A Mixed Optimization Method for Automated Design of Fuselage Structures

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A procedure for automating the design of transport aircraft fuselage structures has been developed and implemented in the form of an operational program. The structure is designed in two stages. First, an over-all distribution of structural material is obtained by means of optimality criteria to meet strength and displacement constraints. Subsequently, the detailed design of selected rings and panels consisting of skin and stringers is performed by mathematical optimization accounting for a set of realistic design constraints. As a result one obtains a procedure whose practicality and computer efficiency is demonstrated on cylindrical and area-ruled large transport fuselage.

Introduction

DURING recent years a number of advances have been made in the automated design of aircraft structures. Such advances generally fall into one of the following two categories: 1) Application of mathematical programing methods to relatively small components of real structures[1]; and 2) application of "engineering" methods, such as the Fully Stressed Design (FSD) procedure to large structural assemblies.[2,3,14]

This polarization of activities in automated design is a result of the large number of repetitive analyses required by the "exact" optimization methods of the first category. While progress in analysis methods and computer efficiency may gradually shift emphasis from the second to the first category, there are presently many design processes of large and complex structures where it is useful to combine the advantages of both methods. This paper presents such a mixed method of optimization and its implementation into an operational computer program for automated preliminary design of a stringer-skin-ring fuselage structure.

Problem Formulation

The preliminary design problem to which the procedure reported here is addressed is that of a fuselage built up of rings, stringers, and skin (Fig. 1) typical of a passenger or cargo transport aircraft. The following basic input is assumed to be known: a) Geometry of the fuselage surface. (Geometry is general enough to include noncircular, nonprismatic fuselages whose axis may not be a straight line.) b) Locations of cutouts in the surface. c) Location of the floor. d) Location of the wing carry-through structure. e) Loading cases in the form of self-equilibrated sets of forces including reactions of wing and/or landing gear. (Internal pressure is included among the loads.) f) Applied load factors and safety factors for each loading. g) Constraints on stresses and mechanical properties of structural material including tangent modulus data. h) Constraints on the fuselage displacements and rotations. i) Constraints on detailed structural dimensions.

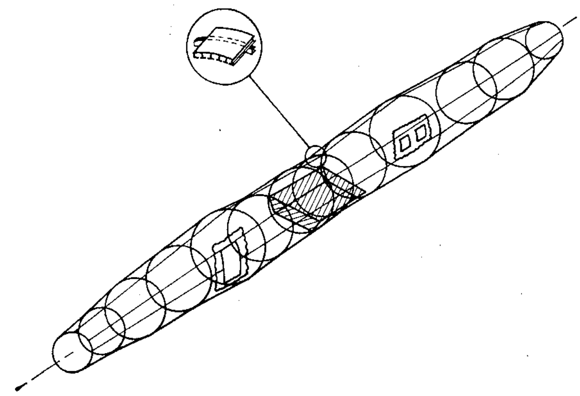

Fig. 1 Typical fuselage layout.

The desired output consists of the following information: a) Detailed cross-sectional dimensions of rings. b) Detailed cross-sectional dimensions of the stringers and skin. c) Fuselage structural mass.

The ultimate goal to be achieved is that of converting the input into output within the constraints imposed by strength, displacements, and minimum/maximum gage requirements in such a way that structural mass is as low as possible; that is, an optimum (primary) structural mass is desired. The program as formulated does not account for nonoptimum (secondary) mass, which can be quite large.

Rationale for the Method

The number of the geometrical parameters needed to describe fully a primary structure within a fixed fuselage geometry is easily in the tens of thousands if all details of individual stringers, rings, and skin are included. To design such a structure with the use of automated procedures, it is not practical to use mathematical programing which treats all these parameters as free variables, although such a procedure would be consistent from a mathematical viewpoint. One is, therefore, forced to reduce the problem size by means of simplifying assumptions to achieve acceptable results in a reasonable time.

Presented as Paper 72-330 at the AIAA/ASME/SAE 13th Structures, Structural Dynamics, and Materials Conference, San Antonio, Texas, April 10-12, 1972; submitted May 1, 1972; revision received September 7, 1972. The authors acknowledge the programing help of C. R. Williams and consultation on optimization received from W. T. Stroud, both of the NASA Langley Research Center.

