BOEING

(NASA-CR-157506) LOAD INTRODUCTION PROBLEMS
IN SPACECRAFT STRUCTURES, (Boeing Aercspace
Co., Seattle, Wash.) 59 p

N78-78237

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Technical Proposal

LOAD INTRODUCTION PROBLEMS IN SPACECRAFT STRUCTURES

D180-17972-1

Prepared for
Langley Research Center
National Aeronautics and Space Administration

March 1974

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1.0 INTRODUCTION AND SUMMARY

The design goal for all aerospace structures is to provide a structure with adequate integrity and minimum weight and cost. For most conventional structures, there is a wealth of experience-proven analysis methods to ensure structural integrity. There is also an extensive background on minimum structural weight optimization. In particular, approaches such as simultaneous buckling mode failure optimization have received considerable attention. The use of the computer has added structural configuration optimization by coupling finite-element analysis with numerical optimization procedures. However, there has been little formal cost/weight optimization and there is a growing need to expand development in this area. Cost/weight optimization has been hampered by lack of uniform industry cost-estimating methods.

This program will address structural weight and cost optimization of sandwich panels for two structural configuration problems of concentrated load diffusion (fig. 1-1). The object will be to develop preliminary design information that will allow a designer to select the best cost/weight structural form for these two configurations. Configuration A is representative of the distribution of a concentrated load to a uniform compression reaction. Configuration B is representative of the distribution of a concentrated load to a uniform shear reaction. Both cases will be studied for various load levels and panel aspect ratios.

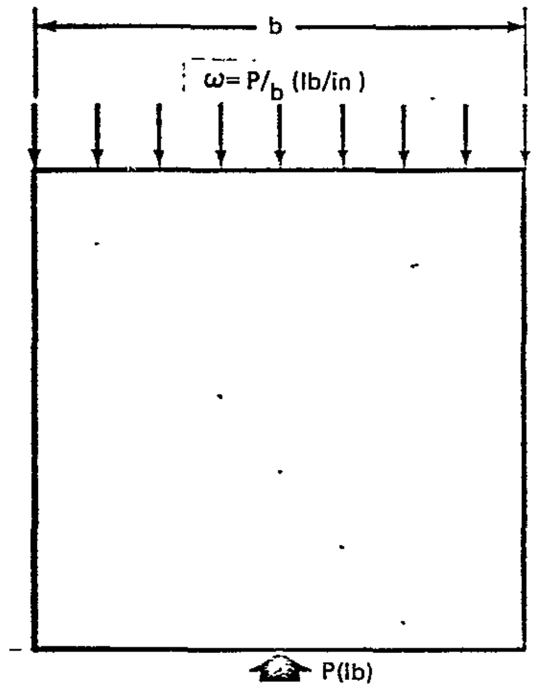
The design information produced from this study can be used for the structural design development of the space tug. The three current industry concepts of the space tug are shown in figure 1-2.

Configuration A (uniform compression reaction) has potential application to the space tug structure in the areas of forward skirt, intertank structure, and aft skirt structure. Typical forward skirt structure accepts concentrated loads from the payload docking/retrieval systems. The aft intertank structure of one concept is representative of configuration A due to the concentrated loads introduced by the shuttle docking interface. The aft skirts of other concepts could also consider this configuration applicable to the shuttle docking interface loads.

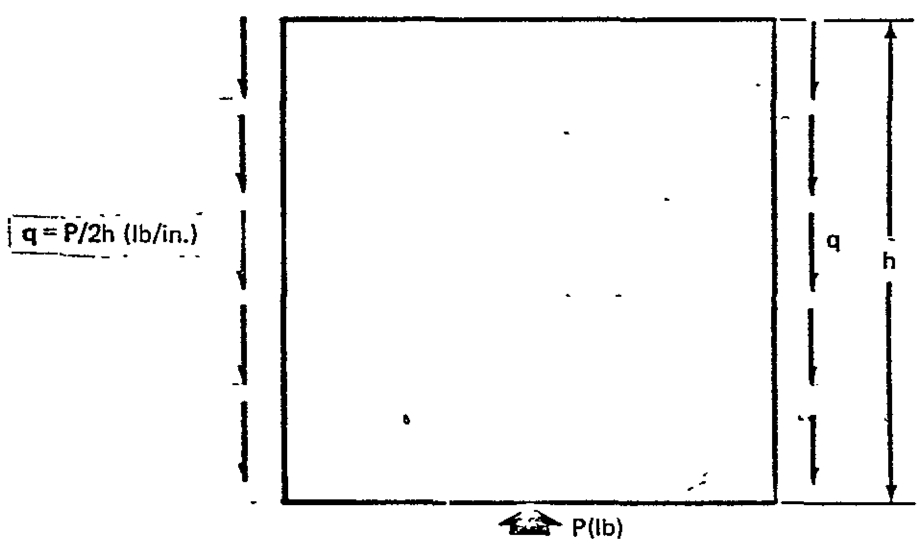
Configuration B (uniform shear reaction) has potential application to space tug concepts that use space/framework-type structure to introduce payload docking/retrieval loads into shell structure. It may also have application to those concepts using suspended tankages, which shear the tankage loads into the outer shell. The shearing of engine thrust loads into the tankage and shell structure is also a potential application area.

The space shuttle program is designed to provide the nation with a more cost-effective means of placing significant payloads in orbit. As part of this program, the space tug must also be designed with cost-effective goals. This study can provide NASA with additional information and design tools to accomplish this design task effectively. The information will also be of use to the aerospace industry in design of other spacecraft and aircraft systems.

The analysis approach to develop sandwich panel design information will be to build a data bank of structural information from an investigation of three material forms. This data



