U_{de}) that would be necessary, on a static load basis, to provide the same wing-strength level is 69 fps, as compared to the current civil criteria value of 50 fps at V_C.

MANEUVER LOADS

The supersonic transport is designed to a positive maneuvering load factor of 2.5 at all speeds from the design maneuver speed (V_A) to the

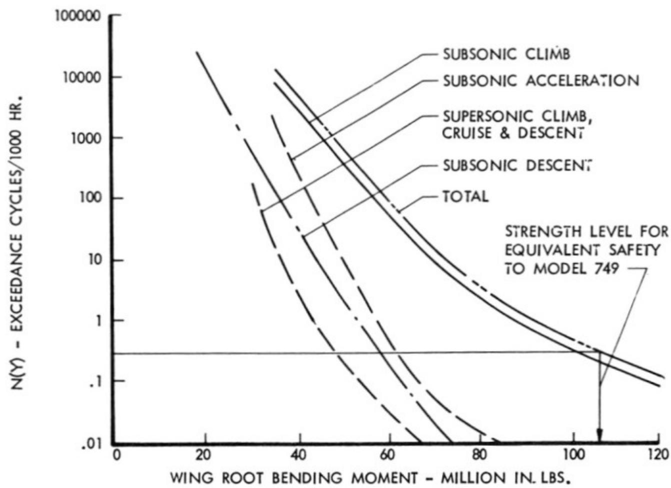

Figure 6. Power spectral gust loads.

design dive speed (V_D). The negative maneuvering load factor is -1.0 at all speeds up to V_C and varies linearly from -1.0 at V_C to zero at V_D. This is in accordance with current U. S. civil requirements. Typical speed-load factor diagrams at six altitudes and representative weights are shown in Fig. 7. These diagrams are superimposed on the design speed-altitude diagram. At each altitude the diagrams show both the minimum and the maximum design weights at the particular altitude. The low-speed portion of these V-n diagrams is defined by a stick-shaker warning device and does not represent a physical stall speed. The warning device is incorporated to limit angle of attack to appropriate values in the different operating speed regimes. Experience with low-aspect-ratio delta wings establishes the fact that no buffet-boundary limits are anticipated within the V-n diagrams shown. This capability provides added flexibility in selecting cruise altitudes to alleviate sonic boom, and freedom for the operator to select transonic acceleration altitudes.

The control system, inertia characteristics, aerodynamic characteristics, flight profile, and speed range of the supersonic transport are sufficiently different from those of current transports to raise questions as to the adequacy of existing civil criteria for abrupt pitch, yaw, and roll maneuvers. Therefore, the time history response to rationally defined control inputs and/or engine failures are determined at appropriate altitudes to evaluate the response of the airplane in the subsonic, transonic, and supersonic speed regimes for all points within the design V·n and weight-center-of-gravity envelopes. Results of these analyses have shown that the supersonic transport is not dependent on stability augmentation to keep the design loads for transient roll, pitch, yaw, and engine failure conditions to acceptable levels.

LANDING CRITERIA

In order to obtain a preliminary insight into the adequacy of current civil aircraft standards for design sinking speed of the supersonic transport, use has been made of two parameters suggested in Part II of NASA TN D-1392 as possible explanations for the difference in operational sinking-speed statistics of subsonic jets and piston aircraft. These are an elevator effectiveness parameter, and a parameter roughly measuring the pilot's height above the ground at main gear contact. Values of the sinking speed exceeded once in one hundred landings are plotted versus these parameters

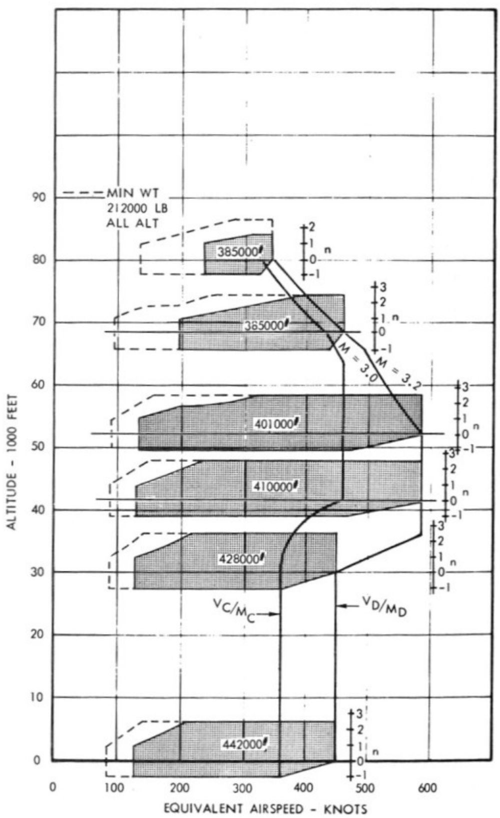

Figure 7. Typical design (V-n) and weight relationship.

for various existing aircraft on Figs. 8 and 9 with values for the supersonic transport superimposed. These results indicate that supersonic transport sinking speeds are likely to be essentially the same as for current subsonic jets. It should be noted that, even if supersonic transport sinking-speed statistics should prove to be greater than those of current jets, it is not apparent that the design level of 10 fps should be raised. This is because current jet statistics show that a sinking speed of 7.5 fps is exceeded only once in 10,000 landings. Supersonic transport sinking-speed statistics could be as much as 33 percent above current jet sinking speeds, yet the sinking speeds reached once in 10,000 landings would still be only 10 fps. This exceedance rate would not appear to be unreasonable for limit design.

A nose landing gear criteria which reflects the airplane's pitch-plunge motion during main and nose gear impact has been employed. The criteria assumes main gear impact at the limit sink rate of 10 fps in the maximum nose-up attitude, most forward center of gravity, and further assumes no pilot action during the contact of the main gear and settling of the nose gear on to the runway. Figure 10 shows the results of such an analysis. In the example shown, it can be seen that the sinking speed of the nose gear is 10 fps at main gear contact, reducing almost to zero as the main gear arrests the vertical descent of the airplane, and then gradually increasing as the airplane rotates until the nose gear contacts at a sinking speed of 7 fps. It is of interest that the resulting nose gear vertical load is less than half that resulting from dynamic taxi.