Index categories: Aircraft Structural Design (Including Loads); Optimal Structural Design; Computer Technology and Computer Simulation Techniques.

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One frequently used simplification is to reduce the number of design variables through "slaving" or "linking" of variables. This includes linking imposed by fabrication requirements, as well as those resulting from judgment. It becomes extremely difficult, however, to devise a slaving hierarchy which would reduce the several thousand free design variables to a tractable number without losing their meaningfulness in the process. An alternative way of reducing the number of variables is to identify weak couplings between groups of variables, and break the complex problem into a number of simple subproblems by ignoring these weak couplings. This latter procedure was found attractive for the fuselage problem; namely, the determination of the over-all material distribution in the structure was separated from the detailed design of the cross-sectional dimensions. A fully stressed design method (FSD) was then used to determine over-all material distribution and mathematical nonlinear programing was used for detailed design.

In the FSD method, the fuselage structure is represented by a "lumped" model in which stringers, rings, and skin are represented only in a gross manner. The detailed design of the lumped model components is then carried out by mathematical programing with design variables whose resolution is sufficient to handle sizing of, e.g., Zee-stiffened skin panels.

Separation of the over-all and detailed design problems obviously neglects their coupling, which can be accounted for by iteration between the two models. In the iteration, repeated analyses are performed on new lumped models whose cross-sectional properties are adjusted in accordance with previous detailed design results, and the next detailed design is performed with the new internal forces resulting from the corrected lumped model. Since coupling between the models is weak, convergence of the iteration procedure is rapid.

While this procedure is appropriate for most aspects of the design, special features may be required for rings. It is well known that the thin wall fuselage bending stress distribution depends on the ring stiffness.[9,10] Numerical studies reported in Ref. 10 for fuselage models suggest that ring stiffness should be kept above a certain minimum value corresponding to the knee of the curve on Fig. 2 for maximum benefit from the ring. This minimal value of the in-plane bending stiffness of rings is incorporated into the procedure when the ring cross section is sized at the detailed design level.

A fuselage structure designed by the above approach satisfies the strength and minimum gage constraints, but it may violate the displacement constraints. As shown in Ref. 11, simply "beefing up" the design to meet stiffness requirements may be wasteful of material. Therefore, a separate procedure is needed to satisfy the displacement constraints. The procedure used is an extension of the "optimality criteria" proposed in Ref. 6. It employs the unit load mechanism to determine the contribution of each finite element to displacements which exceed constraints. Then, the optimum increments of cross section areas required for each element to reduce the displacements to allowable limits are calculated and applied to the lumped as well as the detailed models.

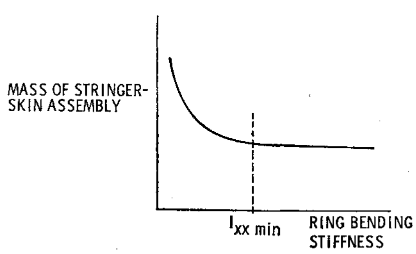


Fig. 2 Total stringer-skin assembly mass as a function of the ring in plane bending stiffness (Ref. 10).

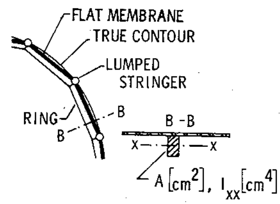


Fig. 3 Details of lumped model.

Modeling of the Fuselage Structure

The structural modeling includes two levels of detail: A lumped model for the over-all stress and deflection analysis, and a refined model for a detailed design of the rings and stringer cross sections and skin thicknesses. A discussion of the models and the transformation between them is given below.

Lumped Finite-Element Model

The lumped model, shown in Fig. 3, is comprised of flat quadrilateral membrane panels to represent the skin and solid cross section rods to represent stringers. Rings are modeled as planar polygonal frameworks. If large cutouts exist, they are accounted for by removing skin, stringers, and rings from the area leaving the cutout bounded by remaining rings and stringers. Each node contains four degrees of freedom: three mutually orthogonal displacements and a rotation associated with the in-plane ring bending. The node locations are determined by specifying the position of each ring plane and describing the shape of ring in that plane by either a circle or a combination of a circle sector and an analytical function.

