TOWARD A SECOND GENERATION FUEL EFFICIENT SUPERSONIC CRUISE AIRCRAFT

STRUCTURAL DESIGN FOR EFFICIENCY

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SUMMARY

The unique challenge of this concept to the structural designer is discussed. The potential of the application of advanced structural design concepts and new titanium fabrication processes is emphasized. Highlighted are the results of a detailed structural analysis, including weight and flutter, showing successful use of the ATLAS structural design and analysis system. It is concluded that blending of the structure may not have an adverse impact on structural efficiency, weight, and manufacturing complexity.

INTRODUCTION

The blended wing-fuselage configuration is of considerable interest for supersonic airplanes because the smaller cross-sectional area of the airframe reduces drag. New manufacturing techniques and stronger efforts to conserve fuel have indicated that a new approach to the problem was justified. This paper examines in some detail a structural design that could be used for the blended wing-fuselage configuration. The analysis involves material and allowable selections followed by the selection of critical load conditions from past studies. An elaborate finite element math model is constructed that is automatically resized. The resulting deflections are examined and finally the estimated weights are determined and flutter speeds calculated.

DESCRIPTION OF STRUCTURE

Figure 1 shows a rear view of a typical frame in the heavily loaded area between the front and rear spars.

Each wing spar blends smoothly into a fuselage frame. This blend is accomplished by a large diffusion bonded titanium assembly. This assembly is spliced into the wing spar at wing station 4.65 and into an opposite hand part at wing station 0.0 at the bottom of the fuselage. A smaller diffusion bonded assembly completes the frame structure across the top of the fuselage with splices at left and right wing station 1.12.

The diffusion bonded process was selected because of efficient use of raw material and the cost effective use of the same parts in several locations with only machining differences.

The selection of a structural concept that could withstand high loads circumferentially as well as longitudinally is an essential part of the success of this study. A well stabilized sandwich structure is one of the most efficient types of structure for carrying biaxial compression loads. Aluminum brazed titanium is presently considered to be the best manufacturing process to make such a sandwich. Several other processes (including superplastic-formed, diffusion bonded) are being considered as alternates for possible cost and weight savings. In this study, these panels are configured as shown in Figure 2. The upper and lower panels splice to basic wing panels at wing station 4.65. They are 5 cm thick except locally where they taper down to 2.5 cm at the outboard end. The upper surface panels have substantial curvature at the side of the body and terminate at a longeron at wing station 1.12. This longeron consists of a heavy plate running fore and aft outside of contour and another inside plate which passed thru a machined step in the frames. Another similar longeron is at wing station .4 for a total of four along the top of the body. Between these longerons are three longitudinal panels, 15 meters long, made of 5 cm deep titanium honeycomb sandwich. The longerons serve three functions, 1) they splice the spanwise panels to the body panels and the body panels to each other, 2) they act as primary members in the fuselage strength and stiffness and 3) they act as fail safe members to arrest any circumferential crack that might start in the panels.

The lower surface wing panels have much less curvature and are continuous between wing station 4.65 and wing station 0 where they are spliced by a keel beam.

The vertical side panels at wing station 1.7 are intercostal to the frames and are mechanically fastened to the upper and lower panels. These panels complete the pressure section and are made of aluminum brazed titanium honeycomb.

The fuselage monocoque forward of the main landing gear wheel well is titanium skin-stringer similar to the 1971 SST. In this section, each wing spar attaches directly to a fuselage frame. The loads from the upper surface of the strake must be carried by bending in the frames. The fuselage aft of sta 64.7 is also skin-stringer construction. Appropriate transition sections are included to blend the non-circular sandwich skin into the circular skin-stringer sections.

ANALYSIS MODELS

The structural analysis and design process is based on mathematical models representing airload distributions, mass distributions and structural members. These models are developed from the geometric arrangement and the structural definition of the airplane. ATLAS, a modular system of computer codes integrated within a common executive and data base framework, is used to perform the structural analysis and design. The external geometry and structure of the wing outboard of the blending was taken from a previous study of a delta wing configuration (Reference 1).The previous study established the structural weight of a wing similar to that of the national SST but with the planform and thickness modified to reduce wave drag. The structural analysis was based on wind tunnel model pressure data from the national SST program and a detailed finite element structural mathematical model.