TAXI CRITERIA

The taxi loads for the supersonic transport are based upon proven criteria and experience with subsonic aircraft. The supersonic transport will, of necessity, have to operate in the same general environment as

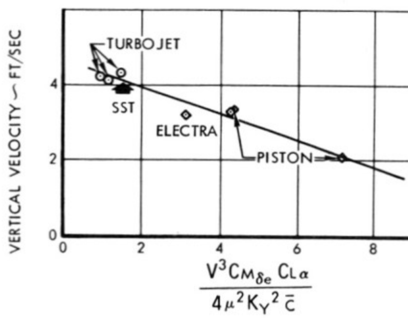

Figure 8. Elevator effectiveness—landing rate of sink relationship.

subsonic jets during takeoff, landing, and taxiing on rough runways or taxiways. There is no significant difference in landing or takeoff speeds; and, because taxiways are generally rougher than runways, the taxi speed on taxiways will continue to be limited by the pilot's judgment based on

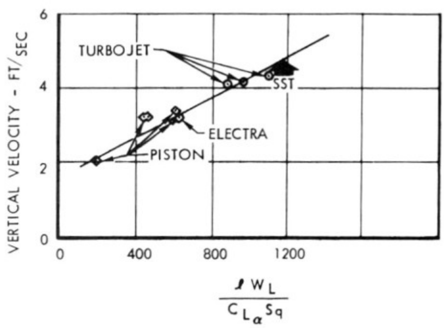

Figure 9. Effect on landing rate of sink of pilot's eye height.

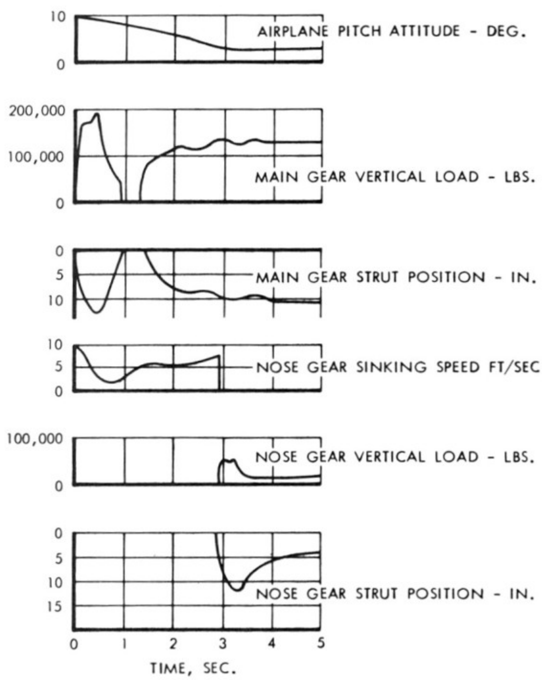

Figure 10. Landing time history.

roughness of the ride. Assurance of the validity of applying criteria developed for subsonic aircraft to the supersonic transport has been obtained by analog analyses in which the supersonic transport is taxied over known runway profiles and compared to existing aircraft. Pilot compartment accelerations determined in this manner for the supersonic transport are comparable to those for subsonic airplanes and are not great enough to cause difficulty to the pilot in the takeoff run or to limit taxiway speeds below those of existing aircraft.

FLUTTER, DIVERGENCE, AND CONTROL CRITERIA

The design approach to flutter safety is based on the requirements in Civil Aeronautics Manual 4*b* with the objective of assuring that the airplane is free from flutter of wing, vertical fin, and control surfaces, and free from panel flutter at all speeds up to 1.2 V_D. Fail-safe criteria include flutter considerations in that no single failure will cause flutter at any speed up to V_D. All parts of the airplane will be demonstrated in flight to be free from flutter and excessive vibration under all speed and power conditions appropriate to the operation of the airplane up to at least V_D.

The design speed envelope previously described contains four regions which may potentially design the airplane or some of its elements for flutter and aeroelastic considerations. At low altitude, with the aerodynamic center located most forward, certain flutter modes may be critical. Under transonic conditions, maximum values of lift curve slope occur indicating another potentially critical area. The other two potentially critical conditions occur along the dive-speed boundary under conditions of maximum dynamic pressure but with various conditions of lift curve slope and structural temperature. In addition to these potentially critical areas of flight, certain configuration features, such as the location of the wing on the fuselage and the engines on the wing, present the possibility of different modes than are common in transport aircraft today.

An extensive analytical program is being augmented by elastic model testing to give precise definition to flutter limits for both normal conditions and for conditions after single element failure. To date, analysis and testing indicate that flutter prevention on the double-delta configuration does not introduce unusual structural stiffness requirements or complications, and the anticipated development program including eventual flight flutter demonstrations will positively establish the airframe integrity.

UNLIMITED SERVICE LIFE

After the selection of rational load criteria based upon the intended usage and environment, the next element of the structural design philosophy for the supersonic transport involves operational life. Here the design objective is to provide an airframe which will experience essentially unlimited life in normal airline duty. In specific terms, the phrase "unlimited life" means that with normal inspection, maintenance, and repair the operational life of major structural elements will be determined by inevitable economic obsolescence factors of the airframe and not by any requirement to retire such elements on the basis of accumulated hours. Not only is this structural design approach similar to that which has been used for the design of most current airline aircraft, but it also is required to substantiate the anticipated depreciation write-off periods visualized as fundamental to the economic operation of such an airplane. For example, operation of the airframe for 3,000 hr each year for a period of 15 years will accumulate 45,000 hr of service time. Such service lives are already relatively commonplace with numerous Lockheed Constellation airframes now nearing 60,000 hr of service on the airlines without replacement of primary structural elements. Achievement of a similar service life for the supersonic transport is essential and is adopted as an element of the structural design philosophy.