The fuselage external loads are considered only on the lumped model. Consistent with the previously defined degrees of freedom, external loads at each node can include three concentrated forces and a concentrated moment associated with rotation in the ring plane. In addition, the internal pressure may be accounted for by automated conversion into a set of appropriate concentrated forces. Thermal loads may also be considered, including a uniform temperature increment as well as a gradient, for all the three types of the finite elements involved.

Refined Model

The refined model consists of an array of stiffened panels. Fig. 4 indicates the degree of geometrical detail included in the refined model. Secondary structural details such as brackets, gussets, rivets, and bend radii are not included. Loads on the components of the refined model are the internal forces generated by a finite-element analysis of the lumped model.

Lumping, Unlumping

An example of conversions between the lumped and refined models is illustrated on Fig. 5. The figure shows that each lumped stringer is formed by collecting stringers from the halves of two adjacent stiffened panels. Longitudinal and shear loads on the stiffened panel are obtained by extracting internal forces from a membrane bounded by a pair of half

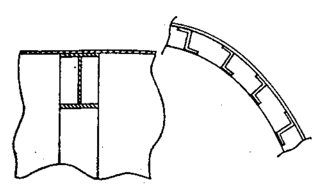


Fig. 4 Example sections of the refined model.

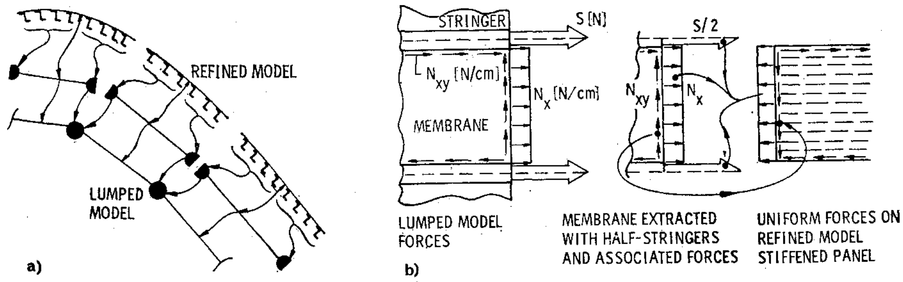


Fig. 5 Conversions between the lumped and refined models. a) example of element lumping, b) example of internal force unlumping.

stringers in the lumped model. Similar transformations are done for rings and hoop forces. Lumping is required only for a limited number of fuselage bays selected by the user for the detail design; properties for intermediate bays are obtained by linear interpolation.

Design Procedure and Program Organization

An outline of the design procedure is indicated in the flow chart in Fig. 6. Inputs are provided to the program for the geometrical data and all stringer and membrane cross sections are assigned an initial uniform area and thickness. All loading cases are multiplied by an appropriate load factor before entering the actual design procedure, while the safety factor is used, when appropriate, as part of the design procedure.

The design process is organized into four consecutive modules shown horizontally in Fig. 6: 1) determination of the minimum in-plane bending stiffness of the fuselage rings; 2) application of the FSD to the lumped model; 3) detailed design of the stringers, panels, and rings; and 4) strengthening of the structure to meet the displacement constraints, if necessary. The major operations are identified by blocks shown on the flow chart and listed in Appendix A. A further discussion of the modules is given below.

Module 1 plays a limited preparatory role. Its only result is determining a minimum stiffness limit for rings. Once the ring stiffnesses have been determined, module 2 generates a circumferential and longitudinal distribution of the material in the lumped model and the internal forces on all its elements. The FSD method is used to determine the material distribution in blocks 2 and 8. It was found that little improvement in the design occurs past the fifth or sixth iteration and, therefore, the number of FSD iterations in block 8 is usually limited to five. The allowable stress for the FSD procedure is kept at or below the yield stress.

On the other hand, a single FSD pass was generally adequate for block 2 to obtain ring stiffness since the design at this stage is too far removed from the end of the process for the question of convergency to have much practical significance. The process in module 1 stops when a test in block 4 detects that the knee of the curve shown in Fig. 2 has been passed.

Module 3 generates optimal detailed dimensions from the lumped mode, but this module also influences the over-all material distribution initially established in module 2. While module 2 determines over-all distribution by the simple FSD operation, module 3 carries out a more elaborate process of detailed design and lumping. In both modules new cross-sectional areas are generated in each pass through the loop but the over-all distribution of the structural material remains relatively constant.