CONFIGURATION A—UNIFORM COMPRESSION REACTION



CONFIGURATION B—UNIFORM SHEAR REACTION

Figure 1-1.—Structural Configurations for Reaction of Concentrated Loads

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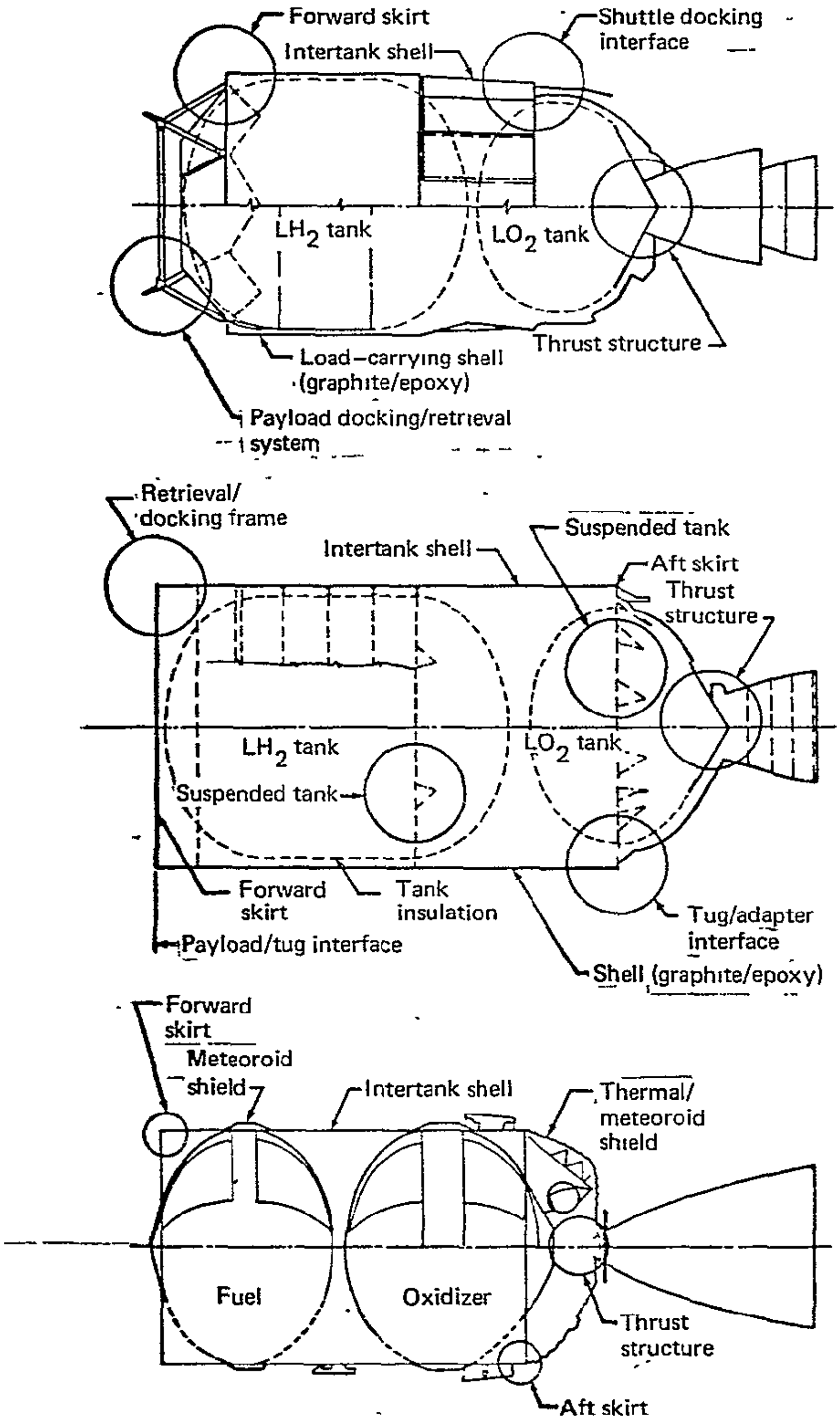

Figure 1-2.–Three Space Tug Structural Configurations

bank will contain detailed structural information for both weight/strength and cost/weight relationships. Finite-element models for these material forms will be established in a computer program format that contains a program direct link to a minimum-weight numerical optimization program. The material distribution obtained for the two configuration cases, load levels, and aspect ratios will then be checked for general stability. The final weight/strength designs will then be prepared in preliminary design layout format to incorporate practical design considerations and provide a base for design cost estimating. The estimated cost of each will be used along with its weight to establish the cost/weight merit function. A value parameter will be used in the cost merit function such that it can be varied over a wide range to cover space, military, and commercial vehicle systems.

The program will be conducted in two phases (see section 2.0). The first phase, of 12 months' duration, will develop the design information. This preliminary design information will be displayed in a format useful to a design engineer. In this phase, design data areas of potential application to the space tug will be identified.

The second phase, of 6 months' duration, will select, design, and fabricate a simulated space tug structural component. The design selection and detail analysis will use the information generated in phase 1. The component will be designed for fabrication using advanced composite materials. The fabrication steps will be monitored to provide additional confidence in the cost-estimating procedures used in establishing the cost inputs to the cost/weight trades of phase 1.

To provide the program with depth in both spacecraft and aircraft structures, it will be staffed with personnel from both the Boeing Aerospace Company and the Boeing Commercial Airplane Company. The Boeing Aerospace Company structures personnel, with their extensive background in space structure studies such as the space shuttle orbiter, solid rocket booster, and space tug, will add validity to the space tug potential application of program-developed data. The Boeing Commercial Airplane Company personnel, from their Boeing 700-series experience, will add commercial cost-effectiveness expertise. Both companies have considerable IR&D and NASA-contract-developed backgrounds associated with all elements of this program.

2.0 WORK STATEMENT AND PROGRAM SCHEDULE

The Boeing Company will perform an analytical study of aerospace structure whose function is to diffuse concentrated loads to either a uniform inplane compression reaction or a uniform inplane shear reaction. The study will produce preliminary design structural information for these two configurations. The application of this information to spacecraft structure will be explored.

The program will be conducted in two phases. Phase 1 will be an analytical study with a duration of 12 months. Phase 2 will apply the phase 1 information to the design and fabrication of two, 30- by 35-in., lightweight sandwich panels characterizing the structural concept most applicable to spacecraft structure. This phase will be of 6 months' duration.

The reporting planned and the program schedule are shown in figure 2-1. Specific tasks are defined in sections 2.1 and 2.2.

2.1 PHASE 1–STRUCTURAL ANALYSIS AND OPTIMIZATION

a) The structural form to be covered in this study will be sandwich panels with no inplane load capability in the core and with face sheets of metal (aluminum), graphite composite, and hybrid composite (high-strength and high-modulus graphite, PRD-49, and fiberglass).

b) Structural and cost criteria will be established for the study.

c) NASA approval of the criteria will be obtained before analytical work is initiated.

d) An analytical study of sandwich panels will be conducted to cover the following two structural configurations:

   1) Diffusion of a concentrated load to a uniform inplane compression reaction.

   2) Diffusion of a concentrated load to a uniform inplane shear reaction.

e) Study consideration will be given to both heavily loaded and lightly loaded structural configurations.

f) Included in the study will be conceptual design consideration of such items as.

   1) Non-optimum-weight items (i.e., minimum gage, etc.)