CREEP CONSIDERATIONS

Perhaps the first requirement for achievement of unlimited service life is to insure freedom from significant creep of materials under the load and temperature environments involved throughout the airframe structure. For the supersonic transport, similar criteria have been adopted as for previous transport aircraft. For example, there will be no requirement to replace major structural elements at specified time intervals, nor to retire the airframe at any arbitrarily defined lifetime.

Control of creep depends upon the proper choice of material for the intended operating environment, together with control of the permissible stress levels under operating conditions. Test results are already available to guide the selection of materials. Typical long-time, high-temperature test data on a number of materials are shown in Fig. 11. The creep stress-to-density ratio information shown indicates that the titanium alloy 8AL-1Mo-1V has a clear superiority over other titanium alloys and stainless steel. Inconel alloys have even greater creep resistance at higher temperatures, but the material density precludes its general use on the supersonic transport. Other alloys of titanium also produce excellent creep characteristics in the temperature range from 500 to 800°F, and comparison of the capability of 8-1-1 titanium with aluminum at much lower temperatures shows that use of proper materials can provide equal or less creep in the structure of a Mach 3 supersonic transport than can be expected at lower Mach numbers with aluminum.

A specific example of the application of titanium 8-1-1 to a critical hot element is shown in Fig. 12, which presents information for the engine inlet duct. In this area, short exposures to temperatures above 600°F are anticipated, but a very large total of a total service exposure of 50,000 hr, half or more, might be at temperatures near 600°F and at a stress level of the order of 40,000 psi. The figure shows the cumulative distribution of hours at stress and temperature levels for this critical component, and compares it with the stress levels producing 0.1 percent creep deformation after lengthy exposure. The figure demonstrates that substantial design margins can be provided, and significant creep can be eliminated in the supersonic transport airframe even in the hottest elements.

FATIGUE CONSIDERATIONS

With freedom from creep insured by the proper selection of materials and stress levels for each environmental area, the important considerations for unlimited service life become those of fatigue. To provide a service life unlimited by fatigue considerations, the supersonic transport will be designed by the same procedures which have been developed over the years on the Constellation series, the Electra, the P-3A, and the supersonic F-104. The fundamental steps are as follows:

1. A fatigue quality level is established. This quality standard is slightly more severe than a simple open hole in the uniform structure well removed from jcints or discontinuities, and all elements and joints are required to meet or better this standard.
2. Reduced design stress levels are utilized. When the quality standard described above is applied, ultimate tensile strength levels cannot be exploited, and reduced design allowable tension stresses are established by analysis and test.
3. Parts designed to meet the fatigue quality standard must be proved by test. All vital joints and structural discontinuities are required to demonstrate acceptable fatigue life in laboratory tests conducted on completely detailed specimens and using the full spectrum of expected service loads.
4. Estimated loads are confirmed by flight-load surveys. Direct flight test measurement of loads and temperatures are made to correlate with estimated loads and to provide a basis for complete airframe fatigue testing.
5. Full-scale airframe fatigue tests will be performed. These tests will be based upon load spectra verified by flight test, and will be performed on a flight-by-flight basis including the fatigue-sensitive ground-air-ground cycle.

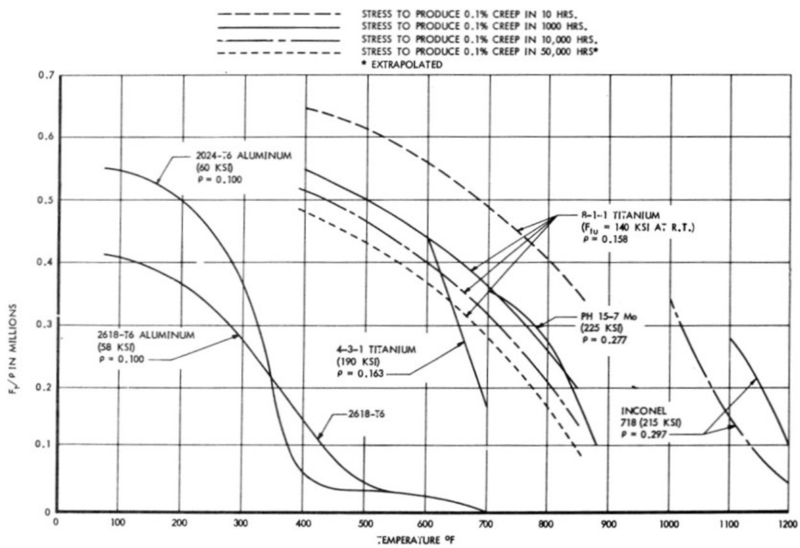

Figure 11. Creep strength to density ratio for sheet alloys.

This approach to design for long fatigue life results in a comprehensive treatment of the airframe element by element, with redesign and retest required where unanticipated failures are encountered.

To achieve adequate fatigue life under the approach described, a conventional approach to the prediction of fatigue effects is utilized, but important differences are introduced in the interpretation of these data. The conventional approach to the prediction of service life assumes that fatigue effects under different loading conditions are linearly cumulative and can be based upon a summation of conditions, each one of which is based upon *S-N* fatigue characteristics developed from constant-load-amplitude test data. These results produce estimates of service life for conditions resulting from various operational usage in terms of the ultimate design tensile stress and a stress concentration factor or fatigue quality index. To correct for the inadequacies of cumulative damage theory and the questionable effects of the ground-to-air cycle, flight-by-flight fatigue test data are utilized to select the proper fatigue quality index K for the airplane.

To accomplish such estimates of fatigue life, various flight profiles representing the anticipated actual applications of the aircraft to transport service are analyzed. Such important factors as short- and long-range flights and the use of the airplane on training flights where it ordinarily experiences somewhat more severe loadings than in normal usage are thus included. Finally, summations of the total life of the airplane involving different mixes of the anticipated flight profiles can be made to determine actual life estimates.

A typical example of the results of the conventional portion of the fatigue life analysis is shown in Fig. 13. The anticipated service life for a typical structural element such as an 8-1-1 titanium lower wing-spar cap is shown as a function of the ultimate tensile stress and fatigue quality index. Similar data are developed for the entire spectrum of the flight profiles anticipated and for other representative primary structural elements of the airplane. The question remaining to complete the specific life estimate is that involving the selection of a specific combination of stress and fátigue quality index.