The finite element structural model for the current study was derived from the model used in the previous study. The model was divided into two substructures to permit the increase in size required for modeling the blended section of the fuselage. The wing substructure, which was modeled using spar and cover elements was taken from the previous study.

A new model was made for the fuselage substructure. Rods were used for the frame caps and plates in pure shear for the webs. Web stiffners were included in the model to provide a path for kick loads. The honeycomb skin was modeled as plates which carry inplane loads only. Rods were used to represent the longitudinal splices or longerons. The fuselage substructure along with a section of the wing is shown in Figure 3.

Secondary models of small sections were used to augment the primary model where secondary loads or structural details had a significant effect on the material requirements. For example, a section of the fuselage was modeled with a fine grid of bending plates to determine the significance of cabin pressurization on the non-circular honeycomb shell. This analysis also determined the secondary stresses induced in the sandwich skins due to curvature.

The mass model used in the previous study was modified to account for the payload and fuel distribution changes that resulted from blending the wing and fuselage. Weight of structural members such as spars, ribs, frames and cover elements is determined from the material density and member sizes.

The retained nodes from the previous analysis were adequate for distributing the outboard wing loads through the structure. However, there was not a sufficient number of retained nodes on the fuselage to properly distribute loads through the blended section. Therefore, the loads were redistributed to each frame as shown on Figure 4. Each fuselage nodal load was distributed to the appropriate frames in a manner that would preserve the net shears and bending moments. The total load on each frame was divided among the eight nodes on the lower frame member. Side of body loads and wing loads were also applied to each frame.

The aerodynamic model and the resulting external loads were taken from the previous study. The model included a system of lifting surfaces, slender bodies and interference flow elements. Six flight load conditions and one taxi condition were selected for sizing the blended fuselage structure (table I). These conditions were selected on the basis of a review of the critical design conditions for the national SST and an evaluation of the effect of bi-axial loads in the fuselage shell. The flight conditions are combined with both zero and 1.5 factors on the fuselage pressure of 79.3 kPa. In addition, an ultimate pressure condition of 3 factors on fuselage pressure is considered. This resulted in a total of 14 load cases.

MATERIALS AND ALLOWABLES

The material used for all frame sections and skins is 6Al-4V annealed titanium. The allowables conform with the data of MIL Handbook V. Tension allowables were appropriately reduced to compensate for fastener "hole-out." Theoretical analyses were made to determine the elastic buckling allowables for the "s" shaped honeycomb panels. These calculations showed that the curved panel buckling stresses are higher than for flat panels. For simplicity and conservatism standard flat panel allowables were used.

RESIZE

The resize of the structural elements is automatically done in ATLAS. Element stresses and internal loads (output of the stress modules) are passed on to the design module for sizing. These elements are resized by calculating their margin of safety and modifying their gauges to give a prescribed margin.

Many elements were constrained by a lower bound against resize. Resize constraints include minimum gage requirements, stiffness requirements and structure that would probably be designed by conditions not being considered. Figure 5 shows the minimum gage criteria that was used for this study. It also identifies the sections of rear spar and trailing edge beam that had been stiffened to improve flutter speed in the previous study. These stiffened sections were constrained to that size. Finally, the upper surface panel above the landing gear wheel well will probably be designed by ground handling type conditions. The gages of this panel were selected based on the 1971 SST data and constrained to that size. Those elements that are constrained from being resized will have those requirements imposed on them by the design program.

The strength designed results are then weighed by the mass subroutine. The analysis of this resized structure is repeated for as many cycles as required for convergence to minimum weight structural elements. For this study, good theoretical weight convergence resulted within three analysis cycles, as shown in Figure 6, where the percentage weight change of the total theoretical structure is plotted vs. resize cycles. The final gages of the elements are then printed out for further analysis. Typical results are shown in Figure 7.