   2) Manufacturability

   3) Repairability

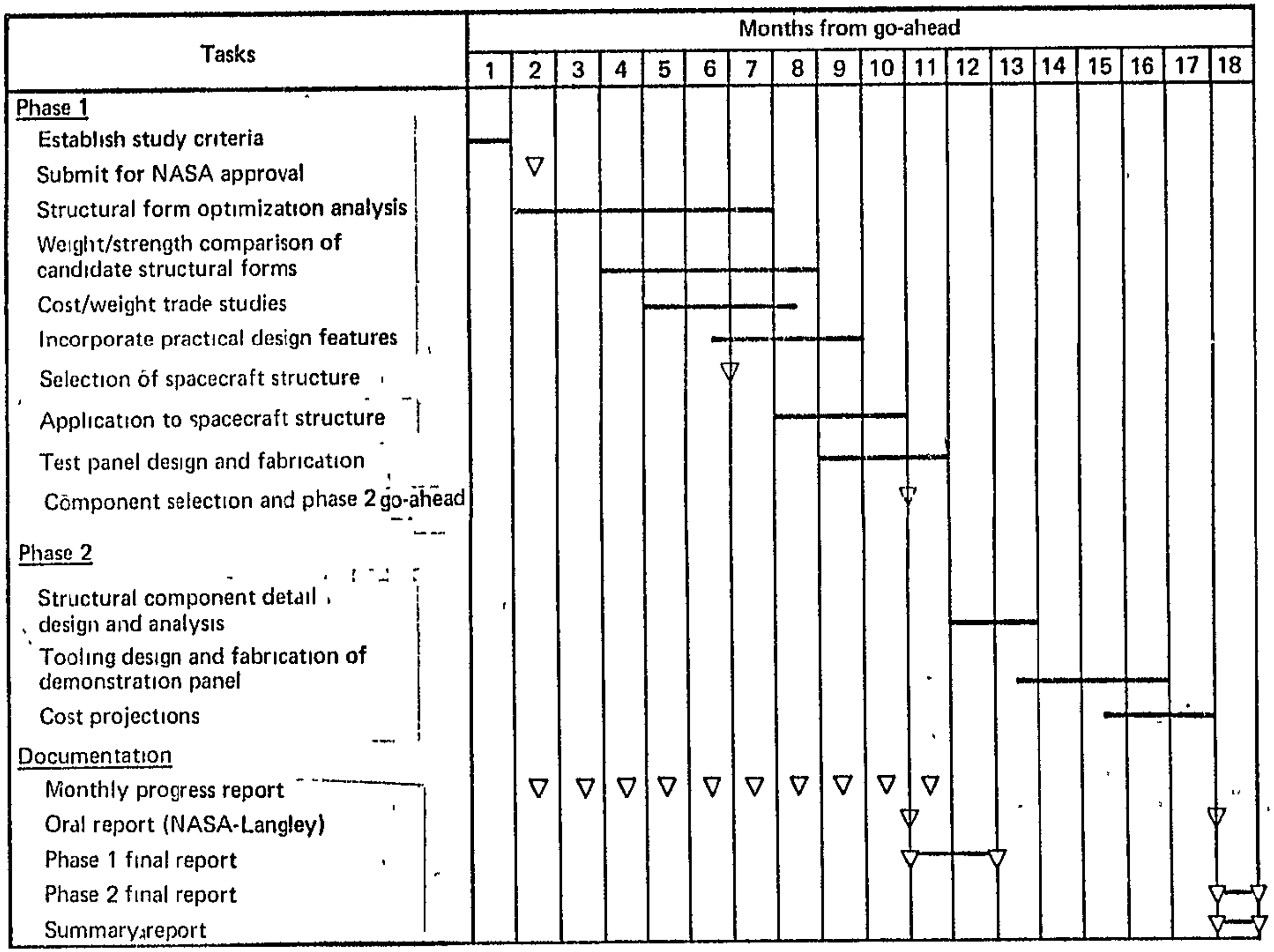

Figure 2-1.—Program Schedule

4) Maintainability

5) Cost

g) Weight/strength relationships using either structural indexes and/or merit functions will be developed for preliminary design application.

h) Cost/weight relationships using a value parameter and merit function will be developed for preliminary design application.

i) Representative spacecraft structures provided by NASA will be reviewed to select potential areas for application of study results.

j) Two small sandwich test panels (12 by 12 in.) with graphite faces and honeycomb core will be fabricated and shipped to NASA for test.

2.2 PHASE 2—FEASIBILITY DEMONSTRATION PANELS (FOR POTENTIAL TEST)

a) A representative spacecraft structural component will be selected for study using results of phase 1. The spacecraft structural arrangement will be provided by NASA.

b) The component selection will be submitted for NASA approval.

c) The selected component will be designed and analyzed to a more detailed level for the specific loading environment.

d) Detailed R&D-hardware-type drawings of the final structural form selected will be made for fabrication of a part representative of a spacecraft structural application.

e) The final design drawings will be submitted to NASA for approval prior to release to the shop for fabrication.

f) Composite materials will be used where appropriate to fabricate the component.

g) Cost of fabrication will be monitored and compared to cost predictions.

h) Part will be shipped to NASA on completion.

2.3 REPORTING AND TRAVEL

Monthly status reports, oral reviews, and final reports will be provided per the schedule (fig. 2-1).

Provisions have been made for two 2-man trips to NASA-Langley during the duration of the contract. These trips will occur at the end of each program phase.

3.0 BACKGROUND TECHNICAL DISCUSSION

3.1 INTRODUCTION

All vehicles, including their structure, are produced in a competitive environment, which necessitates that the performance requirements of the user and the systems cost be the guiding design criteria. For aerospace vehicles, weight is a key performance parameter Structural weight is a major part of the vehicle weight: therefore, the design and analysis technology associated with its minimization continues to receive considerable attention. It is significant also that for aerospace vehicles a high dollar per pound value can be allowed for weight reduction. The cost of producing a highly efficient structure can be recovered through savings in overall system operating cost.

The objective of this study is to develop additional preliminary design tools that will aid the designer in developing the most weight/strength- and cost/weight-effective vehicle structure. The two specific loading configurations to be studied are shown in figure 1-1. Configuration A is representative of the distribution of a concentrated load to a uniform compression reaction. Configuration B is representative of the distribution of a concentrated load to a uniform shear reaction.

The first step in the design of any structure is understanding its functional requirements. These requirements are displayed for the structures engineer in the structural requirements (noted above) and in the design criteria established for ensuring that the structure will meet its minimum functional requirements. This study, as shown by the generalized program flow charts (fig. 3-1), will be initiated by establishing the criteria under which it will be conducted.