Exceptions have to be made to the preceding procedure, such as in the case of a single bending load in the vertical plane (pullup maneuver, for example). The detailed design process can accumulate more material on the compression side of the fuselage than on the tension side, while the FSD makes no such distinction. The material distribution resulting from module 2 differs significantly from that generated in module 3 and convergence in the latter will not be accelerated by the use of module 2. For this case, it is better to bypass module 2 entirely.

Module 4 is executed only if it is found that the design generated by the first three modules does not meet displacement constraints. Its task is to reduce the critical displacements by increasing fuselage stiffness. This is accomplished by the addition of a minimum amount of structural material to satisfy displacement requirements.

The Component Subprograms

The fuselage design procedure is composed of several independent subprograms which have been isolated to facilitate modifications and subsequent updating. The characteristics of the major subprograms are described below and referred to as specific blocks in Fig. 6.

Analysis Program

The program which performs the structural analysis in blocks 1, 6, 11, and 15 is a general-purpose static finite-element analysis code. In the present configuration, a modified version of the "ELAS" program documented in Ref. 4 is used. The program employs a conventional matrix displacement method and Cholesky decomposition as the equation solving tool. It has a library of basic finite elements which is sufficient for analysis of a broad scope of practical aeronautical structures. Several modifications have been made to the version described in Ref. 4, including addition of a multiple load capability, changes to the stress computation routine, and use of regeneration of data to reduce communication with the peripheral storage.

Detailed Design Program

A mathematical nonlinear programing method is used in the detailed design program (block 9) for stiffened panels and rings. The general organization of the program is shown

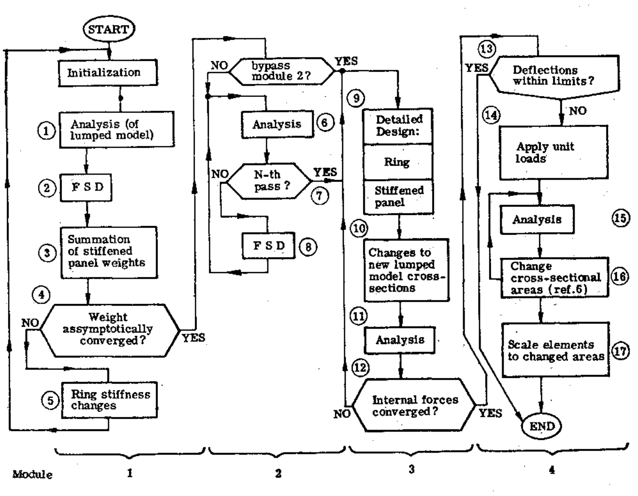


Fig. 6 Flow chart of the program (See also Appendix A).

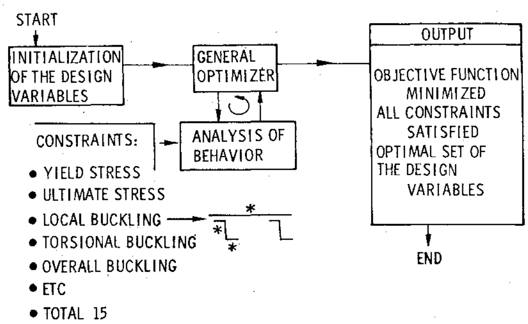


Fig. 7 General block scheme for computerized mathematical nonlinear programing in detail design module (See Appendix B).

in Figure 7 and a discussion of its details are given in Appendix B. Its main blocks include: 1) initialization of the design variables, 2) a mathematical programing algorithm, 3) a set of algorithms that analyze the system being optimized, and 4) the result of the optimization.

The capability of the detailed design procedure currently includes Zee or channel stiffened panels and rings of I or channel cross sections. The design variables for these cross sections are denoted by the numerals in Fig. 8. Mass (cross-sectional area) is considered as an object function for both stiffened panel and ring design; the design constraints include: 1) Minimum and maximum dimension limits. 2) Yield and ultimate stress. 3) Local buckling of the thin wall cross sections (for stringers and rings) and skin between stringers. 4) Stringer-skin torsional buckling. 5) General buckling of stiffened curved panel, with local buckling of the skin taken into account. 6) Over-all instability of the fuselage approximated as an orthotropic cylinder under action of bending moment, torque, and transverse force.