To determine the proper fatigue quality index, flight-by-flight tests are performed on specimens with a simple hole and utilizing a loading sequence representative of actual use such as shown in Fig. 14. The results of these

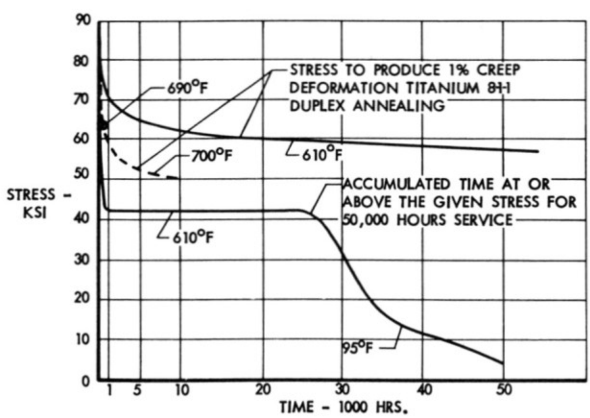

Figure 12. Engine inlet duct—accumulated stress temperature history in 50,000 hours of service.

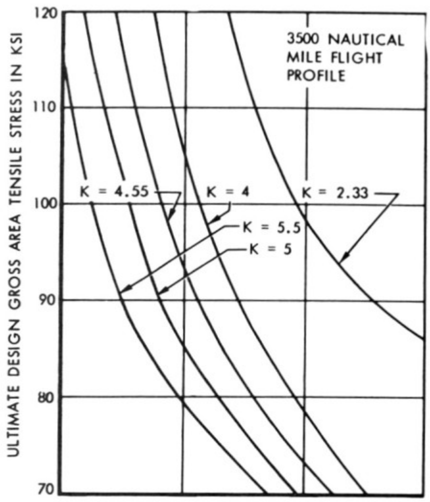

Figure 13. Calculated service life—3500 N. Mi. flight profile.

tests are analyzed by the simple cumulative damage hypothesis in combination with S-N data obtained from constant-amplitude tests. Results show that accumulation of fatigue damage in actual sequential loadings, including ground-air-ground effects, results in allowable stress levels less than those predicted from the constant amplitude data. An example of this is shown in Fig. 15, where a fatigue quality index of 4 for coupons with simple holes has been found to be representative of characteristics resulting from the actual flight-by-flight load sequences. This is significantly more severe than the data resulting from constant load-amplitude tests on simple hole fatigue specimens resulting in a value of K of 2.76.

From testing of this type on the materials of interest for the supersonic transport, a quality index based upon a simple hole in material removed from a joint area would have a minimum value of $K = 4.0$. As indicated in the design approach to fatigue life outlined above, the quality index adopted for design is intentionally selected to be somewhat more severe than the simple hole, and an index of 5 is expected to be both realistic and attainable in actual airframe structure. Therefore, this quality index is used as a basis for the structural life estimates.

Results of typical service life calculations utilizing this approach are shown in Fig. 16. Various flight profiles, including pilot training and check, are considered, and estimates are made for various anticipated combinations of the flights. An ultimate design gross area tensile stress of 90,000 psi for this particular material was selected in conjunction with the fatigue quality index $K = 5$ as a basis for the life estimates. The results shown predict essentially unlimited life for all of the composite flight spectra examined. These results provide justification for the selection of an ultimate design tensile stress of 90,000 psi for the wing structure in 8-1-1 titanium, and also indicate a basis for unlimited fatigue life of the actual airframe at this stress level.

To provide further substantiation of the fatigue characteristics of the airframe under actual operation, extensive research programs under joint FAA-Lockheed sponsorship on the fatigue characteristics of duplex annealed 8-1-1 titanium, precipitation hardening 14-8 steel, and nickel alloy Inco 718 are in progress. Over 6,000 specimens are being evaluated under various conditions of temperature and stress exposure, and the effects of various types of contaminants on these materials are being determined. In addition, specimens are being subjected to real-time flight-by-flight testing in which the time at temperature during each flight is approximately one hour.

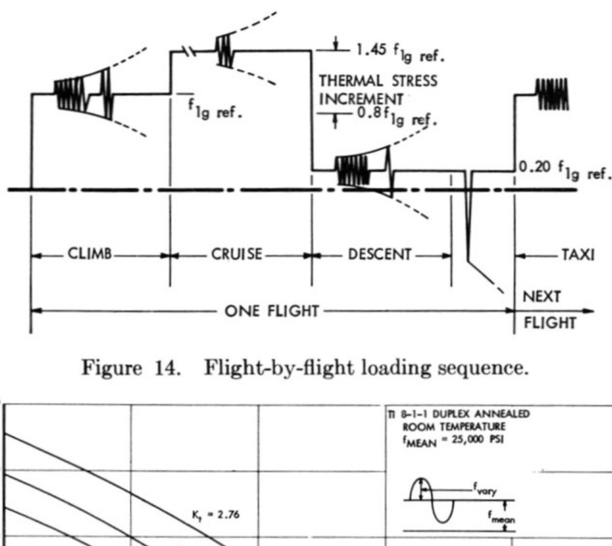

Figure 14. Flight-by-flight loading sequence.

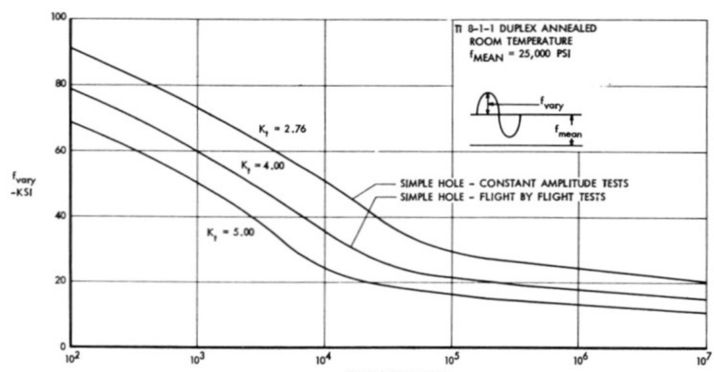

Figure 15. Typical constant amplitude fatigue data.