—The second step will be to select the structural materials and forms to be reviewed. For this study, the definition of structural form will include the selection of materials, cross-sectional geometries, and joining techniques necessary to meet the loading configuration and criteria requirements.

3.2 PHASE 1–STRUCTURAL OPTIMIZATION ANALYSIS

3.2.1 Criteria

The applicability and validity of the results of this study will depend on the constraints placed on the data development and the resulting usable ranges of the developed information. The study will cover and use such data, procedures, and information as:

a) Material properties and allowables

b) Analysis methods and procedures

1) Structural/stress analysis

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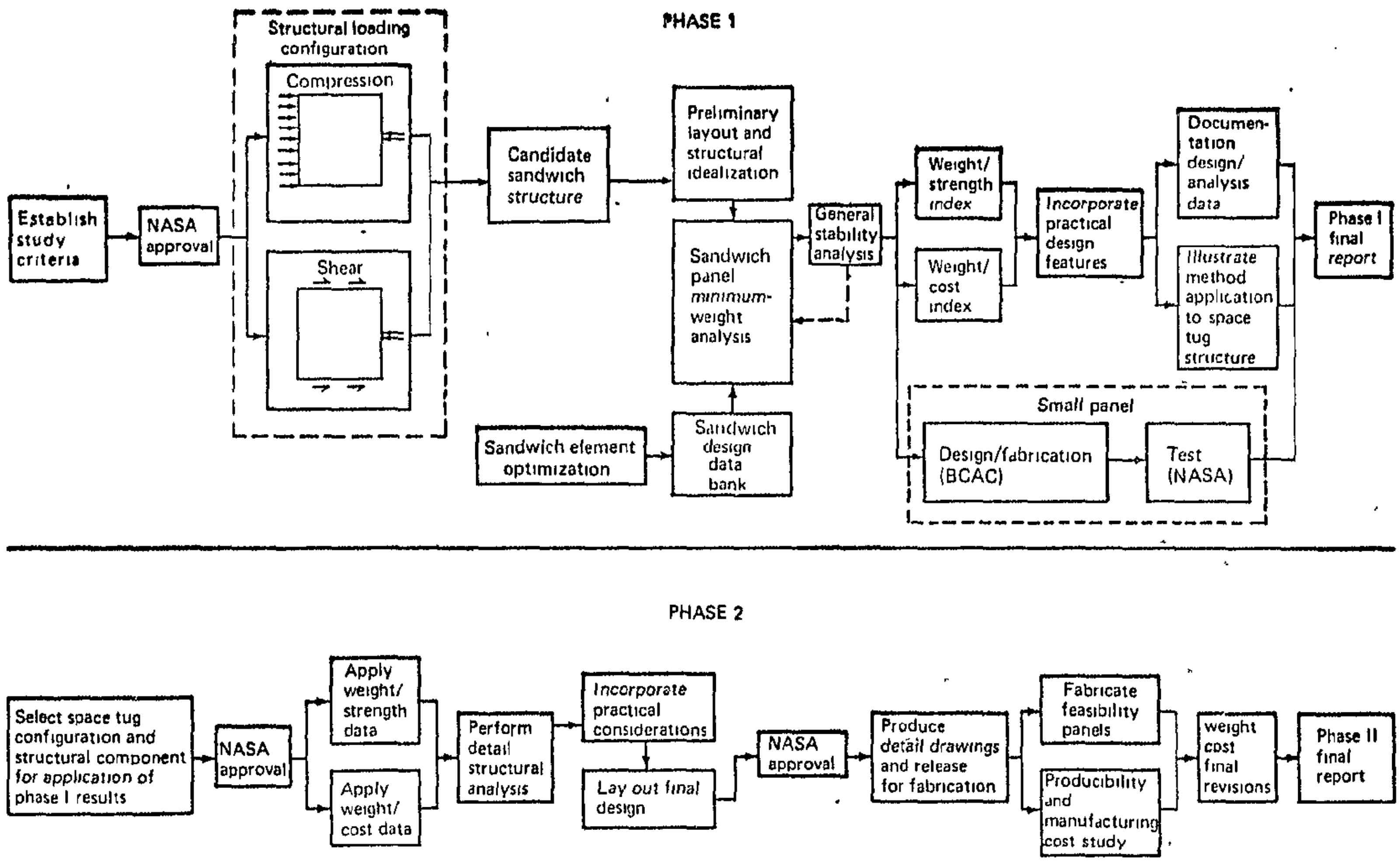

Figure 3-1 –Program Plan Flow

2) Weight estimation

3) Cost estimation

This section addresses some of the above items and contains most of the criteria that will be used in this study. However, as noted in figure 3-1, the criteria for the study will require NASA approval.

3.2.1.1 Loading Environment

For the structural analysis, two types of loading environments will be considered. A concentrated inplane load will be diffused to (1) an inplane uniform compression reaction and (2) an inplane uniform shear reaction. A range from light to heavy loading will be considered. For the structural idealization, the panels will be subjected to an applied uniform loading and reacted with a concentrated load.

3.2.1.2 Panel Edge Boundary Conditions

The boundary constraints strongly influence the load distribution and hence the structure. All edges of the panel are assumed to be simply supported with no inplane displacement restrictions. Using panel symmetry and appropriate displacements at the load axis of symmetry, only one-half of the panel will be used in the structural optimization. For the general stability analysis, the edge constraint on the load axis of symmetry may be modified to permit symmetrical buckling modes.

3.2.1.3 Failure Modes

The panels will be designed with emphasis on meeting strength and elastic stability requirements. The panels will be designed to react the applied loading without permanent deformation or elastic instability. Local instability failures associated with each structural form will be evaluated. Examples of failure modes for particular structural forms are given below:

a) Sandwich panels

1) Panel instability

2) Face sheet wrinkling, intracell buckling, shear crimping

3) Yield

b) Composite structure

1) Maximum allowable fiber strain

2) Maximum laminate stress level

3) Hill-Tsai (ref. 1) failure criteria

4) Panel instability

5) Interlaminar shear

3.2.1.4 Minimum-Weight Design

The optimum structural panel is a function of weight and cost, as discussed in section 3.2.4. The selection of structural forms (sec. 3.2.2) considers cost and practical design features. A minimum-weight structural analysis will be performed on these selected structural forms to assist in establishing an efficient design.

For the weight comparison of structural forms, panel weight will include the total weight within the natural panel boundary. These weight items consist of:

a) Edge closures for free edges

b) Joints to transfer load to supporting structure

c) Concentrated load introduction member

d) All other nonoptimum weight items associated with a practical structure (see sec. 3.2.5)

Edge closure members, joints, and load introduction members must be included in the weight comparison, particularly for small panels where these nonoptimizable weight items can contribute significantly to the panel weight. The portion of the load transfer joints and load introduction member outside of the natural panel edge will not be included in the total panel weight.