The preceding constraints are evaluated using handbook formulas or quickly convergent iteration equations which lead to short computer run time for a single pass through the complete constraint evaluation (0.05 sec of CPU time on CDC 6600). A full-fledged stability analysis using a program such as that described in Ref. 12 would require computer times of the order of 25 minutes. The constraint equations are similar to those in general use by designers for preliminary design.

Displacement Constraints Program

The program used in module 4 is coded according to the algorithm described in detail in Ref. 6. The basic idea of the method is that contributions of all finite elements to a deflection at a given point of the structure may be determined by a unit load method. An appropriate unit load is applied and well-known expressions involving appropriate stiffnesses and the internal forces due to the unit load and external loads are evaluated for each finite element of the lumped model.

This leads to a displacement expressed as an explicit function of cross-sectional areas of the membrane state-of-stress elements. For flexure elements, the function was modified to include cross-sectional radii of gyration. On the basis of the function, an optimal distribution of the cross-sectional area increments required to obtain the prescribed reduction in deflection is determined by means of a Lagrange multiplier technique.

Although this method yields a closed-form solution for each incremental cross-sectional area, it has an iterative character because internal forces due to the unit load and external load are implicit functions of all the cross-sectional areas in the statically indeterminate structure. It is, therefore, necessary to repeat the analysis after the cross sections are incremented in order to compute new internal forces. The process is repeated until these forces converge sufficiently and the displacement is reduced below the limit. The convergency is normally very fast and usually does not require more than two or three iterations.

Results

The program was run on an array of examples. A sample of results obtained is described in this section, including information on the computer performance.

Fuselage Structure Results

A series of calculations were carried out to demonstrate the use of the program as a tool for tradeoff studies in fuselage primary structures. A simple constant diameter fuselage (Fig. 9) having a length of 2800 cm and a diameter of 250 cm was adopted as a reference structure. The objective of the series of numerical experiments was to automatically generate fuselage structures and to compare masses for the reference shape and several alternate shapes including those with area ruling, with half ellipticity of their cross section and with changes to the slenderness (L/D) ratio. For all fuselages, the assumed loads correspond to those typical of an advanced technology transport of current interest.

The area-ruled fuselage was included since it is outside of conventional design experience, and statistical data are limited for evaluating the influence of area ruling on structural mass. Furthermore, a simple beam approach used in some preliminary design synthesis programs[13] is unable to account for the "kick load" effect associated with the area-ruled shape.

To perform the studies, the program was executed first with input corresponding to the reference fuselage whose shape was then varied for the several subsequent runs. In each run, eight reference bays (shaded in Fig. 9) were used for detail design. Figure 10 shows relative mass change as a function of area ruling, in terms of an area-ruling intensity factor measured by a longitudinal curve amplitude "b" and constants "α" and "β". The fuselage's longitudinal distribution mass is shown in Fig. 11 for both reference and area-ruled shapes.

Further illustration of the potential of the fuselage design procedure is provided by Figs. 12 and 13, which depict comparative mass changes when the cylindrical reference fuselage is given a degree of half ellipticity and its slenderness ratio (L/D) is varied. The mass increases slightly when the half ellipticity is added because skin in the upper half of the

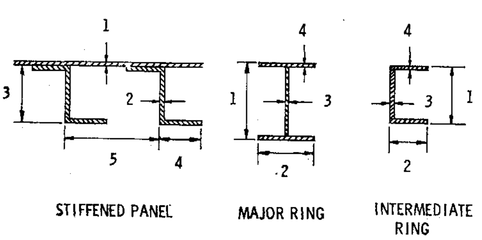


Fig. 8 Design variables.

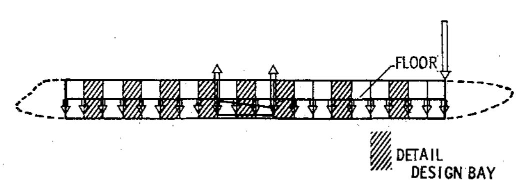


Fig. 9 Reference cylindrical fuselage and an example of a loading force.