		Composite No. 1		Composite No. 2		Composite No. 3	
Flight Profile	*Fraction of Calculated Life Utilization in 1000 Hrs. of Service*	*Per Cent of Total Time in Composite Service Use*	*Fraction of Calculated Life Utilization in 1000 hrs. of Service*	*Per Cent of Total Time in Composite Service Use*	*Fraction of Calculated Life Utilization in 1000 Hrs. of Service*	*Per Cent of Total Time in Composite Service Use*	*Fraction of Calculated Life Utilization in 1000 Hrs. of Service*
3500 Nautical Mile	.01613	25	.00403	30	.00484	10	.00161
2100 Nautical Mile	.00625	40	.00250	50	.00312	50	.00312
1200 Nautical Mile	.00667	29	.00193	14	.00093	20	.00133
350 Nautical Mile	.00050	4.5	.00002	4.5	.00002	18.5	.00009
Pilot Training & Check Flight	.00588	1.5	.00009	1.5	.00009	1.5	.00009
Fraction of Life Utilization in 1000 Hrs. of Service			.00857		.00900		.00624
Calculated Service Life			116,686 Hrs.		111,111 Hrs.		160,256 Hrs.
Average Flight Time			1.59 Hrs. (95 Minutes)		1.71 Hrs. (103 Min.)		1.44 Hrs. (86 Minutes)

Figure 16. Fatigue life estimate for composite service use of supersonic transport.

FAIL-SAFE DESIGN

The objective of the fail-safe criteria is to insure the structural integrity of the airframe in the event of the failure of a single primary structural element or the occurrence of partial damage in extensive structure such as skin panels. Such failures or damage may arise from unreported accidental impact, minor collisions, turbine-blade penetration, small-arms fire, or other sources as well as from cracks initiating from fatigue. Under such conditions, integrity of the structure must be maintained for both strength and stiffness requirements. Fail-safe strength and stiffness are provided at limit-design load level in all normal design flight, takeoff, and landing conditions. These levels exceed current U. S. Civil Air Regulations requirements by approximately 25 percent. The approach to achieving the fail-safe objective is through the design process in which careful attention is given to the application of redundant structural members, appropriately located barriers such as splices, and selection of reduced stresses in critical areas in which slow rates of crack growth must be maintained if such should occur.

REDUNDANT STRUCTURE

The double-delta wing provides an excellent example of the use of redundant structure to attain fail-safe objectives. The wing spars are closely spaced and, on the tension side of the wing, the spar caps are in multiple pieces so that damage to one piece will not propagate into another.

Should any spar cap and adjacent skin fail for any reason whatsoever, the surrounding structure has sufficient reserve strength to carry the incremental loads. In addition, the multiple spars provide a very intimate connection between the wing and the fuselage. Features of the wing described above are illustrated in Fig. 17.

Mounting of the engines is another area in which fail-safe structural design philosophy is applied. The supersonic transport engines will be mounted by a link system which is tolerant to the failure of any one link. Figure 18 shows schematically the concept of this engine-mounting system. Under normal conditions, the front left mount takes side, vertical, and thrust loads. The front right mount has resilient features and normally carries no significant load in any direction. The aft top mount takes side load only, and includes secondary side snubbers, normally unloaded. Both aft side links take vertical loads only.

For failure situations, the front right mount is capable of absorbing the full limit loads in all directions with a limited motion of the engine. The aft top mount secondary snubbers are designed to carry limit side loads while restricting side motion of the engine. Either of the aft side links is sufficiently strong to carry the entire vertical load after failure of the opposite side link.

Various failures are provided for by this mounting system. For example, after a front mount failure, the other front mount takes over. If either aft side link should fail, the other side link carries vertical load and the resilient front mount comes into action and takes torque loads. If the aft top mount fails, the secondary snubbers come into action and carry side loads. In addition to these fail-safe characteristics, the mounting system provides self-aligning features for ease of installation, and thermal growth of the engine is permitted without restraint. Although fail-safe, the mounting system is not redundant and does not transmit airframe loads to the engine.

Windshields and windows are designed to stringent fail-safe requirements. As illustrated in Fig. 19, the windshield installation contains four panes. In addition, the windshield post is a multipiece construction so failure from one piece cannot propagate to the other. Protection for the pilot is provided for bird impact and strikes by four-inch hailstones at speeds up to V_C at sea level. With any one single-load-bearing element of the windshield installation failed, it can still withstand maximum cabin pressurization. With two elements failed, safe flight of the airplane can be maintained with a cabin pressure altitude of not more than 15,000 ft. The cabin windows are also a four-pane construction as illustrated in Fig. 19. They are designed to meet the same fail-safe requirements as the windshield insofar as they are affected by cabin pressure. In addition, the windows are designed with a small area (6-in. diameter, clear vision) to balance with cabin air-source capabilities so as to limit maximum cabin-pressure altitude to a safe value even in the remote possibility of a complete rupture of all elements of the window.

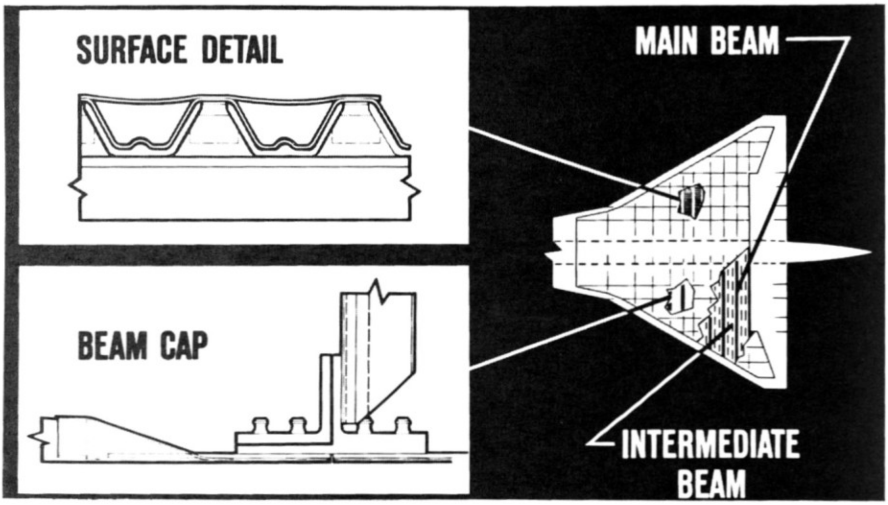

Figure 17. Wing construction.