Note that the third rod from the centerline of the inner chord of the upper frame has an area significantly smaller than those on either side. This small area occurs because of an inflection point in the bending moment for the symmetrical load condition. Such areas must be identified and smoothed over before final weight estimates are made.

The stiffness matrix of the final results was used to perform stress and deflection analysis which were printed out for evaluation. The deflection results of the 2.5g balanced maneuver at V_D were used to plot the deflected shape of the rear spar frame as shown in Figure 8. As expected and as shown by the deflection values of wing station 2.67, the load path for the upper spar caps is softer than the lower caps therefore deflecting more and resulting in larger wing tip deflections for the blended wing as compared to the conventional wing. These values are indicated in the wing deflections at the rear spar as shown in Figure 9.

Since the blended wing airplane fuselage is not as deep as the conventional configuration the deflections will be higher for the blended wing airplane. This fact is demonstrated by the crown deflection plot for both airplanes shown in Figure 9.

FLUTTER ASSESSMENT

As shown in the calculated static deflections and confirmed in a review of the vibration mode shapes and frequencies, the blended wing configuration is somewhat more flexible in both body and wing bending.

Figure 10 shows that this additional flexibility results in a flutter speed for the blended wing 20 m/sec EAS below the $1.2V_D$ flutter requirement. The flutter mode also changes from a 2.6 Hz mode on the conventional airplane to a 1.9 Hz mode on the blended wing airplane. Moderate additional wing stiffening would probably restore the 2.6 Hz flutter mode at a speed above the flutter requirement.

WEIGHTS

In the ATLAS analysis process, theoretical member sizes required to carry loads and the associated weight based on the member material density and dimensions are calculated. Determination of the total weight of the structural components requires that weight adjustment factors be developed and applied to the theoretical weight. The total estimated weight of the sized primary structural members is the product of the adjustment factor and the theoretical weight. These factors are incorporated in the aeroelastic analysis cycle to provide an iterative capability for evaluating strength and flutter requirements and related weight effects.

Weight adjustment factors provide for so called "non-optimum" features i.g. reinforcement pad-ups, dense honeycomb core edges, braze material, splices, material tolerances, etc. These factors can represent an appreciable weight as shown by Figure 11 which reflects the adjustment factors to be applied to the theoretical skin weight in a blended titanium honeycomb pressurized body. These are only skin weight adjustments and do not include the core, core edge members, braze etc. Lower surface factors are higher in this case because of the greater number of cut outs in the lower surface.

Total structural weight for the blended body section analyzed includes the primary structure previously discussed and secondary structural items, i.e., bulkheads, doors, decks, etc. The total estimated weight of the structure analyzed was 29,500 kg for an airplane having a design gross weight of 340,200 kg and body length required for 269 passengers. A preliminary comparison of this body section for a mid-wing design versus a comparable section in a low wing design indicates that the mid-wing design body structure is only slightly heavier.

CONCLUDING REMARKS

This study demonstrates that this is a practical, efficient structural design that can be used to achieve the blended wing fuselage configuration that is desirable to improve the performance of a second generation supersonic cruise configuration.

REFERENCE

1. Advanced Supersonic Configuration Studies Using Multi-Cycle Engines for Civil Aircraft. NASA CR-132723, 1975.

TABLE 1.- DESIGN LOAD CASES

CONDITION DESCRIPTION	FLAP SETTING	MACH NO.	ALTITUDE m	GROSS WEIGHT kg	C. G. m	n	U_{de} m/sec	PRESSURE FACTOR
BAL. MAN. AT V_C	SUBSONIC	0.60	2000	335,000	53.5	2.5	0.0	0 & 1.5
POSITIVE GUST AT V_B	SUBSONIC	0.60	4500	202,300	53.4		20.1	0 & 1.5
BAL. MAN. AT V_D	SUBSONIC	0.95	6300	325,900	53.6	2.5	0.0	0 & 1.5
POSITIVE GUST AT V_B	SUBSONIC	0.95	11,000	202,300	53.4		14.9	0 & 1.5
BAL. MAN. AT V_C	SUBSONIC	0.95	8800	325,900	53.6	-1.0	0.0	0 & 1.5
BAL. MAN. AT V_A	TRANSONIC	1.30	11,400	316,800	53.7	2.5	0.0	0 & 1.5
TAXI		0.0	0	339,500	54.4	2.5	0.0	0
ULT. PRESSURE								3

