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SIZING-STIFFENED COMPOSITE PANELS LOADED IN THE POSTBUCKLING RANGE

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THE POSTOP PANEL SIZING CODE

Stiffened panels are widely used in aircraft structures such as wing covers, fuselages, control surfaces, spar webs, bulkheads, and floors. The detailed sizing of minimum-weight stiffened panels involves many considerations. Use of composite materials introduces additional complexities. Many potential modes of failure exist. Analyses for these modes are often not trivial, especially for those involving large out-of-plane displacements. Accurate analyses of all potential failure modes are essential. Numerous practical constraints arise from manufacturing/cost considerations and from damage tolerance, durability, and stiffness requirements. The number of design variables can be large when lamina thicknesses and stacking sequence are being optimized. A significant burden is placed on the sizing code due to the complex analyses, practical constraints, and number of design variables. On the other hand, sizing weight-efficient panels without the aid of an automated procedure is almost out of the question.

The sizing code POSTOP (Postbuckled Open-STiffener Optimum Panels) has been developed (refs. 1 and 2) to aid in the design of minimum-weight panels subject to the considerations mentioned above. Developed for postbuckled composite panels, POSTOP may be used for buckling resistant panels and metallic panels as well. The COPES/CONMIN (refs. 3 and 4) optimizer is used in POSTOP although other options such as those in the ADS (ref. 5) system could be substituted with relative ease. The basic elements of POSTOP are shown in figure 1. Some of these elements and usage of the program are described on the following pages.

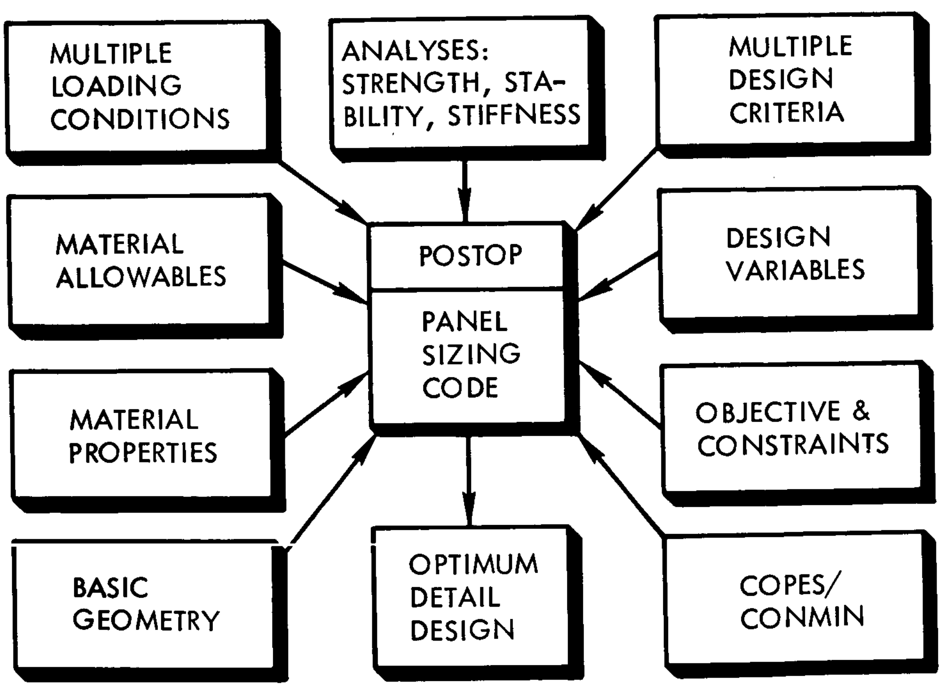


Figure 1.

PANEL GEOMETRY AND LOADS

The basic geometry and types of loads that are considered in POSTOP are shown in figure 2. The stiffener spacing is assumed to be small compared to the panel length. This is normally the case for stiffened panels used in transport aircraft. The stiffeners may have any cross-sectional shape that can be derived from an I-section. The stiffeners may be integral with the skin or separate elements bonded to or cocured with the skin. Examples are shown in the figure.

Combined inplane shear and biaxial loads may be specified. Normal pressure and temperature changes are also considered in the analyses. The bending effects of an initial bow over the panel length and eccentricity of applied loads are included. The interaction of bending due to pressure or eccentricities and inplane loads is accounted for. The effects of stiffness reductions due to postbuckling on this interaction are considered. This interaction can have a significant effect on the panel design and must be considered during sizing.

Aircraft structures are subjected to a large number of independent loading conditions. Often different design criteria are imposed for different load cases. For example, panels may be allowed to operate in the postbuckling regime at certain load levels and be required to be buckling resistant at lower load levels. Conditions associated with high temperature may require different material properties and allowables. Limit and ultimate loading conditions obviously use different material allowables. Nonlinearities require that both limit and ultimate conditions be analyzed. Often many load cases may be eliminated by inspection as being noncritical. However, several load cases usually remain that must be evaluated. The POSTOP code and other available panel sizing codes have this multiple load-cases capability.