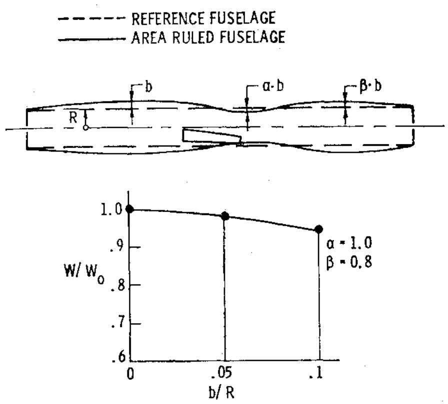


Fig. 10 Sensitivity of the fuselage mass to the area ruling.

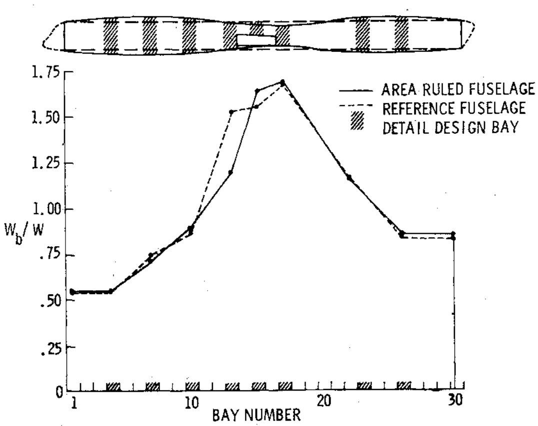


Fig. 11 Longitudinal distribution of fuselage, mass. W_b single bay mass, W total mass of reference fuselage.

fuselage is designed by minimum gage in the reference fuselage (in this particular case) and, therefore, cannot be made thinner due to the decrease in pullup bending stress caused by the fuselage shape change. Figure 13 shows a mass curve plotted through three data points where slenderness ratios were selected to correspond to the following seating arrangements for the same total number of passengers: three-aisle-three, three-aisle-two, two-aisle-two, corresponding to diameters of 346 cm, 298 cm, 250 cm, respectively.

A sample of the detailed dimensions generated by the program is given in Fig. 14. The detailed design program was checked using an existing transport aircraft wing panel which was designed by conventional methods and verified experimentally. The data for the panel design are shown in Fig. 15. The actual cross section design is superimposed on the one generated by the program for the same input data. It is seen

Fig. 13 Influence of the slenderness ratio (length/diameter) on the fuselage mass. $(L/D)_0$ corresponds to two-aisle-two seating arrangement.

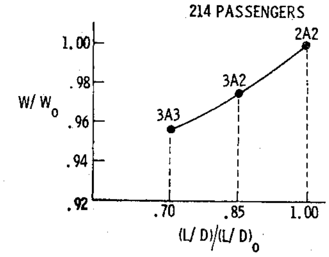
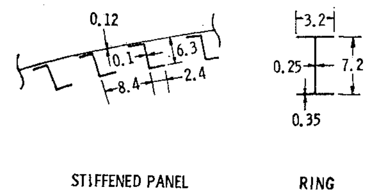


Fig. 14 Sample of the detailed dimensions in (cm) output by the program.



that the program creates a design very similar to and somewhat lighter (11%) than the one used in the vehicle. In 12 similar tests, it was found that masses of the automatically generated panels were consistently 0.84–0.92 of the mass of actual panel designs. Since the program currently does not account for standard gages or test-theory correlation, these comparisons provide confidence of the capability to produce reasonable preliminary designs, rather than an indication of mass decrease. The program used 5 sec of CPU time on CDC 6600 and approximately 600 trial configuration to establish the optimal stiffened panel cross-sectional dimensions shown in Fig. 15.

Computer Requirements of the Program

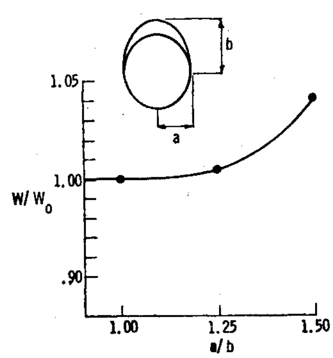
Computer requirements of the program are characterized by the following example. A representative fuselage consisting of 30 bays was modeled by 12 membranes, 12 lumped stringers and ring sectors in each bay. This corresponds to about 1900° of freedom and 1200 finite elements. Eight bays were selected for detailed design for three load conditions (two cases of bending loads and internal pressure). All the component programs were overlayed using the link editing technique.

The resulting core requirement was 90K. Total CPU time was 1950 sec on a CDC 6600 and its distribution between four modules defined in Fig. 6 was 480 sec, 585 sec, 540 sec and 300 sec, respectively.