V_C MAXIMUM CRUISE SPEED
V_B ROUGH AIR SPEED
V_D DIVE SPEED
V_A MINIMUM SPEED
U_{de} GUST VELOCITY
n LIMIT LOAD FACTOR

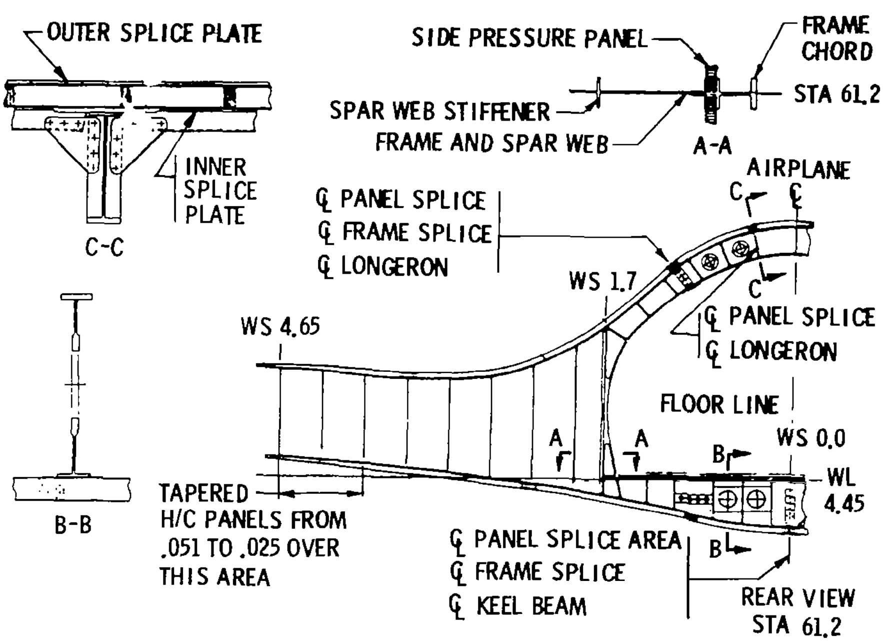

Figure 1.- Frame and spar concept blend area.