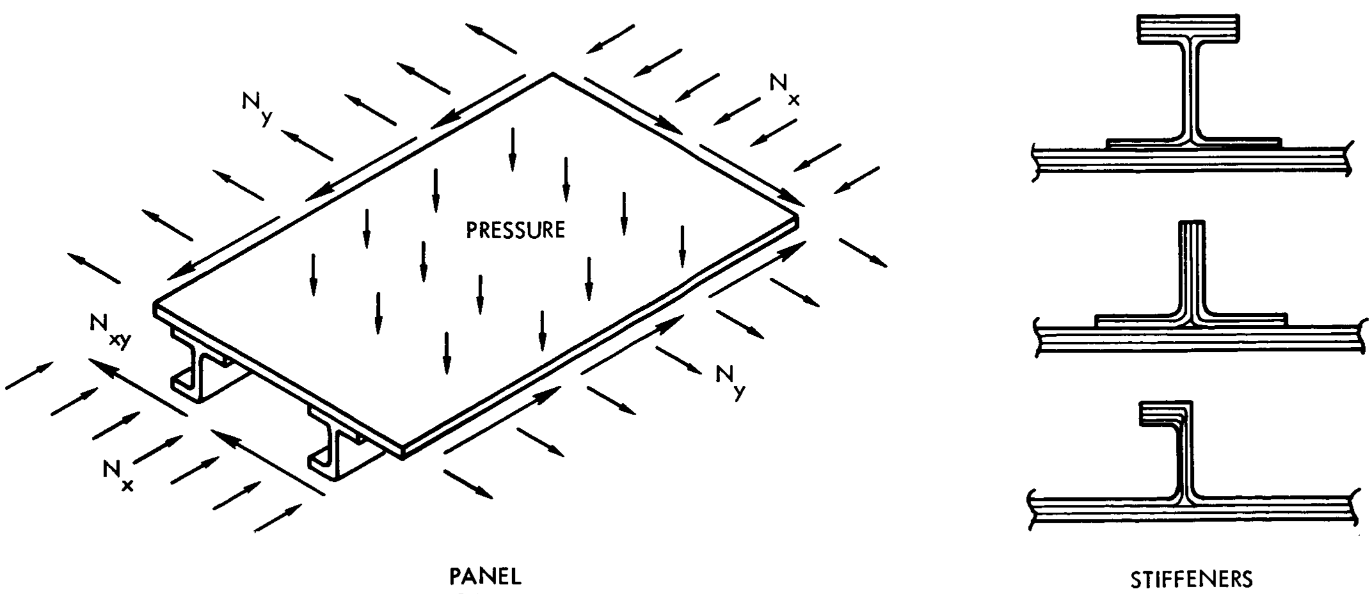


Figure 2.

POSTBUCKLING AND STABILITY ANALYSES

Strength and stability analyses performed in POSTOP include initial buckling of the skin and stiffener, postbuckling of the skin, torsional/flexural buckling of the stiffener, and ply-level membrane plus bending strains in the skin and stiffener elements. Various nonlinear effects enter into these analyses.

If the skin is not buckled, the only nonlinearity in the load-deformation relationship results from the interaction of inplane loads and panel bending as mentioned previously. If the skin is buckled, as shown in figure 3, several additional nonlinearities enter into the analysis. After buckling, the compression load in the skin is redistributed, with an increased percentage of the load being carried near the edges, where it is supported by the stiffeners. The secant and tangent stiffnesses of the skin are reduced after buckling. The reduced secant stiffness causes an increased proportion of the panel load to be carried by the stiffener. This increase affects the local and torsional/flexural buckling of the stiffener. The reduced tangent stiffness of the skin also affects the stability of the stiffener since it offers less restraint to incremental deformation. The reduced tangent stiffness increases the interaction of inplane loads and panel bending.

Since the skin and individual stiffener plate elements do not typically buckle at the same load level or in the same wavelengths, the restraint of adjacent elements is considered when computing the skin and stiffener local buckling loads. Likewise, the restraining effects of the skin at the edges of the stiffener attached flanges are included in the torsional/flexural buckling analysis. Local and torsional/flexural buckling analyses are performed for a series of admissible buckling wavelengths and the lowest buckling load level is sought.

Local bending strains are significant in a postbuckled skin. While the membrane strain in the center of the plate may be small, as shown in the figure, the total compressive strain on the concave surface at the buckle crest may exceed the edge strain. On the other hand, the total strain on the convex surface may actually be tensile. Ply-level stresses and strains are computed at critical locations in the skin and stiffener elements and margins of safety are computed based on the maximum strain or the Tsai-Hill criterion.

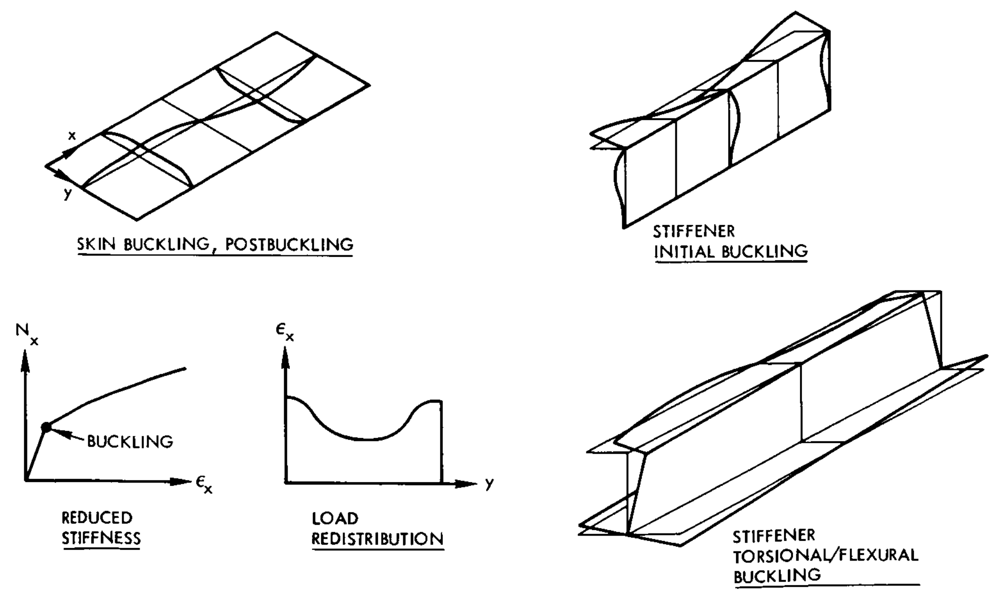


Figure 3.

SKIN/STIFFENER INTERFACE STRESSES

Separation of the skin and stiffener is one of the most commonly occurring failure modes in postbuckled and pressure-loaded composite panels. A self-contained analysis procedure has been developed and incorporated in POSTOP to evaluate the normal and shear stresses in the interface between the stiffener attachment flange and the skin. Typical deformations and the structural model are shown in figure 4. The flange and skin are modeled as plates connected by an elastic interface layer. The length of the buckling half-wave defines the length of the model. Sinusoidally distributed moments and shears computed from the postbuckled plate analysis are treated as applied loads in the skin plate near the free edge of the attached flange. The effects of the longitudinal compression loads in the plates are included and have been found to be significant. Interface stresses may be computed at any point along the half-wavelength and across the flange width. Normal and short transverse shear stresses are maximum at the buckle wave peak. The long transverse shear stresses are maximum along the buckle node line, where failure involving shear crippling has been observed.

Parametric studies performed with this analysis have shown that the interface stresses may be minimized by proper detail design techniques. For example, the addition of a pad in the skin under the stiffener reduces all interface stresses significantly. The effect of a skin pad on the shear stresses is shown in the figure. Other design variables including flange width and stacking sequence are also available to control the interface stresses. The success or failure of an optimum postbuckled panel design may depend on attention to design details such as these.