Summary and Conclusions

A computer program for the automated preliminary design of an aircraft fuselage has been presented. The program may be characterized as follows: 1) A fuselage structure with

Fig. 12 Influence of the half ellipticity on the fuselage mass. W_0 reference fuselage mass.



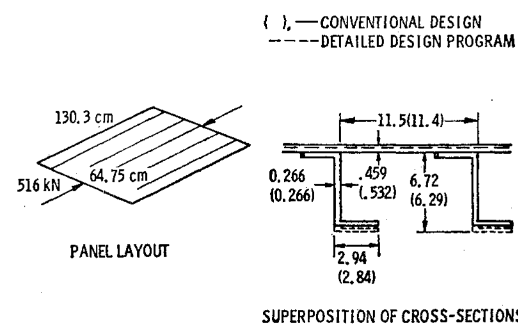


Fig. 15 Comparison of the detailed design given by the present program with the conventionally designed stiffened panel.

fairly general geometry can be automatically designed taking into account numerous load conditions. 2) Nonlinear mathematical programing is used in all operations where economically feasible; standard engineering FSD methods are employed in the remainder of the process; the number of repetitive analyses of the whole fuselage is held to a minimum. 3) Coupling between the ring stiffness and stringer-skin stresses is taken into account in the design procedure in order to obtain a structurally efficient design. 4) A comprehensive array of local buckling and yielding conditions is accounted for in the local detailed design of the skin and rings, including collapse loads. 5) Dimensions generated by the program are considerably more detailed than is customary at the preliminary design stage. 6) Modular organization of the program facilitates extension to a variety of types of skin-stringer or ring assemblies as well as to integrally stiffened and sandwich skins, and different materials including composites.

The essential merit of the method presented lies in the degree of automation carried out in bringing fairly detailed design and analysis into the preliminary design stage. Thus a "preliminary detailed" design becomes economically feasible. It extends the scope of configurations and alternative designs that may be tried and evaluated. Program applications are illustrated with examples of automated structural design and trade-off studies for typical transport aircraft fuselages, including the effect of area ruling on the total fuselage mass and its distribution for potential advanced technology vehicles.

Appendix A: Blocks of the Flow Chart (Fig. 6)

Module 1

1) Analysis of the lumped model. 2) FSD modification of the dimensions of the panels and lumped stringers. 3) Summation of the weights of the panels and lumped stringers. 4) Test of whether the total weight summed above has decreased sufficiently below the knee of the curve shown in Fig. 4. 5) Increase of the ring in-plane bending stiffnesses to drive the weight farther down the curve shown in Fig. 4. Module 1 operation ends when test 4 is met. The result of the module is minimum requirements on the ring stiffnesses.

Module 2

6) Analysis 1. 7) Test whether to continue the FSD procedure. 8) FSD. Module 2 operation ends when test 7 is met. The result is a set of internal forces.

Module 3

9) Detailed design of the stiffened panels and rings at several user-determined locations on the fuselage based on internal forces calculated in module 2. 10) The new cross-sectional properties are assigned to the elements of finite-element lumped model on the basis of the above detailed design results. 11) Analysis of the lumped model (as in block 1). 12) Convergency test on the internal forces on the finite elements of the lumped model, and transfer to block 9 if the convergency criterion is not satisfied. Module 3 operation ends when test 12 is met. The results of the task are detailed cross-sectional dimensions of the stiffened panels and rings.

Module 4

13) Test of whether the displacements of the fuselage emerging from task 3 satisfy the constraints. If they do, task 4 is bypassed. 14) Unit loads associated with critical displacements are applied to the lumped model. 15) Analysis of the lumped model for the above unit loads. 16) Corrective increments of the cross-sectional dimensions of the lumped model finite elements are computed using a method described in Ref. 6. Since this method is iterative, it requires to loop through step 15 until a convergency criterion built into that method is satisfied. 17) Cross-sectional dimensions of the stringer, rings, and panels in the detailed model are changed to accommodate the corrective increments of the cross-sectional areas.