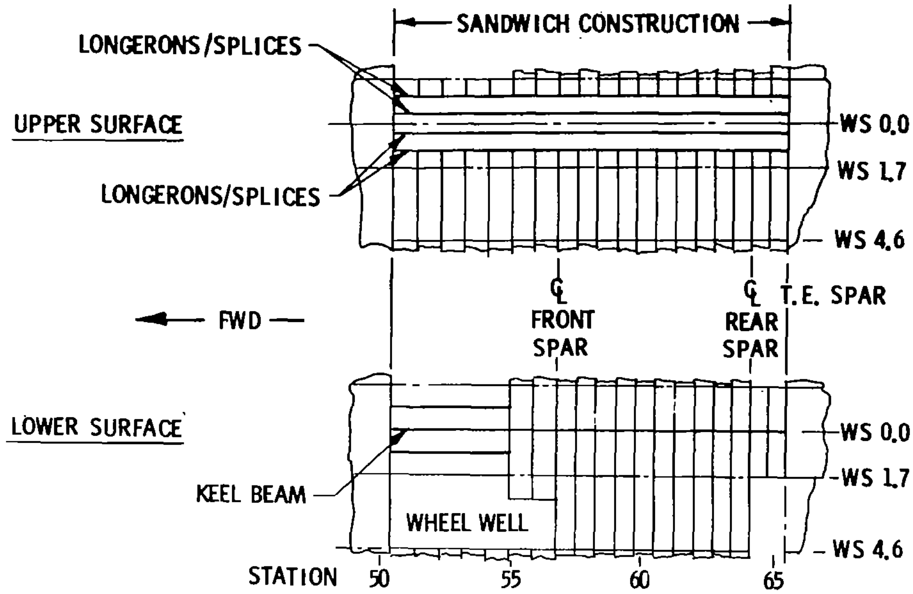

Figure 2.- Panel configuration.

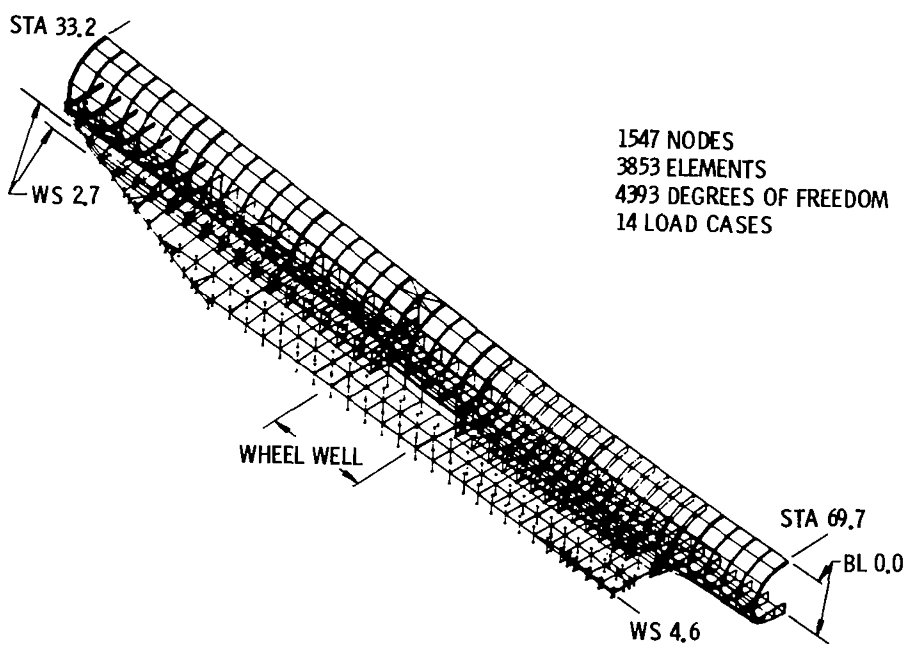

Figure 3.- Resized blended wing/fuselage math model.

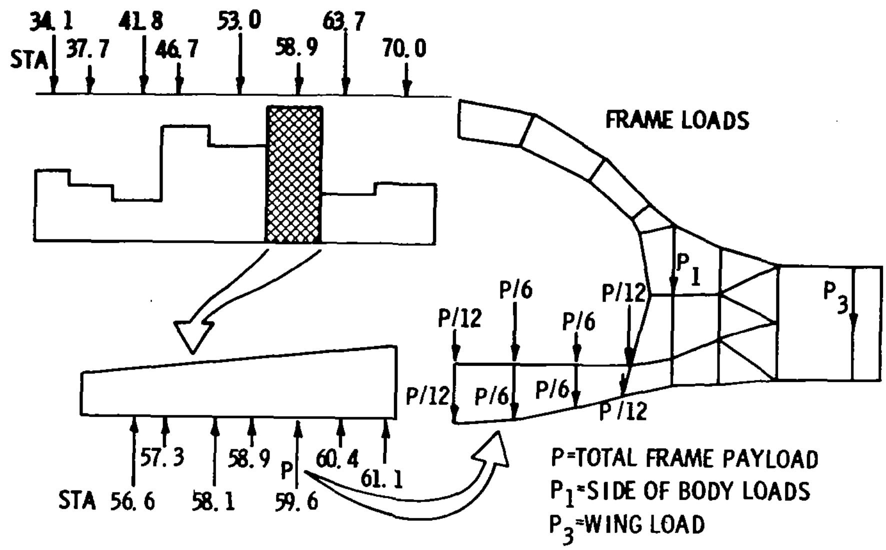

Figure 4.- Fuselage/frame load distribution.