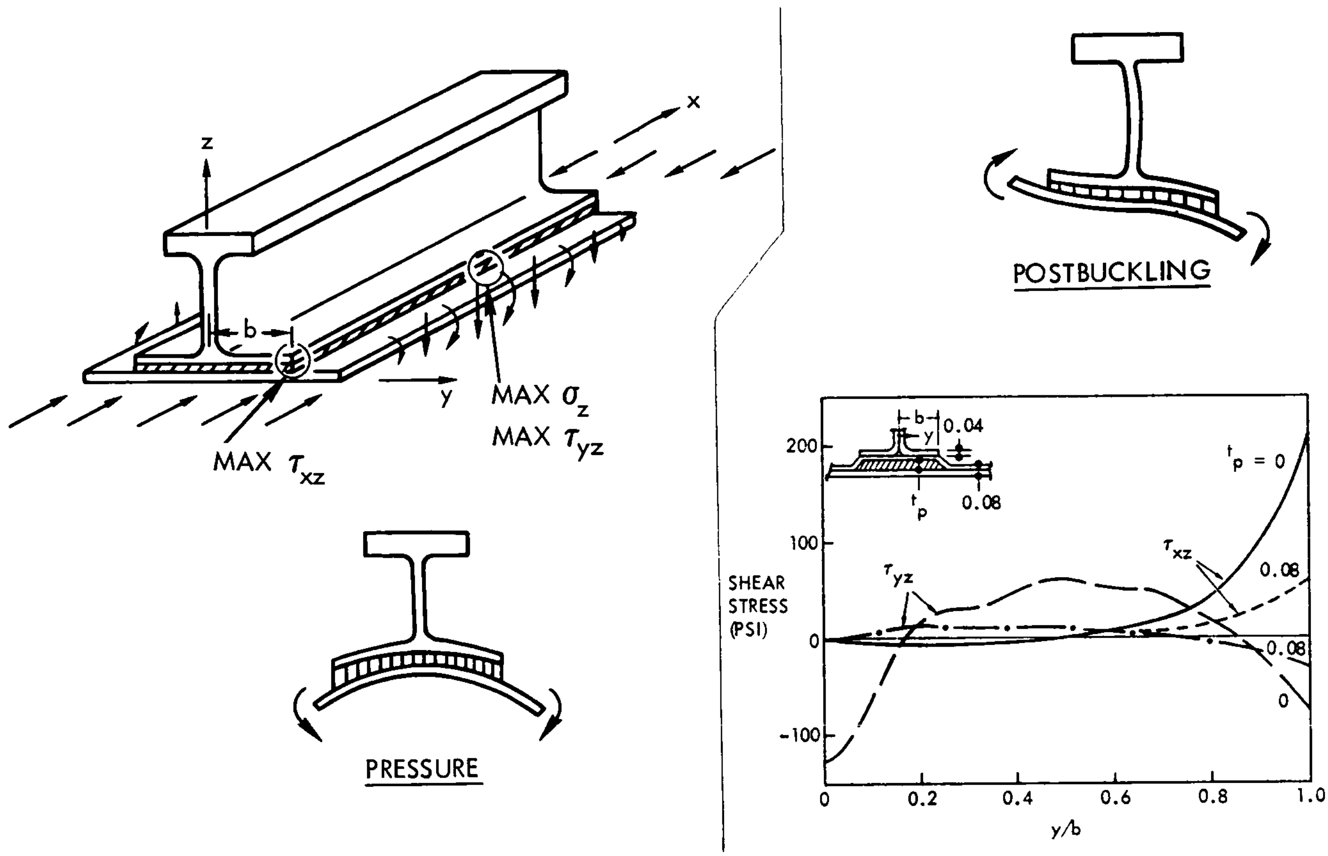


Figure 4.

OPTIMIZATION PROBLEM FORMULATION

The design variables in POSTOP are shown in figure 5. They are the five element widths of the I-section stiffener, the stiffener spacing and the lamina thicknesses in the skin and the stiffener elements. All design variables are considered to be continuous. Any width except the stiffener height may be set equal to zero to produce stiffeners with cross sections other than the I-shape. Currently up to 20 width and thickness variables may be specified. Any design parameter may be linked to a design variable with a constant multiplier. Using linking to achieve practical designs allows the total number of independent design variables to be in the range of 10 to 15 for most stiffened composite panels. The requirement for lamina thicknesses to be integer multiples of available ply thicknesses and treatment of stacking sequences are discussed later.

The most common objective function in aircraft panel sizing is minimum panel weight. Maximum stiffness or maximum margin of safety in a particular failure mode could be specified as objective functions in certain instances.

Constraints may be placed on the magnitude of the design variables, ratios of selected design variables, panel stiffnesses, and individual margins of safety. When minimum weight is not the objective function, panel weight should be constrained. Proper specification of these constraints allows practical optimum designs to be determined. Added safety may be ensured in certain major failure modes, such as panel instability, by specifying a higher lower bound for the margin of safety in that mode.

The CONMIN program used in POSTOP is a widely used optimizer based on the method of feasible directions. POSTOP uses CONMIN with finite-difference gradients due to the nonlinear nature of the optimization problem and of the structural response.