3.2.1.5 Cost

The most effective display of cost/weight data is that which can be used by most aerospace vehicle systems. A cost merit function (see sec. 3.2.4) will be used to clearly display the optimum cost/weight concept for a wide range of vehicle systems.

With these basic criteria for effective cost/weight data display must go criteria and/or ground rules for estimating the design concepts cost. It must be recognized that each system has costs related to production costs and to system operation and maintenance (O&M) costs. As an example, the O&M costs for commercial aircraft are well defined by rules for establishing direct operating costs. However, similar costs for military and space systems are not as well defined. Since these costs directly affect establishing a value of weight to vehicle performance, it is suggested that with NASA's approval, a range of values for space, military, and commercial vehicles be established.

The cost to produce the structural concept will be established using standard preliminary design estimating practices. For each system, unit quantity, production rate, standard learning curves, complexity, and cost rationg ground rules need to be established

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to give consistency to the cost estimate. In addition, it is suggested that nonrecurring cost be evenly spread over the total number of units selected. The recurring cost, which will be summed with the nonrecurring cost and the material costs, should be based on the average unit cost over the total number of units.

Material costs for metals are reasonably stable; however, advanced composite material costs continue to show a potential for cost reduction with time. The advanced composite costs will therefore be selected from a reasonable cost projection curve and a time (calendar year) approved by NASA (fig. 3-2).

These items will be established and submitted to NASA for approval in the study program.

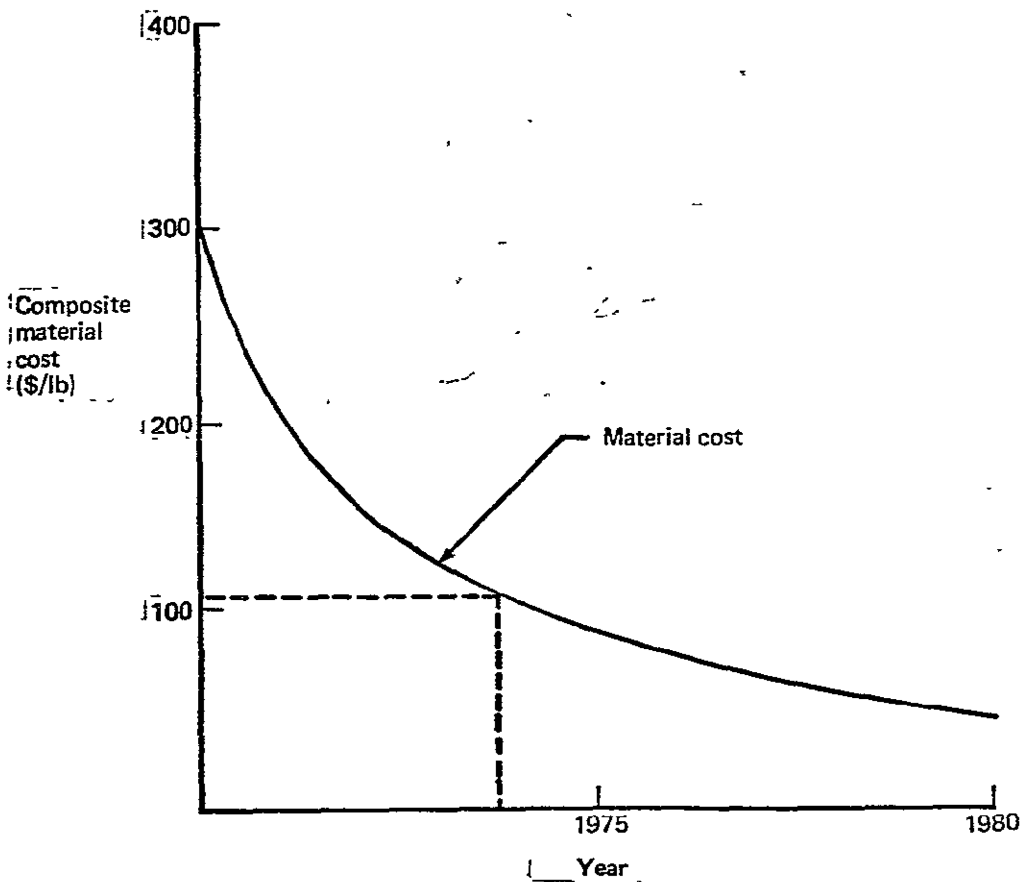


Figure 3-2.—Advanced-Composite Material Cost Projections

3.2.2 Material and Structural Form Selection

Material selection and application require a careful examination of candidate materials with regard to their ability to perform in specific structural configurations, material and fabrication costs, and mechanical and physical properties. No single material has all of the attributes necessary to produce an optimum structure; thus, optimum structure ideally consists of different materials, each selected with regard to the unique requirements of the component.

Baseline metal panels to be evaluated in this program will be selected from one of the following materials: aluminum alloys 2024-T3, 7075-T6, and 7475-T61; titanium alloy 6Al-4V; and stainless steel alloys AFC77 and AFC77B Isotropic material properties as defined in MIL-HDBK-5 and Boeing design manuals will be considered.

The principal advanced composite materials to be considered will be high-modulus, high-strength, and intermediate-strength graphite/epoxies. Secondary consideration will be given to hybrid systems where the load level and stiffness requirements indicate a weight savings is possible.

Composite materials and metals will be analyzed as linearly elastic materials. The elastic properties and strength characterization of composite laminates will be compatible with the *Advanced Composites Design Guide.* Symmetrical, composite laminates with a standard ply orientation ofof 0°/±45°/90° will be considered for this study. The number of individual plies in each orientation will be allowed to vary; however, symmetry of layup will be maintained. Optimization effects of ply orientation will be evaluated by means of sensitivity studies.

The following factors will be considered in the material selection:

a) Basic mechanical properties

b) Environmental stability

c) Ease of handling during fabrication processes

d) Present and projected material costs

Boeing proposes to achieve the objectives of this program by studying a series of sandwich geometries. Each geometry will be evaluated and ranked in terms of structural efficiency, producibility, and cost. General layouts of each geometry will be made to evaluate practical design features.

Two joining techniques will be evaluated. Methods of interest will be the conventional forms of mechanical fastening and adhesive bonding.

3.2.2.1 Isotropic Materials

Three principal isotropic materials will be evaluated. Aluminum, titanium, and steel alloys have received wide application in aerospace structures and provide a range of extensional modulus values from 10×10^6 to 30×10^6 psi The specific stiffnesses of these three systems are practically equal. However, if the configurations are subject to local instability failure, aluminum can be more efficient at low end load ranges. Some general considerations for the materials to be reviewed and their potential application in this study are given below.