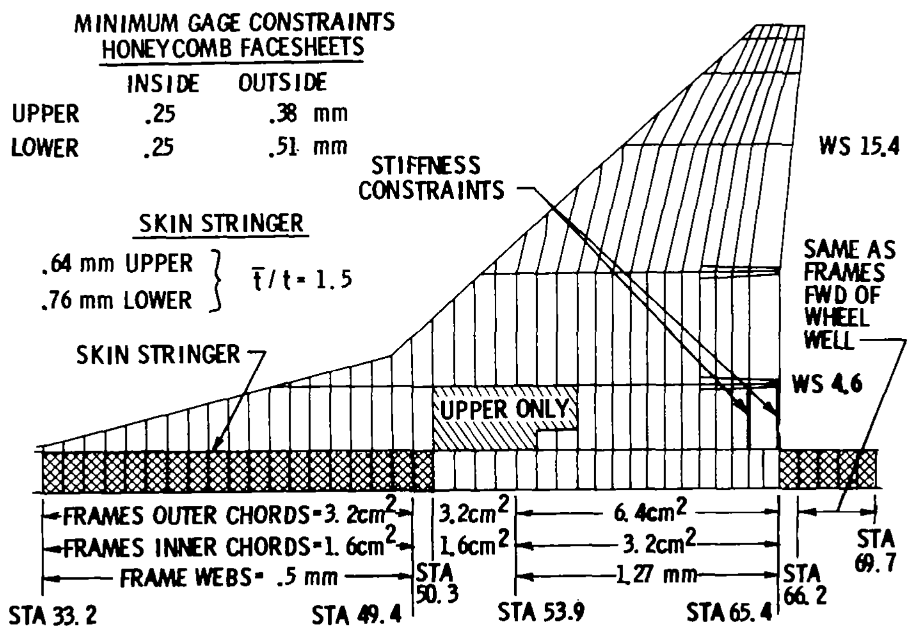

Figure 5.- Resize constraints. ($\bar{t}/t$ is the ratio of skin plus stiffeners area to skin area.)

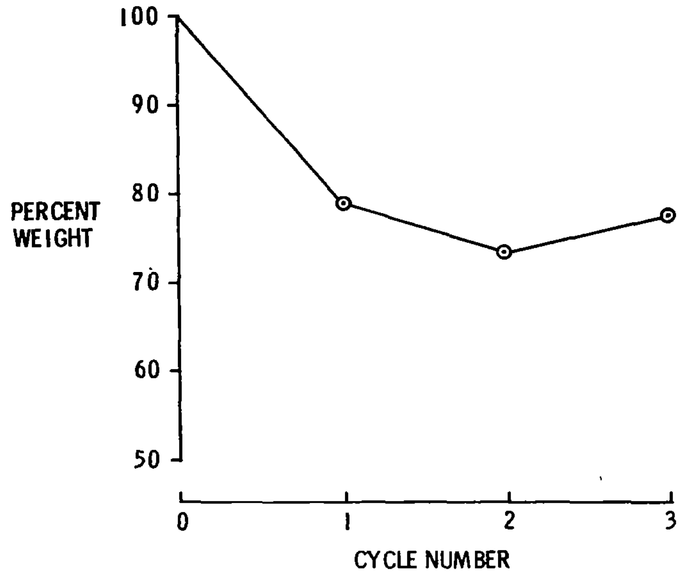

Figure 6.- Weight versus design cycle.