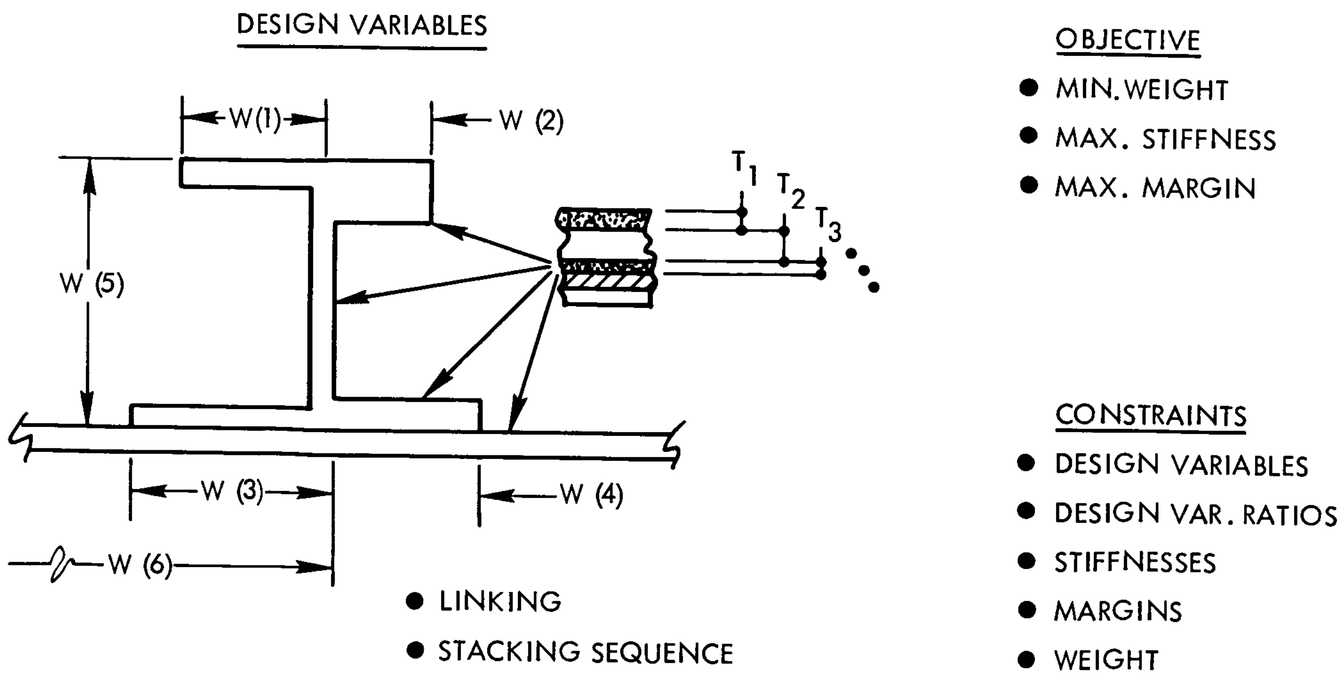


Figure 5.

INTEGER NUMBERS OF PLIES

Considering lamina thicknesses as continuous variables is a requirement for the CONMIN optimizer used in POSTOP. Optimum designs generally contain laminae having fractional numbers of available plies. In thick laminates, simply rounding the optimum lamina thicknesses up or down to the nearest integer number of plies may have a negligible effect on panel weight. Such rounding of lamina thicknesses becomes more significant when the total laminate thickness is small or when preplied laminae are used to lower fabrication cost. When off-axis material, such as ± 45 degree plies, is used it must be supplied in multiples of four to maintain a balanced symmetric laminate. Here the rounding effect is multiplied by four.

The negative aspects of this rounding procedure are generally lessened in importance by several factors. Often if one lamina is rounded up, another can be rounded down, cancelling to some extent the weight penalty. If truly continuous design variables such as spacing and widths are available, a second optimization on only the continuous variables may be performed after lamina thickness rounding. This currently suggested approach to be used with POSTOP is outlined in figure 6. Experience has shown that after rounding and reoptimizing, the weight penalty is usually less than three or four percent compared to absolute optimum fuselage panel designs. This penalty can decrease further when thicknesses vary along the structure length, and plies may be dropped at any point along the length whenever a smaller integer number of plies is required.

There are cases, however, when the current approach leads to the wrong solution. For example, if a $[\pm 45_n/0_m/\mp 45_n]$ plate is to remain buckling resistant in pure compression, an optimum design might require n = 1.15 and m = 0.0, since a laminate with only 45-degree plies is optimum for this case. The rounding procedure would require a $[\pm 45_2/\mp 45_2]$ laminate resulting in a 74-percent penalty. If optimization on integer numbers of plies were used, a $[\pm 45/0/\mp 45]$ laminate might prove optimum resulting in only a 9-percent penalty. Although this example exaggerates the problem, a method of optimizing on continuous and discrete value design variables simultaneously would be of value in composite panel sizing.

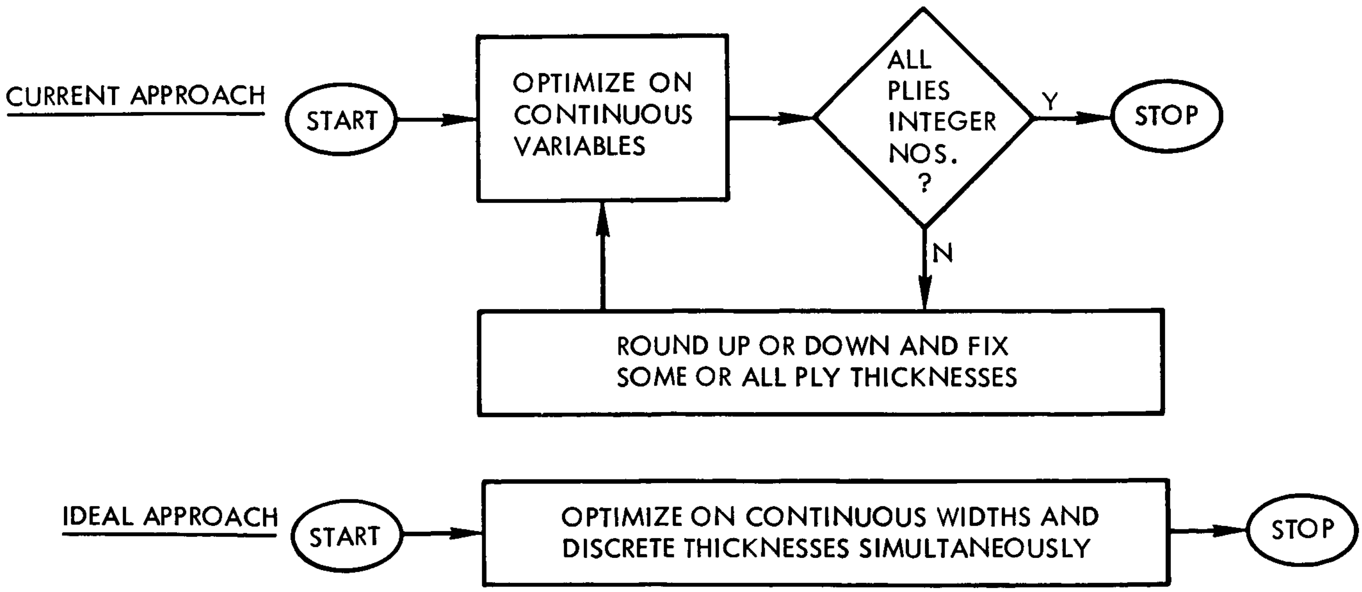
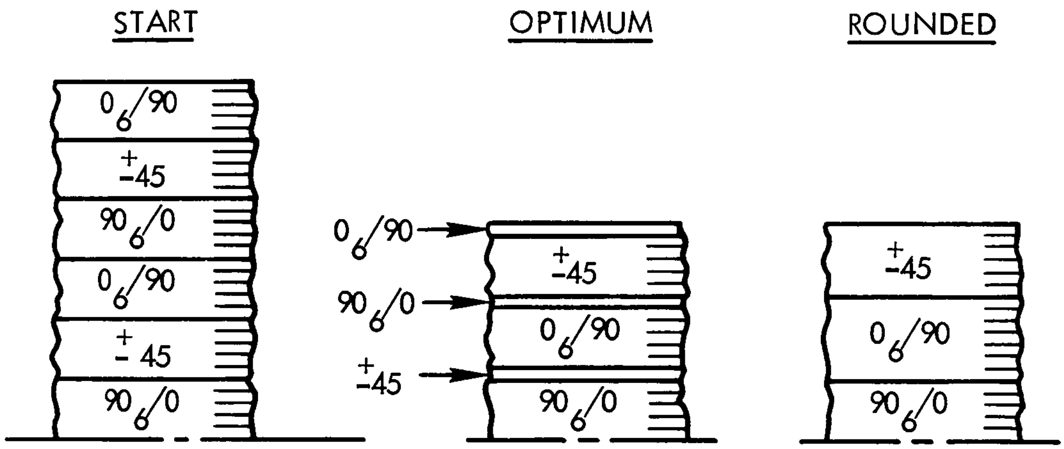