Steels—Steel alloys, with their high absolute strength and stiffness, are ideally suited for highly loaded structure where section geometry does not introduce a stability-critical failure. In lightly loaded structure, the resultant thinner sections of steel become stability critical, and steel structures are inherently less efficient unless the thinner sections are more thoroughly stabilized against buckling failure. From an environmental standpoint, not all of the alloy systems have sufficient inherent corrosion resistance. Many require protection by painting, plating, etc.

Aluminums—The aluminums have been nearly ideal aerospace structural materials, having the lowest absolute density of the three major material systems, equal specific stiffness, nearly equal specific strength, inherent corrosion resistance, and being relatively economical and easy to fabricate. Their light weight has permitted section thicknesses that minimize the stability problem for a wide range of load levels. This is particularly true when aluminums are considered in relation to the steels.

Titaniums—The titanium alloys were developed primarily as a lightweight, high-temperature-capability material compared to the aluminums. One of the major limitations to their extensive use has been their high material and fabrication costs. The combination of density and strength, specific strength and stiffness, and inherent corrosion resistance makes them attractive for use in a wide range of structural configurations.

One material will be selected to form the baseline metal material panel.

3.2.2.2 Advanced Filamentary Composites

Advanced filamentary composites combine strong, stiff fibers with a relatively weak and ductile matrix to produce an increase in specific strength and specific modulus that no foreseeable metallic material can match. Of the many filaments that possess sufficiently high elastic modulus and strength to be classified as advanced composite fibers, boron and graphite are the only materials available in sufficient quantity to be considered for production programs. These fibers are combined with a supporting matrix of epoxy or aluminum to produce a material having exceptional strength and stiffness in the fiber direction.

These highly directional composite materials are produced in the form of thin (0.005- to 0.008-in.) tapes that may be laminated to produce a finished component. The form of the material brings a new design flexibility to the aerospace industry in that laminae may be oriented to match the load requirements of each specific component.

Through a combination of IR&D programs and government contracts, Boeing has developed the capability to design, analyze, and fabricate cost-effective composite structures using these material systems. Due to the high cost of these material systems, cost-effective design has been emphasized in all phases of Boeing's work on advanced-composite materials.

3.2.2.3 Candidate Structures

Sandwich structures are efficient, especially in stability-critical applications, due to their inherent high-stiffness and low-weight characteristics. The least-weight structure is obtained when the best skin material is properly distributed on core having just the right thickness and density to satisfy the stability required for a given design load case. Core cell sizes are also selected to preclude local skin instability from occurring before general instability. Several core cell configurations are available, the most popular being a honeycomb of hexagonal or square cell structure. These core configurations are usually assumed to contribute only shear-carrying ability and not inplane load capability, as in the cases of corrugated, tubular cores, or trussed cores. The shear-only-type cores are the ones of predominant interest to this study. Analysis of various combinations of skin material, skin material geometry, core thicknesses, and core densities will lead to the optimum geometry for each load case.

3.2.2.4 Joining Methods

Boeing has extensive experience with various joining techniques for all structural metals and composite structures. Factors governing the selection of joining techniques include structural performance, inspectability, and cost. Methods of joining to be considered in this program are given below.

Mechanical Fastening—Mechanical fastening represents a category of assembly joining that includes rivets, various threaded fasteners, and patented systems such as Hi-Lok, lockbolts, and Riv-Bolts. Installed costs are based on procurement, hole preparation, and fastener installation.

Adhesive Bonding—Adhesive bonding will be considered as a joining method for metal-to-metal, metal sandwich, and composite construction. Bonding normally requires careful consideration of fit-up and manufacturing sequences during design.

3.2.3 Structural Analysis and Optimization

The phase 1 structural analysis and optimization procedure is shown in the program plan flow of figure 3-1. The sandwich element optimization and the sandwich panel minimum-weight analysis will proceed on a parallel effort. As shown in the figure, these two analysis paths are related through the sandwich data bank. From the analysis viewpoint, these two parallel paths solve different problems. The sandwich element optimization is for uniform panels (elements) under uniform load. The other analysis path is for optimization of nonuniform panels under complex load distribution. The distinction between elements and panels is further characterized as follows:

Sandwich Elements	Sandwich Panels
Uniform-load distribution	Non-uniform-load distributions
Optimum dimensions constant throughout element	Optimum dimensions varying throughout panel
Stress distribution uniform	Stress distribution from computer analysis
General instability from explicit buckling equations	General instability from computer analysis

From these characteristics, it is evident that panels designed for load diffusion cannot be accurately analyzed without computer analysis. In phase 1, the stress distributions, minimum weight, and general instability of the nonuniform panels will be evaluated through computer-assisted analysis. Computer program availability and proposed use are as shown in table 3-1.

A composite sandwich element data bank would be useful for preliminary sizing of composite sandwich structure. A data bank may be readily compiled for a range of loads and aspect ratios. This sandwich element data bank would be directly applicable to uniformly loaded panels. For the two structural loading configurations shown in figures 1-1 and 3-1, the data bank is not directly applicable. Comparisons of the results from the computer analysis with the sandwich element data bank may establish simple preliminary relations for the design of panels to diffuse concentrated loads. Another function of the data bank will be to relate the idealized orthotropic finite elements of the automated computer design programs to specific composite laminate layups while maintaining equivalent strength and stiffness.

3.2.3.1 Sandwich Element Optimization and Data Bank

Optimum honeycomb sandwich elements will be synthesized and compiled in data banks. Minimum-weight honeycomb sandwich elements will be evaluated for a range of loads and aspect ratios. Biaxial loading ratios N_x/N_y, combined with shear loading N_{xy}, will be included. Minimum-weight sandwich elements will be obtained that satisfy the following failure modes: metal yielding, ultimate composite strain, face/honeycomb core instability, orthotropic panel interactive general instability, and Johnson-Euler instability. Except for Johnson-Euler instability, all buckling analyses will be of the elastic type. The analytical methods will be rigorous to the extent that all important potential failure modes will be analyzed with methods found in structural analysis literature and advanced methods used in previous NASA contracts. As part of the data bank presentation, the analytical methods and equations will be concisely documented. Minimum-weight sandwich elements will be displayed in a data bank format similar to that shown in figure 3-3 and reference 2. Key design parameters in the data bank will include:

a) Load levels and load ratios N_{xy}/N_x and N_y/N_x

b) Weight of element

c) Optimum laminate layup and detail dimensions

d) Margins of safety for each failure mode

e) Nominal sandwich element stiffnesses

f) Element strains at ultimate design load.