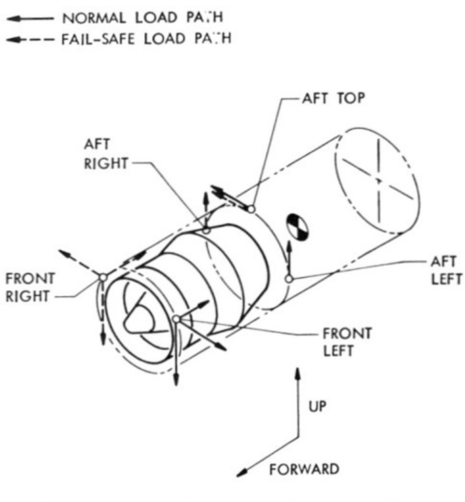

Figure 18. Redundant engine mounting.

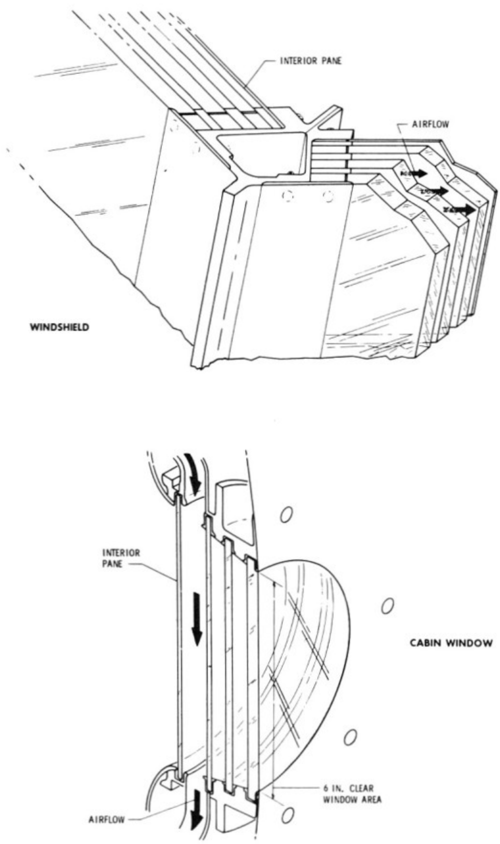

Figure 19. Windshield and cabin windows.

BARRIERS

The appropriate use of splices provides an excellent inhibitor of cracks or damage which may occur in the structure. As an example, wing skins are panelized at each of the main spars. Splices in critical areas of the fuselage will be provided to act as barriers to crack or damage propagation. The integrally stiffened structure of the fin is made from panels appropriately spaced so damage from one cannot propagate across the splice line into the adjacent panel.

REDUCED STRESSES FOR SLOW CRACK PROPAGATION

In extensive portions of the structure, fail-safe objectives are achieved through the appropriate selection of stress levels to maintain slow rates of crack growth. An excellent example of this type of structure is the skin of the fuselage. Pressurization stress in the duplex annealed 8-1-1 titanium skin is limited to 25,000 psi, not only to provide excellent fatigue characteristics but also to provide for a very slow rate of crack growth should such occur in the structure.

To substantiate the selection of this stress level, a fuselage panel was constructed approximately 5 ft by 10 ft of 8-1-1 titanium duplex annealed 0.030 skin material with 0.050 rings and 0.032 stringers representative of the supersonic transport construction. This panel is shown in Fig. 20. The curved panel, mounted on a flat test bed and shrouded for hot-air heating on the external surface, was cyclically loaded by internal air pressure to a pressure of 12.5 psi, stressing the skin primarily in the circumferential direction.

The test temperature ranged from 550–650°F. Initial damage was produced intentionally by a sawcut approximately 4 in. long, centrally located in the skin between two rings and two stringers. A true fatigue crack was produced at the ends of the sawcut at room temperature before testing at elevated temperature. This procedure was repeated twice to accelerate the test. After the crack in the skin had progressed substantially beyond the two adjacent rings, the redistribution of the load caused these two rings to fail, and the crack growth rate increased. However, after the crack had extended over three bays (crack length = 33.4 in., Fig. 20), the crack growth was arrested by the next pair of rings. The panel was then failed statically at room temperature. The test showed that, even with this extent of damage, the design configuration would withstand an internal pressure of 12.25 psi.

From the recorded test data, an estimate of the number of pressure cycles required to fail the titanium fuselage structure from any initial crack length can be made. For example, a 1-in. initial crack would require approximately 46,000 cycles, or flights, to reach unstable proportions; or a 10-in. initial crack would require approximately 2,100 cycles, or flights, to reach unstable proportions. The 46,000 flights to failure from a one-inch crack are of the order of the required lifetime of the supersonic transport, while the 2,100 cycles to failure from a 10-in. crack would approximate a year's flying. This test provides substantiation of the excellent fail-safe characteristics of titanium 8-1-1 duplex annealed material for the supersonic transport environment, and demonstrates the feasibility of design to low crack growth rates, thus insuring catching of such faults during routine inspections.