Figure 6.

STACKING SEQUENCE OPTIMIZATION

The stacking sequence of the plies in a laminate can have marked effects on buckling loads, postbuckling response, local bending stresses and stiffnesses, free edge interlaminar stresses, skin/stiffener interface stresses, and delamination growth. Provided an accurate analysis is available to evaluate such effects, optimization on lamina thicknesses can be used directly to determine the optimum stacking sequence as well as the total amount of material required in the various ply orientations.

The approach that can be used in POSTOP to determine optimum stacking sequence is summarized in figure 7. If 0-, 90- and ±45-degree orientations are to be used in a laminate, the laminate specified to start the optimization process should have approximately equal numbers of plies in the three directions. More importantly, material with each orientation should be repeated at least once and the thickness variables should not be linked. Optimization will reduce the thickness of laminae with undesirable orientations to relatively small values, as shown in the figure. These reduced thicknesses are then rounded out of the laminate and the optimum stacking sequence remains. Reoptimization should be performed after rounding.



- PROVIDE CHOICE OF LOCATIONS FOR EACH ORIENTATION
- OBSERVE RELATIVE THICKNESS TRENDS
- ELIMINATE LAMINAE WITH RELATIVELY SMALL THICKNESSES
- REOPTIMIZE

Figure 7.

BENEFITS OF POSTBUCKLED DESIGN

The weight savings of postbuckled panel design relative to buckling resistant design have been recognized in metallic fuselage construction for many years. Reluctance to use postbuckling composite panels exists due to the low out-of-plane strength and stiffness of composites. Recently, design details such as the padded-skin concept and attachment methods such as stitching have been shown to be effective in preventing skin/stiffener separation failures in postbuckled composite panels. Questions still remain as to the durability of such panels in fatigue loading, particularly if interlaminar damage or defects are present. Other failure modes such as shear crippling may become critical when separation is suppressed. Assuming these questions can be answered with new analytical/experimental developments, postbuckling design will become widely used in composite fuselage structures. POSTOP has been used to determine the benefits of postbuckled design for composites as compared to a buckling resistant design approach.

The potential weight savings of postbuckled composite fuselage panels as compared to panels that are required to remain buckling resistant is shown in figure 8(a). Here the mass index (panel weight per unit surface area, W, divided by panel length, L) is plotted as a function of the load index (compressive stress resultant, N_x, divided by panel length) for both buckling resistant and postbuckled designs. Weight savings ranging from 25 percent at the lower load levels to 15 percent at the higher load levels are possible with postbuckled design.

Another advantage of postbuckled design is illustrated in figure 8(b). The effect of stiffener spacing on panel weight is shown for stiffened panels designed for a given load level. Again, postbuckled designs and buckling resistant designs are compared. For the buckling resistant panels, there is a significant weight penalty to increase the stiffener spacing. For the postbuckled panels, on the other hand, there is almost no weight penalty associated with an increase in stiffener spacing. Since increasing the stiffener spacing translates into fewer parts, cost savings may be realized with postbuckled design in addition to weight savings.

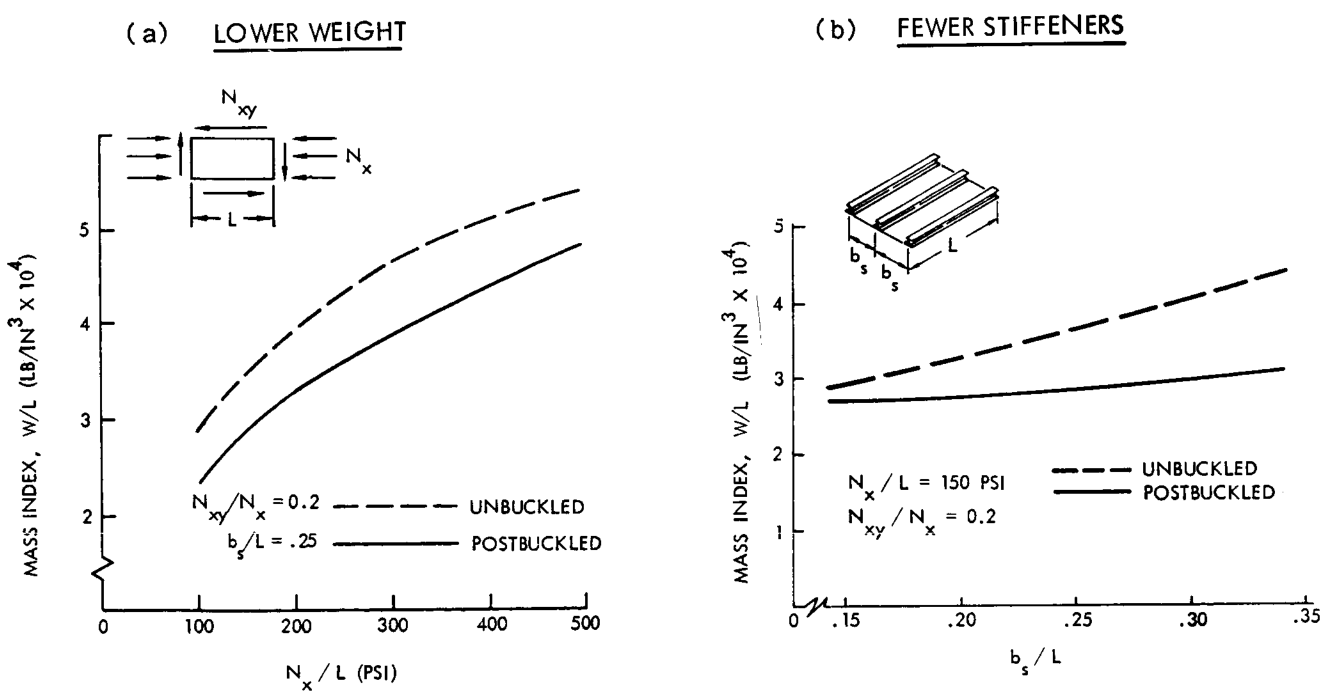


Figure 8.