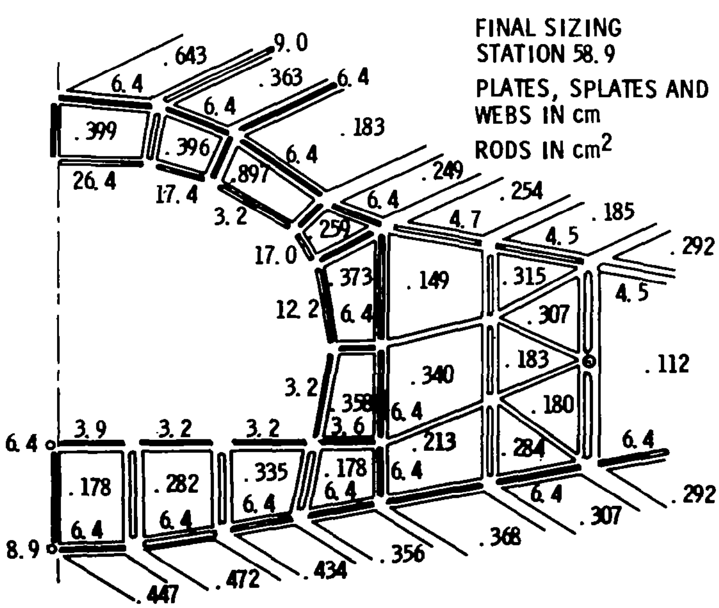

Figure 7.- Typical sizing for blended area.

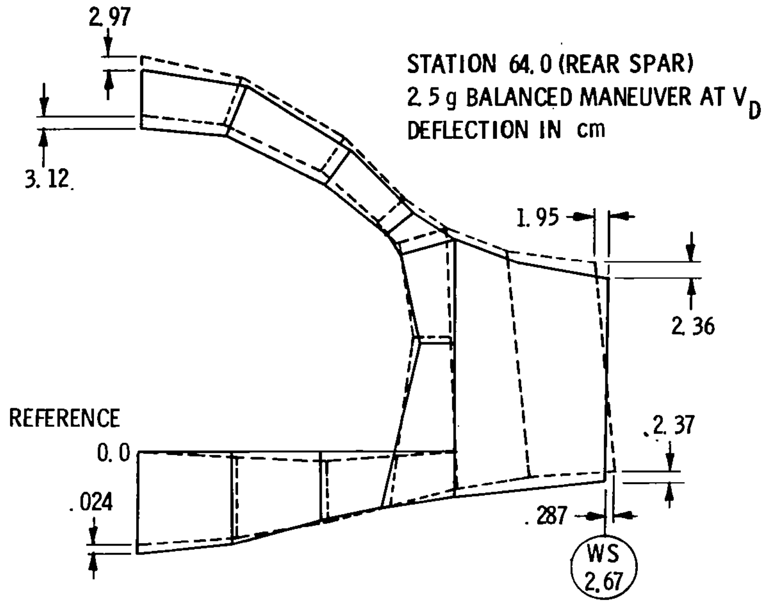

Figure 8.- Frame limit deflections.

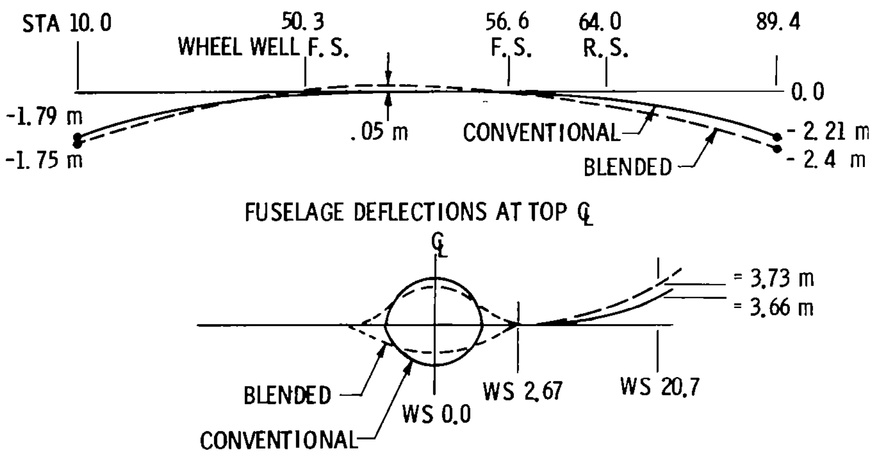

Figure 9.- Comparison of limit deflections for blended and conventional wing airplanes. 2.5g balanced maneuver at V_D.