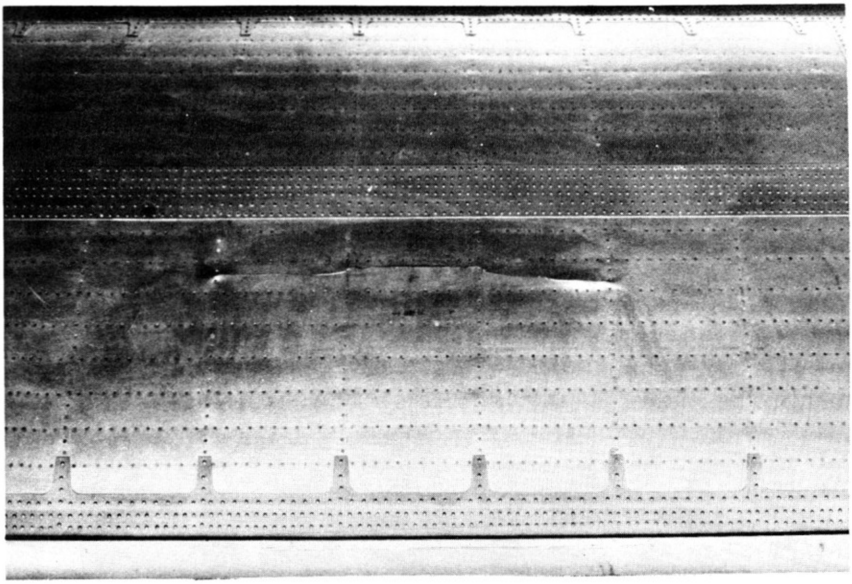

Figure 20. Fuselage crack investigation panel.

Figure 21 further illustrates the excellent crack resistance of titanium 8-1-1 duplex annealed when compared with aluminum and stainless steel. The rate of crack growth in stiffened structure of various materials is compared on a strength/density basis in which X is the crack length. At 550°F, comparable structure of titanium 8-1-1 duplex annealed has an appreciably slower rate of crack growth than the aluminum alloys used in current subsonic transports.

FAIL-SAFE TESTING AND ANALYSIS

The attainment of actual fail-safe structure utilizing the three basic techniques described above is implemented by a combined program of analysis and testing. The allowable strengths of damaged structures are determined experimentally for the specific structural configurations and external environment of the supersonic transport. Simple and successful empirical methods have been developed for translating experimental data into useful and accurate analytical methods for aluminum structures. These design procedures have been proved in the design, testing, and service record of current subsonic transports. Test programs are under way to provide the necessary experimental data for the materials and environment of the supersonic transport which are required to apply these analytical procedures. In addition, extensive tests of specific components will be performed to further validate the analysis. These tests include windows, windshields, components of the wing, and fuselage. These tests and analysis are the prelude to the full-scale tests described in a later section.

INSPECTION AND MAINTENANCE

The fourth fundamental ingredient of the supersonic transport structural design philosophy is based upon positive recognition of the fact that inspection and maintenance of structure must be considered from the outset. As has been discussed previously, inspection and repair are considered to be fundamental ingredients of both the fatigue and fail-safe aspects of the airplane. In this manner, inspection and maintenance contribute toward the achievement of the primary goal of unlimited service life. In addition to these aspects, consideration of inspection and maintenance requirements plays a large role in the selection of the basic structural configuration of major airframe elements and their detailed execution. For example, inspection and maintenance requirements are responsible for the basic decision to utilize essentially conventional, built-up aircraft structure for the supersonic transport, rather than to make extensive use of Honeycomb or other sandwich-type structures.

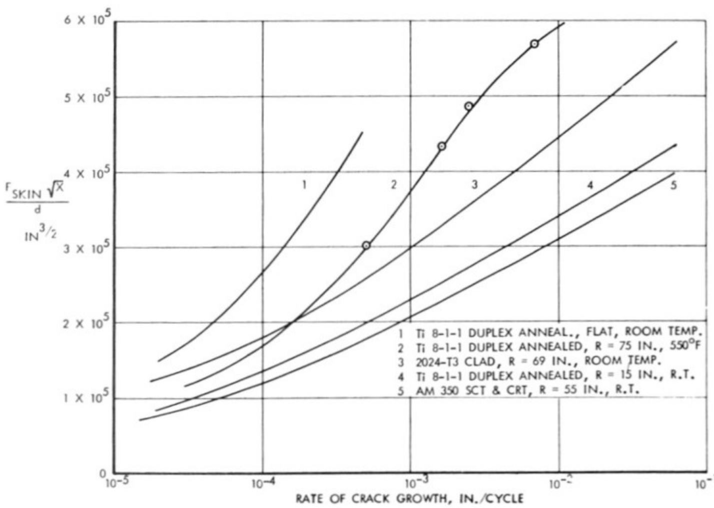

Figure 21. Rate of crack growth on stiffened structure.

The double-delta wing is well suited to the provision of adequate inspection capability. The extremely long root chord makes possible a structural depth of the order of 5 ft in the root region, making it possible to provide access to the interior of the wing where large cavities having all open-type structure are available for simple visual inspection. The use of integral fuel in the inboard portion of the wing in lieu of bladder or other fuel cells facilitates visual inspection in this critical area. In addition, the multiple spar construction of the wing utilizes open trusses in many areas contributing to both accessibility and visual inspection. The built-up skin construction is similar to that which has been used in previous transport aircraft and can be repaired by conventional techniques. Leading and trailing edge elements are both repairable and replaceable by detachment from the primary-wing box structure.

The fuselage is constructed in a conventional manner with a single skin combined with rings and stringers. The single-skin thickness is inspectable on the exterior, and the stringers are of open section and available for inspection upon removal of cabin trim and insulation bats.

The adoption of these essentially conventional and well-proven approaches to transport aircraft structural inspection and maintenance insure a maximum carryover of the knowledge and techniques developed from years of operation. The principal new element is the introduction of the new materials, an area which has minimum effect on airline procedures, experience, and cost.

STRUCTURAL PROOF APPROACH

The final element of the structural design philosophy for the supersonic transport is a very comprehensive test program. Flight-loads measurement, ultimate-strength tests, fail-safe strength tests, and fatigue tests will be conducted. The static and fatigue tests demonstrate the strength and life of the airframe for the loads imposed. The test loads will be verified by extensive and accurate loads measurements in flight and ground operations of the aircraft.