MULTI-STATION SIZING PROCEDURE

The feasibility of obtaining optimum designs for stiffened wing, empennage, or fuselage surface panels has, to date, been constrained by the required point-by-point application of most panel sizing codes. Optimum designs obtained at each point satisfy all the design requirements but are not necessarily geometrically compatible with adjacent designs. The panel sizing code, POSTOP, has been extended to allow determination of designs at a number of adjacent stations that are compatible and that minimize the weight of the total surface panel. This improved sizing code increases the structural efficiency, the computational efficiency, and the designer efficiency over that obtained using previous sizing procedures.

Suppose, for example, that a wing surface is to be designed. Point optimum designs may indicate stiffener spacings of 8, 6, 7, and 4 inches at adjacent stations. If a constant stiffener spacing is required, the designer must select an intermediate spacing, weighted in some way to reflect the wider surface dimensions nearer the wing root, and reoptimize the panels. If similar geometric requirements dictate the relationship of stiffener heights, widths, and lamina thicknesses as well as stacking sequences from station to station along the wing, the number of arbitrary decisions required by the designer may soon become overwhelming. Numerous modifications of these decisions and subsequent reoptimizations may be required in attempting to minimize the total weight of the wing surface. A true minimum weight design may never be obtained, even after extensive effort by the designer.

The improved sizing code eliminates the difficulties and inaccuracies described above. Lamina thicknesses, stiffener dimensions, and stiffener spacing are assumed to vary smoothly from station to station. Up to a second-order longitudinal variation of any dimension or thickness is currently allowed, as shown in figure 9. Here X_{oi}, a_i, and b_i are the design variables for the ith design parameter $X_i(x)$. If optimum values for n design parameters are to be determined at each station on the structure, no more than a total of 3n design variables must be optimized regardless of the number of stations specified. In this way, the size of the optimization problem remains relatively small, the required computer time is decreased, and the likelihood of determining a successful optimum design is increased.

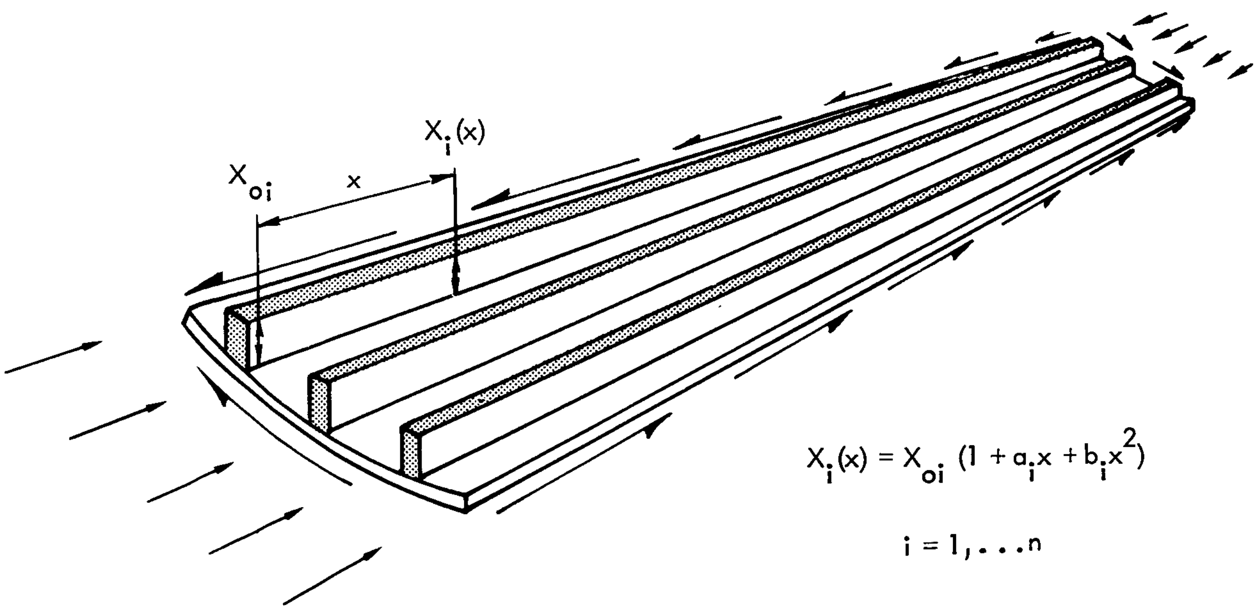
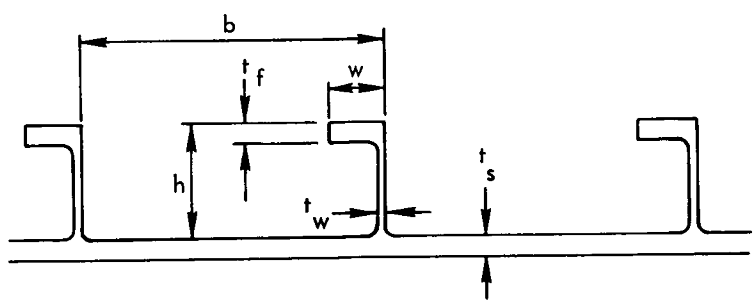


Figure 9.

EXAMPLE OF MULTI-STATION SIZING

In the optimization procedure, the minimum total weight of the structure is the objective function. The width of the structure may be specified at each station so that the weight of structures with tapering planform, such as wing covers, may be accurately determined. As an example of the application of this procedure, consider a wing surface subject to the ultimate loads listed in figure 10. The wing chord widths and minimum shear stiffness requirements are also shown in the figure. For simplicity, assume that the surface panel is to be aluminum with integral stiffeners, as shown in the figure. The allowable effective stress is 53 ksi. Local buckling is not allowed. The station-to-station geometric constraints are (1) constant stiffener spacing; (2) linearly varying stiffener height, flange width, and web thickness; and (3) second-order variations in the skin and flange thicknesses. The six design parameters and the 13 associated design variables are listed in figure 10.