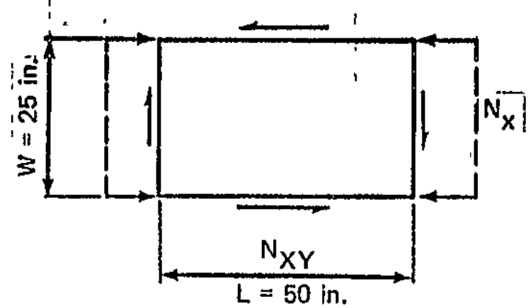
The data will be plotted in terms of weight versus loading to provide a basis for visual assessment of load/weight interaction between competitive designs as represented in figure 3-4. In this figure, material systems are compared. Other significant parameters would

Table 3-1.—Proposed Usage of Structural Analysis and Optimization Programs

Computer program	Structural element optimization	Structural configuration optimization	Stress analysis	Stability: Uniform load	Stability: Nonuniform load	DESCRIPTION
BUCLASP-2				X		Built-up composite structure
COOP	X			X		Laminate panel optimization
OPTRAN	X			X		Flexible optimization code with user-defined design constraints
PANBUC II				X		Orthotropic panels
SAMECS RESIZE		X	X			Finite element stress analysis coupled with element optimization program
STAGS					X	
TES 222		X	X			General optimization program
TMI	X			X		Laminate orientation optimization

Concept: Graphite-skinned sandwich

Design conditions:

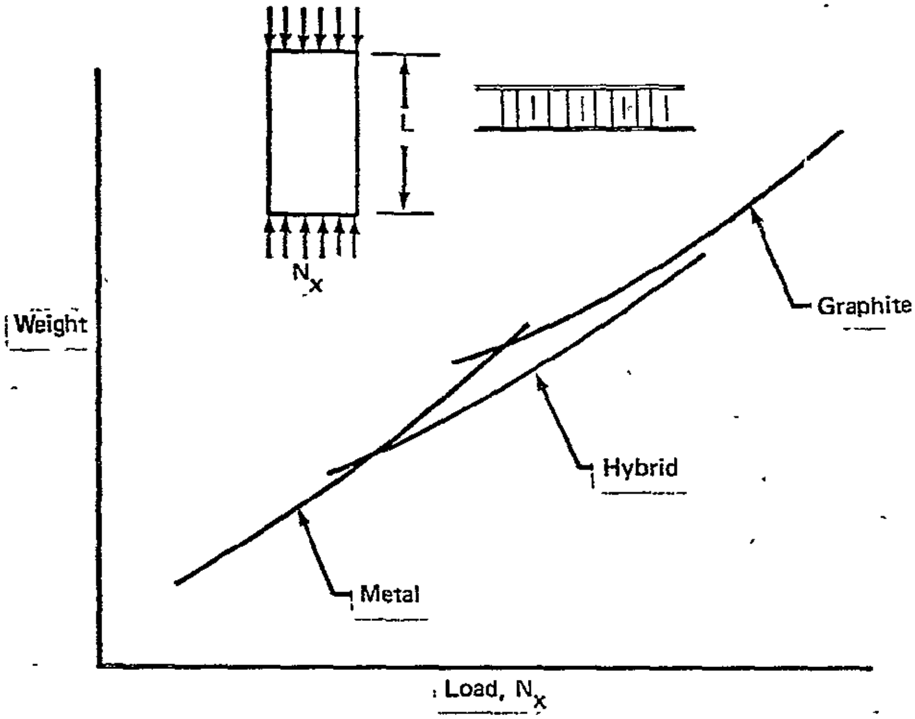


N_x = variable
N_y = 0
N_{xy} = 200 lb/in.
p = 0
Temp = 72° F
Material. 7075-T6 aluminum
Face sheet layup: 0°/±45°/90°

Material : 7075-T6 Aluminum

Case	Design ultimate load, N_x	Weights	Detail Dimensions	Margins of safety	Element stiffnesses	Element strains	Critical element strains	Effective structural parameters

Figure 3-3.—Element Design Data Bank



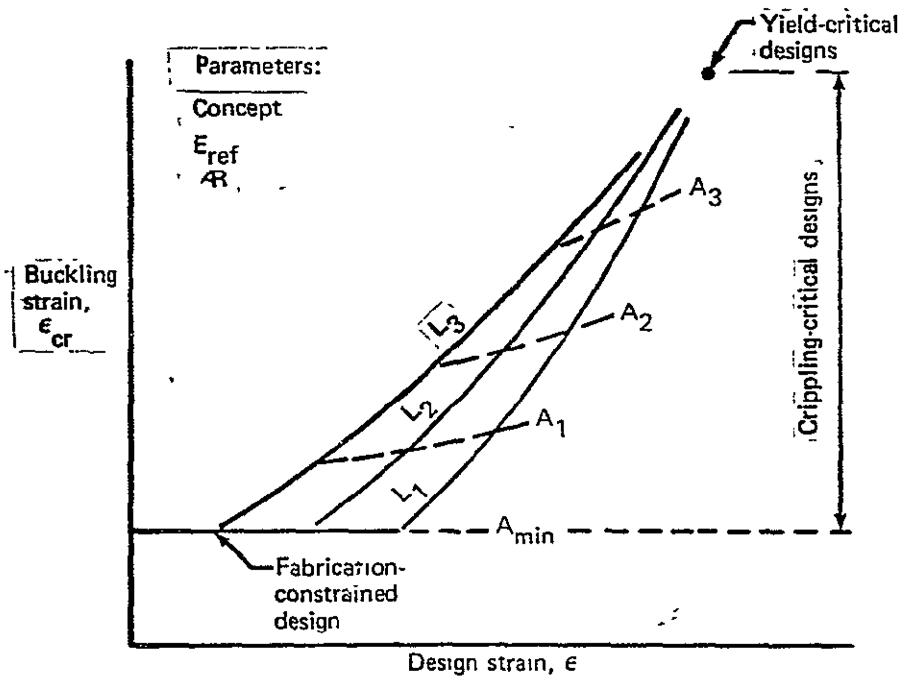


Figure 3-4.—Data Bank Plots

include items such as element aspect ratio and biaxial load ratios. Plots of critical element strain versus design parameters will be displayed as also shown in figure 3-4.

The honeycomb element data banks will be generated using the COOP or OPTRAN computer programs. A description of these program capabilities is given in section 5.2. The COOP program is specifically designed to obtain minimum-weight composite honeycomb panels. The OPTRAN optimization program is more general and may be used for numerous structural forms.