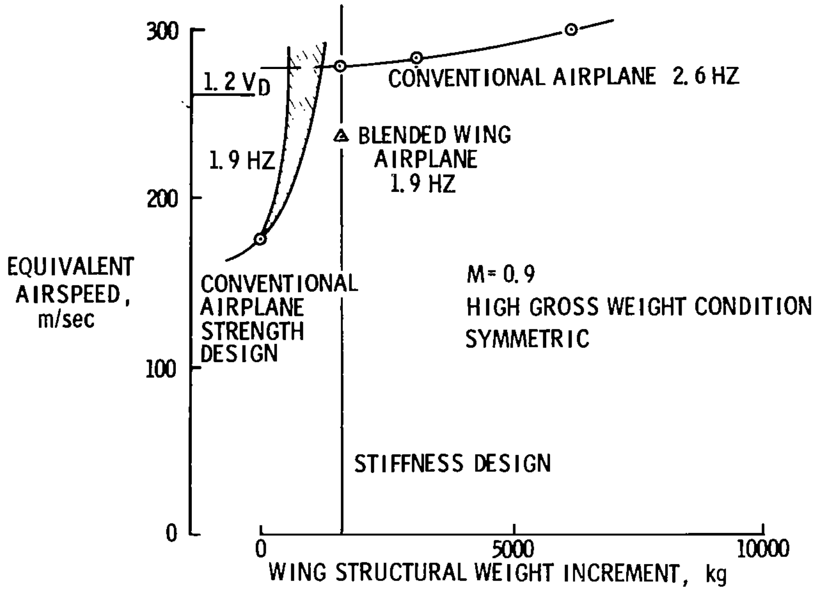

Figure 10.- Flutter speed comparison for blended wing and conventional airplane.

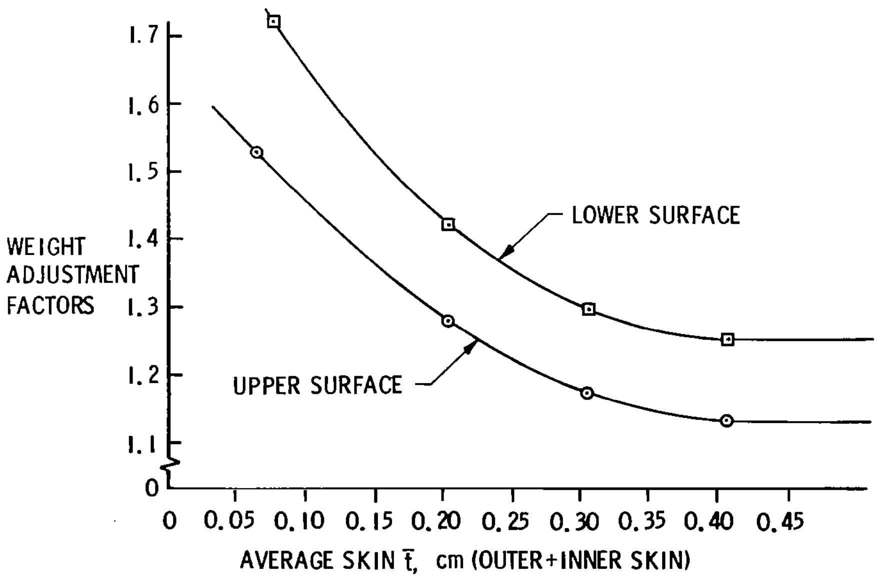

Figure 11.- Skin weight adjustment factor. Body pressurized; titanium honeycomb panels.