STATION (IN)	SPANWISE LOAD (LB/IN)	SHEAR LOAD (LB/IN)	CHORD WIDTH (IN)	SHEAR STIFF. REQ'D. (LB/IN X 10^6)
79	20,850	2850	144	1.05
142	17,890	2880	139	0.94
260	13,070	2280	129	0.75
368	8,990	1940	120	0.65
522	4,400	1450	107	0.51
673	1,220	660	95	0.36



PARAM. / VARIABLE	w	h	b	t_f	t_w	t_s
X_{oi}	X_{o1}	X_{o2}	X_{o3}	X_{o4}	X_{o5}	X_{o6}
a_i	a_1	a_2		a_4	a_5	a_6
b_i				b_4		b_6

Figure 10.

COMPARISON OF RESULTS

The results of the sequential application of the point-by-point optimization procedure to the same wing surface panel are shown in figures 11(a) through (g). In each design cycle, optimum designs were obtained at each of the six stations with six independent computer runs. In Cycle 1, all six design parameters were allowed to vary freely. The resulting designs, shown in figure 11(a), violate all of the station-to-station constraints. Using the optimum stiffener spacings from Cycle 1, a constant spacing of 6.22 inches was computed with the panel weight per unit length at each station as weighting factors. Using this constant value for stiffener spacing, a second optimization cycle was performed with the remaining five parameters as design variables. The resulting designs are presented in figure 11(b) with the constrained stiffener spacing shown as a short dash line. Next, the stiffener height constraint was applied. A third optimization cycle was performed using the remaining four parameters as design variables. The resulting designs are presented in figure 11(c) with the newly constrained parameter, h, shown as a short dash line and the previously constrained parameter, b, shown as a long dash line. This process was continued until all station-to-station constraints were imposed. The resulting final design is shown in figure 11(g). The total weight of the optimum surface panel is 1912 pounds, only 2 percent heavier than the multi-station optimum. However, 42 separate computer runs were required by the point-by-point procedure, and 1400 computing units were used.

The dimensions of the optimum design obtained with the new sizing code are shown in figure 11(h). The total weight of this surface panel is 1881 pounds. This design was obtained in one computer run that used 1000 computing time units.

This simple example shows the benefits of multi-station optimization. Compared with point-by-point optimization, a small reduction in structural weight and a 30 percent reduction in computer time were achieved. The designer time was greatly reduced by eliminating the cycle-to-cycle decision concerning practical constraint should be applied next and how it should be applied. Reduction of the number of data setups and computer runs from 42 to 1 results in the most dramatic improvements in efficiency. Improvements in structural, computational, and designer efficiencies increase as the number of design variables increases.

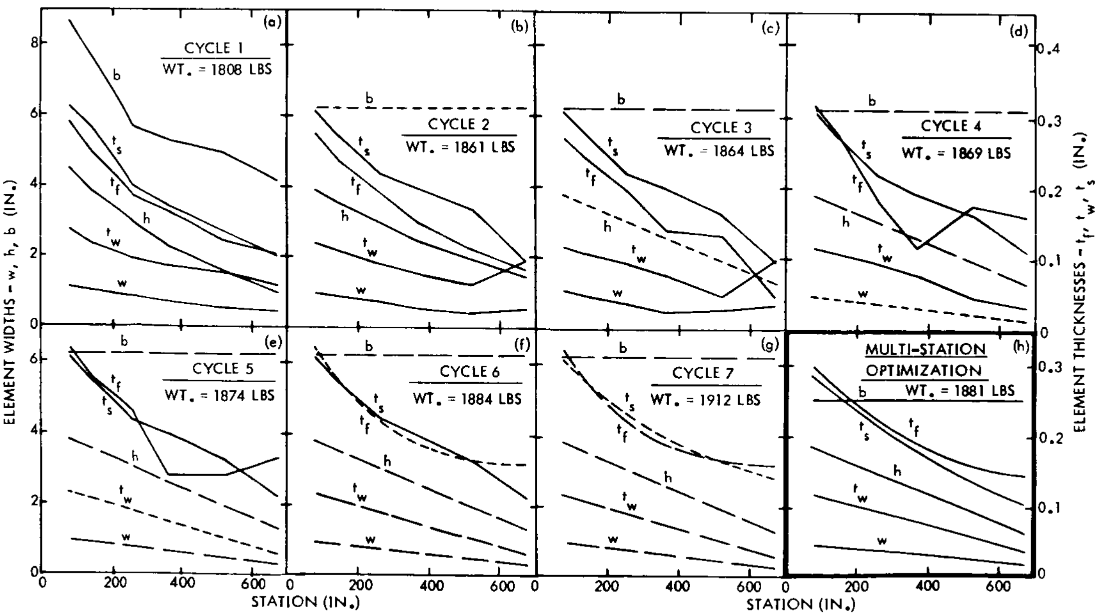


Figure 11.

POSTBUCKLED FUSELAGE INTERNAL LOAD REDISTRIBUTION

A fuselage subjected to multi-axis bending, shear, and torsion will experience panel-to-panel as well as skin-to-stiffener load redistribution after skin buckling. This circumferential redistribution is due to the effect of reduced skin stiffnesses on the overall bending and torsional stiffness of the fuselage. An iterative procedure has been developed to compute this redistribution and the reduced global bending and torsional stiffnesses associated with skin buckling. Reduced global stiffnesses may, in turn, affect the computation of external loads on the fuselage.

As an example of the internal load redistribution, consider a circular fuselage subject to a vertical shear V_z, a torsion M_x, and a bending moment M_y. Figure 12 shows the shear flow and axial load distribution as a function of load level. The neutral-axis shift toward the upper tension-loaded portion of the fuselage is clear. As a result, the tension loads increase at an increasing rate after buckling. Likewise, an increasing proportion of the compression loads is carried by the panels close to the sides of the fuselage after buckling. In this single-cell example, no redistribution of shear load occurs as it does in the case of a multi-cell fuselage. However, even in this example, consideration of combined shear and biaxial loads is important due to their interactive effect on postbuckled plate stiffnesses.