FLIGHT-LOADS PROGRAM

A flight-test program will be conducted on the first airplane with the objective of demonstrating the structural integrity of the supersonic transport. This program will include measurement of external applied loads and temperatures by flight-load survey methods; measurement of dynamic loads for various conditions such as rough air, landing, and taxi; and measurement of internal load distributions and detail stresses for both external applied loads and loads induced by structural heating. This program will culminate in demonstration-type maneuvers to limit conditions for selected maneuvering conditions. Airload survey instrumentation will include provisions for obtaining fuselage and wing loads with both pressure distribution measurements and calibrated load-measuring strain gage instrumentation.

All load measurements will be available to correlate measured loads with design loads before conducting the ultimate load static tests. Any adjustments in loads or load distributions required to make the static test more properly reflect the actual flight loads can be made prior to completion of the static tests. The airplane fatigue-test program starts late enough in the development program to allow full advantage to be taken of the structural flight test results.

A flight-flutter test program will be conducted with the objective of demonstrating freedom from flutter for any flight condition and airplane loading at speeds up to V_D/M_D. In addition, a substantial amount of data will be obtained to compare with design flutter data to enable verification that satisfactory flutter safety margins are obtained.

ULTIMATE STRENGTH TESTS

Ultimate strength tests and fail-safe demonstration tests are to be conducted on one airframe. These static tests will be conducted in the fatigue-loading apparatus. The reason for this is that the fatigue-test equipment has evolved into a very flexible, easy changeover system. Magnetic tape controlled hydraulic servo valves control the load distribution over the airframe. Thus, changing to a different loading condition is no longer the time-consuming project of shifting jacks, repiping, and recheckout. A new preprogrammed load-control magnetic tape is sufficient to accomplish the complete change to a new loading case. Testing more conditions with a considerable savings of time and expense is realized by this approach.

Heat will be introduced, where required, by program-controlled thermocouple monitored quartz lamps. Cooling water-fog and forced air distribution will remove heat at the required rate. Automatic lockup circuits are provided to save the test article in case of malfunction overload or failure in any portion of the test equipment or airframe. Automatic data readout and recording systems will be used for compatible loading and heating times. Selected channels will be monitored throughout the test.

FAIL-SAFE TESTS

Fail-safe tests will be conducted on the static test airframe on completion of the ultimate-strength series described above. These tests will be conducted both statically, with intentional cuts introduced prior to loading, and dynamically, with the cuts introduced suddenly while the structure is loaded. Heat, pressurization, and static loading will be applied as necessary to represent the requirements of the specific condition.

FATIGUE TESTS

The fatigue philosophy described previously relies heavily on accurate laboratory simulation of service loading and environmental conditions, and to a lesser degree on theoretical fatigue analyses. As a result of this emphasis, a refined fatigue test system has been evolved. Magnetic-tape-controlled hydraulic servo valves with load cell or position feedback loops have proved very practical and versatile. Flight-by-flight loading sequences are used to avoid arbitrary decisions for the definition of the very sensitive ground-air-ground cycle per flight.

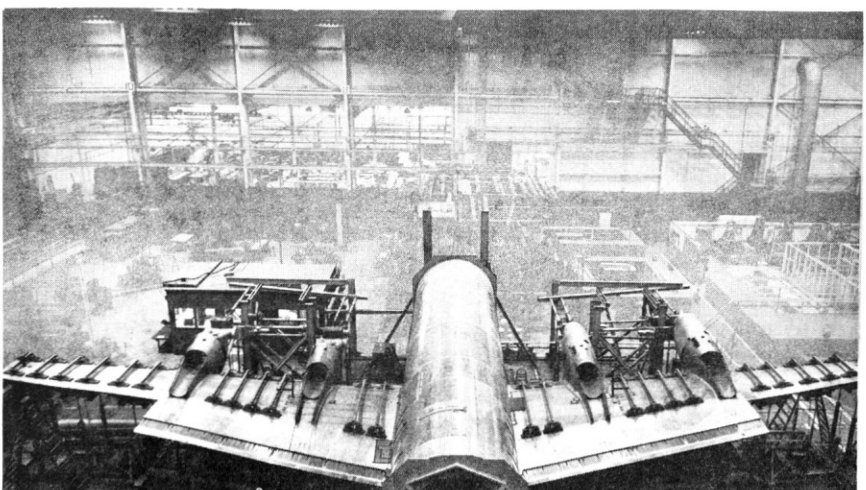

Figure 22. P-3A wing landing gear fatigue tests.

This technique has been developed extensively and used to complete a fatigue test program for the United States Navy on the P-3A antisubmarine patrol airplane. A photograph of the test setup appears in Fig. 22 where the tape-controlled loading jacks are clearly evident. Over 2,000 flights, complete from takeoff to landing with such detail operations as flap extension and retraction, propeller loads, and landing-gear loads, were completed in a calendar period of approximately 4½ months. This technique is fully developed and ready for application on the supersonic transport except for the additional need to simulate thermal cycles.

In addition to the influence of heat on the material properties, heating and cooling of the structure during supersonic flight is an important source of fatigue stress cycles. Ideally, since parts of the structure do not reach temperature equilibrium during service flights, each fatigue-test flight should be carried out on a real-time basis. Practically, this is not possible in a test program which must provide proof of adequacy within a limited time. Long-time experimental programs are now underway to determine the possible acceleration of the heating cycles during fatigue tests without undue compromise on the results of the test.

CONCLUSION

This paper has presented a broad view of the structural design philosophy being employed on the supersonic transport. As indicated at the outset, this philosophy is based upon the five elements of rational load criteria, unlimited life, fail-safe for both strength and stiffness, inspectability and maintainability, and complete structural proof testing. In each of these areas, the approach adopted is to carry over the extensive knowledge and experience which has been gained over the years and to apply it in the new environment and to the new materials as required. By taking full advantage of experience and full account of new conditions, the detailed execution of this philosophy will result in a supersonic transport of sound structural characteristics. The depth of knowledge available from experience and the promise shown by the new materials leads to the conclusion that the supersonic transport will have even greater reliability and safety than that associated with the current subsonic transports.