Appendix B: Description of Detailed Design Program

The general organization of the detailed design program is illustrated in Fig. 7. As shown in the figure, block 3 receives values of the design variables from block 2 and returns to block 2 of the corresponding value of the object function and information on satisfaction or violation of the constraints. The loop between blocks 2 and 3 is continued until convergence occurs. The results in block 4 comprise a minimized object function, a corresponding set of design variables, and numerical evidence that all the constraints are satisfied. A program denoted AESOP[5] is presently used in block 2. It is a general-purpose optimizer containing a selection of several optimization algorithms. The "adaptive-creep-with-pattern" procedure is the algorithm used since it was found to be the most efficient in this application.

The advantage of using a general-purpose optimization program (block 2, Fig. 7) in the detailed design module is to facilitate accommodation of various types of cross sections.

Under this organization the only block of Fig. 7 which has to be tailored to the type of cross section to be optimized is block 3, which contains the behavior analysis subroutines. Addition or exchange of these subroutines is simple since the interface of block 3 with block 2 is entirely general, and independent of the content of block 3.

Mass (cross-sectional area) is considered as the objective function for both stiffened panel and ring design. The presently implemented list of constraints is as follows: 1) Minimum and maximum limits on the design variables. 2) Yield stress (using Huber-von Mises effective stress where appropriate). 3) Ultimate stress. 4) Skin buckling between stringers taking into account interaction of all three components of the planar state of the stress. The buckling is allowed at a value of the load factor, which may vary from 1 to the ultimate load factor and is an input parameter. 5) Stringer (ring) web local buckling. 6) Stringer (ring) flange local buckling. In this condition as well as in two previous ones the classical formulas for instability of rectangular plates are used. 7) Stringer-skin torsional buckling (Ref. 15, p. 241, Eq. 5.56). 8) Buckling of stiffened curved panel with skin between stringers locally stable (Ref. 7, p.16 and Fig. 3, also Ref. 8). 9) As above with skin locally buckled between stringers and appropriate effective width computed according to the skin postbuckling behavior theory.[16] 10) Over-all buckling of the fuselage approximated as an orthotropic cylindrical shell subjected to a transverse force (tension field) (Ref. 7, p. 50, Eq. 100). 11) Over-all buckling of the fuselage approximated as above subject to a simultaneous action of bending moment and torque (Ref. 7, p. 51, Eq. 103 and p. 63, Table 1).

The panels to be designed are considered under the action of two orthogonal forces and shear; a number of simplifications are incorporated in the design process. For example, all constraints are evaluated using handbook type closed formulas or quickly convergent iteration equations. Critical stresses that exceeded plasticity limits are modified through the use of tangential moduli, and local buckling modes are assumed to be independent. Closed-form orthotropic shell formulas are used for the over-all fuselage buckling analysis (constraints 10 and 11) neglecting both circumferential and longitudinal variations of geometry and internal forces.

The constraints are divided into failing (ultimate collapse) and nonfailing category. In the former, which consists of the constraints 3, 5, 6, 7, 8, 9, 10, 11, loads on the panel are multiplied by a safety factor (1.5) to increase them to an ultimate level while loads remain at the applied level for constraints

2 and 4. For the rings, constraints 1, 2, 3, 5, 6, from the previous list are used considering in-plane bending moment, axial, and shear forces.

The ring cross section is designed at circumferential nodes of the lumped model. Since lumping leads to discontinuities of the axial and shear forces (but not the bending moments) in the ring at these points, these discontinuities are smoothed out for the cross-section design purposes by averaging. Averaging is not performed, however, at nodes where an external concentrated load occurs or where other structural components branch off the stringer-skin-ring assembly (such as floor members or wing carry-through structure). At such points, two ring cross sections are generated.

The program automatically adjusts the stringer spacing so as to maintain an integer number of stringers on a panel. In addition, for manufacturing reasons, stringer number in each longitudinal tier of panels should remain constant. Those numbers are dictated by detailed design executed for each panel of a "master" bay (usually the one immediately aft the rear wing spar) and are maintained along the fuselage. For other bays the stringer spacing is not a free variable, but a fixed parameter determined by that number and the panel width which may vary due to fuselage taper.

Since variation in ring spacing has several practical limitations, the present version of the program restricts ring spacing to a user-determined input parameter. With a minor modification, it could be made a design variable in the optimization process. This would require joining the rings and stiffened plate constraints and object functions in block 3, Fig. 7, but could be easily accomplished due to the modular organization of the program.

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