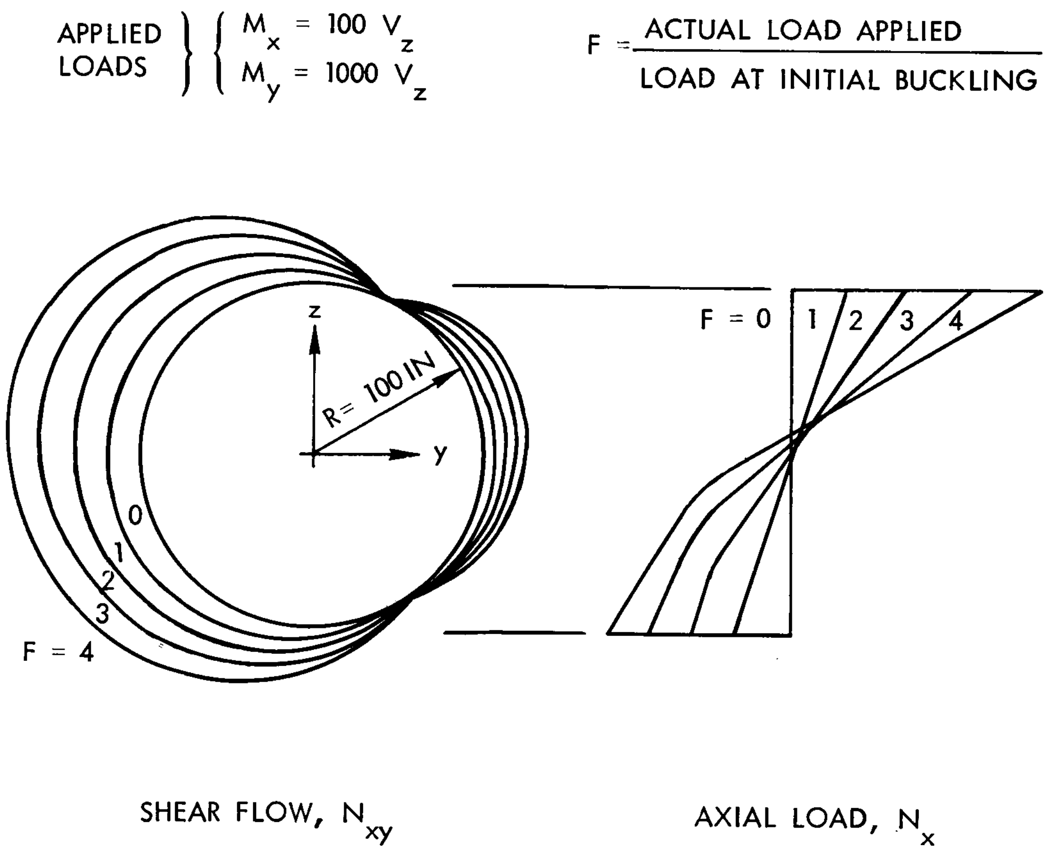


Figure 12.

MULTI-LEVEL STRUCTURAL DESIGN INTERACTION

Aircraft structural design is carried out on several levels of detail. Optimization at any level causes interaction with the others. Nonlinearities due to postbuckling stiffness reductions cause external and internal load redistribution and additional interaction between design levels. Figure 13 illustrates potential interaction between five levels of analysis and design detail. Dashed lines between major components and subcomponents, and between stiffened panels and laminates indicate that the two adjacent items are sometimes not treated separately.

For a fixed aircraft configuration, approximate external loads (rigid loads) are computed. Based on these loads, initial component designs are determined. Refined external loads (flexible loads) are determined iteratively, accounting for the effects of structural deformations. Optimization to minimize undesirable deformations may be performed. If significant response changes (Δ) occur, the flexible loads must be recomputed. Otherwise, refined analyses at the subcomponent level begins. Internal loads on panels are computed. If any panels are buckled, stiffness reductions occur and the loads must be redistributed in an iterative procedure such as the fuselage load redistribution described previously. If postbuckling stiffness reductions cause significant overall stiffness changes (Δ: buckle), it is necessary to return to the major component analysis to recompute the flexible external loads. If optimization at the subcomponent level (e.g., the multi-station approach discussed previously) causes significant changes (Δ: opt.), it may be necessary to recompute the flexible external loads and/or to restart the subcomponent analysis.

Once interaction at the three upper levels is complete, panel loads are defined and detail panel sizing begins (e.g., with POSTOP or equivalent). Postbuckling requires an iterative redistribution analysis for the skin and stiffener loads. Detailed stress, stiffness, and stability analyses are then performed. If panel sizing causes significant panel stiffness changes, it may again be necessary to return to the subcomponent or major component level. This multi-level interaction, along with complex analyses and iterative nonlinear procedures required at each level, provides a challenging problem. Interaction with nonstructural disciplines provides additional challenges. Multi-level optimization approaches (refs. 6 and 7) appear to be promising solutions to the problem.

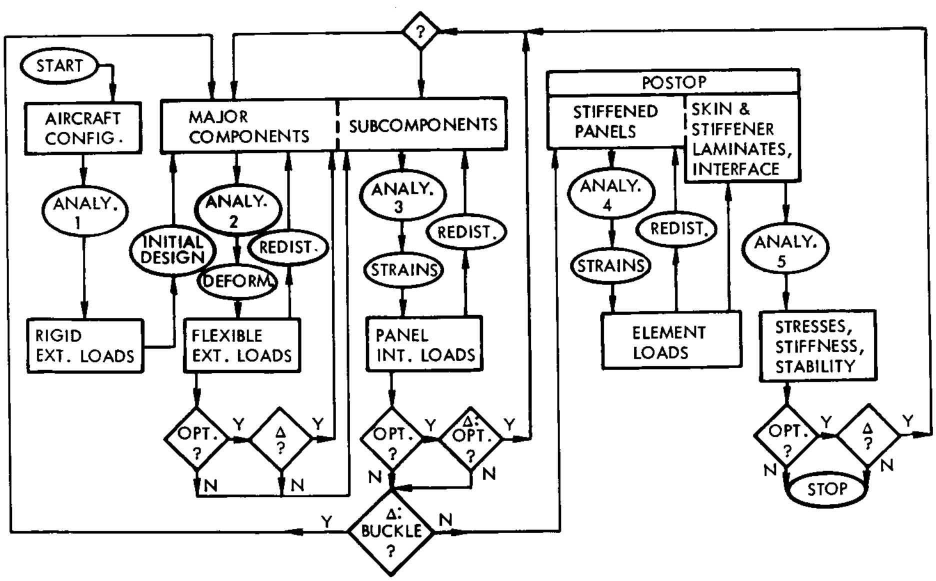


Figure 13.

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