3.2.3.2 Sandwich Panel Minimum-Weight Analysis

Minimum-weight sandwich panel designs will be evaluated for a significant number of load levels, structural materials, and panel sizes. The sandwich panels will be optimized for two concentrated loading configurations of reaction by uniform compression and uniform shear. For each loading configuration, a minimum of four load levels will be evaluated. Three material constructions—metal, graphite composite, and a hybrid composite—will be studied for all load levels. For each loading configuration and a selected load level, the influence of various panel aspect ratios will be evaluated. Panel stability will be evaluated for both loading configurations, at least three aspect ratios, and one material system.

The automated structural design program, TES-222 (ref. 3), will be used as the principal computer analysis and optimization program for this study. This program is a powerful engineering tool for designing structures of minimum weight that perform satisfactorily under a variety of design conditions. It is a combination of an extremely high speed finite-element stress analysis computer program and a sophisticated, directed-search algorithm for obtaining a minimum-weight design. In essence, the program will determine the minimum weight of an idealized structure that is compatible with user-specified geometric size limitations, allowable stresses, allowable displacements, and allowable compressive strains (or buckling strains). Buckling strain levels, from the optimized sandwich element data bank, may be input to the automated structural design program as an allowable strain constraint. For this analysis, the panel will be treated as a nonuniform plate. In the case of orthotropic composite panels, coincident special finite elements (orthoplates) will be considered to represent the composite laminate stiffness. The sandwich element data bank will provide a stiffness and strength correlation between specific laminate layups and the stiffness of the orthoplates associated with the automated-design minimum-weight panel. Local and general stability checks will be conducted after the automated-design procedure, and the allowables will be modified as necessary if the panel is unstable.

3.2.3.3 General Instability Analyses

The optimum structural assemblies generated by the TES-222 code will be analyzed for critical general instability (bifurcation) buckling loads. General instability buckling analyses are necessary because the TES-222 code does not have provisions for general instability buckling constraints (such constraints would require prohibitive execution time). The STAGS code (ref. 4) will be used to conduct the general instability buckling analyses of the panel assemblies. This code is easy to use and is relatively fast for a general-purpose-type code.

The STAGS code models of the various structural assemblies will simulate the following structural features as required:

a) Isotropic plates

b) Orthotropic plates

c) Variable plate properties (A_{ij} and D_{ij})

d) Discrete orthogonal stiffening

e) Discrete skewed stiffening

f) Variable stiffener properties (EI_y, EI_z, GJ, EA)

The treatment of variable structural properties will be accomplished with user-supplied subroutines, as was done in the NASA-Langley composite shear web contract (ref. 5).

The buckling eigenvalue analysis will be based on the assumed conditions used in the structural assembly optimizations. The concentrated load point will be treated as a reaction point. This approach will preclude mixed boundary condition problems. The theoretical buckling loads will be factored with selected, consistent "knock-down" factors to establish allowable buckling loads.

A plot will be generated for each case showing critical buckle mode shape; these plots will support qualitative study of buckle mode versus structural form The resulting mode shape characteristics will be useful in specifying effective panel lengths for the TES-222 code or a manual assembly optimization using the sandwich element design data bank.

3.2.4 Merit Functions

The basis for comparing or ranking structural concepts is generally established in the design criteria. These criteria define a measure of value and accordingly provide the ability to make a decision between any two candidate design configurations. Often the expression of criteria is self-contradictory, for example, the desire to have both minimum weight and minimum cost structure can be a contradiction. The process of compromising between two or more contradictory requirements is referred to as tradeoff. Generally, the principal part of the criteria or an appropriate tradeoff can be expressed analytically in terms of a configuration-dependent function, where the resulting configuration, in an extreme of this function, is optimum. This function, when it exists, is termed a merit function. In structural applications, the merit function is normally weight, cost, or a cost/weight tradeoff.

3.2.4.1 Weight Merit Functions

In general, structures may be compared on a strength/weight basis or a weight/strength basis. For structures sensitive to buckling failure criteria, weight/strength and strength/weight are not reciprocals of each other because of the strong influence of structural proportions. Accordingly, structural loading indexes are oriented toward a weight/strength comparison and will lead to predicting structural weight required to achieve a given strength rather than the strength associated with a given weight. Various structures may be compared for structural efficiency by plots of their structural loading index versus their relative weight.

A common structural loading index for compression panels is P/b^2, where b is the panel width. This index is derived from the classical plate buckling equation and reduces to the form $P/b^2 \propto f^{3/2}/(EK)^{1/2}$. For builtup structures, it is usually convenient to replace the buckling stress, f, by $P/\bar{t}$, since the smeared thickness or $\bar{t}$ is directly relatable to structural weight for a given material density.

Insights to the relative merits of various structural forms and proportioning have been offered by some investigators in the form $N_x/bE = \xi\,(\bar{t}/b)^m$. The efficiency factor, ξ, is a function of structural geometric proportioning and can be calculated. The highest achievable ξ will allow the least $\bar{t}$ and, therefore, least panel weight for a given structural index of N_x/bE.

To achieve maximum structural efficiency, precise proportioning of the geometry is prescribed by pertinent mathematical expressions. For example, the core height is a function of operating stress. Some geometries may or may not be realistic for a given structure due to space envelope constraints. Consequently, compromises may be necessary that lead to a reduced efficiency and an associated but identifiable weight penalty.

3.2.4.2 Cost Merit Function

The property of the minimum-weight configuration that results in high cost has to do with the exotic nature of lightweight concepts. The initial cost input is generated from the application of high-strength, low-density, advanced material systems which, because of their specialized properties and limited applications, impose high material costs. The principal part of the cost merit function, however, results from tight tolerance machining and fabrication processes designed to minimize weight. Machining and fabrication entail equipment, tooling, and labor costs and must, in general, be evaluated independently for each specific candidate structural concept. In some cases, empirical expressions can be derived to express costs as a function of geometric variables that critically affect cost. Figure 3-5 illustrates the machining cost per square foot of an integrally stiffened plate as a function of stiffener spacing and depth.

In the absence of an empirical expression for the cost merit function, the cost for a discrete number of candidates may be determined and a tabular merit function of cost, in terms of geometric parameters, established.

3.2.4.3 Cost/Weight Merit Functions

All transportation vehicles are unique in that their performance can be measured in terms of cost to perform a transportation function. This is true for trains (dollars per ton-mile) and spacecraft (dollars per pound in orbit). Since the value of vehicle functional performance can be measured and weight (including structural weight) is a key to the vehicle's cost-effective performance, a weight-value relationship can be established. A typical

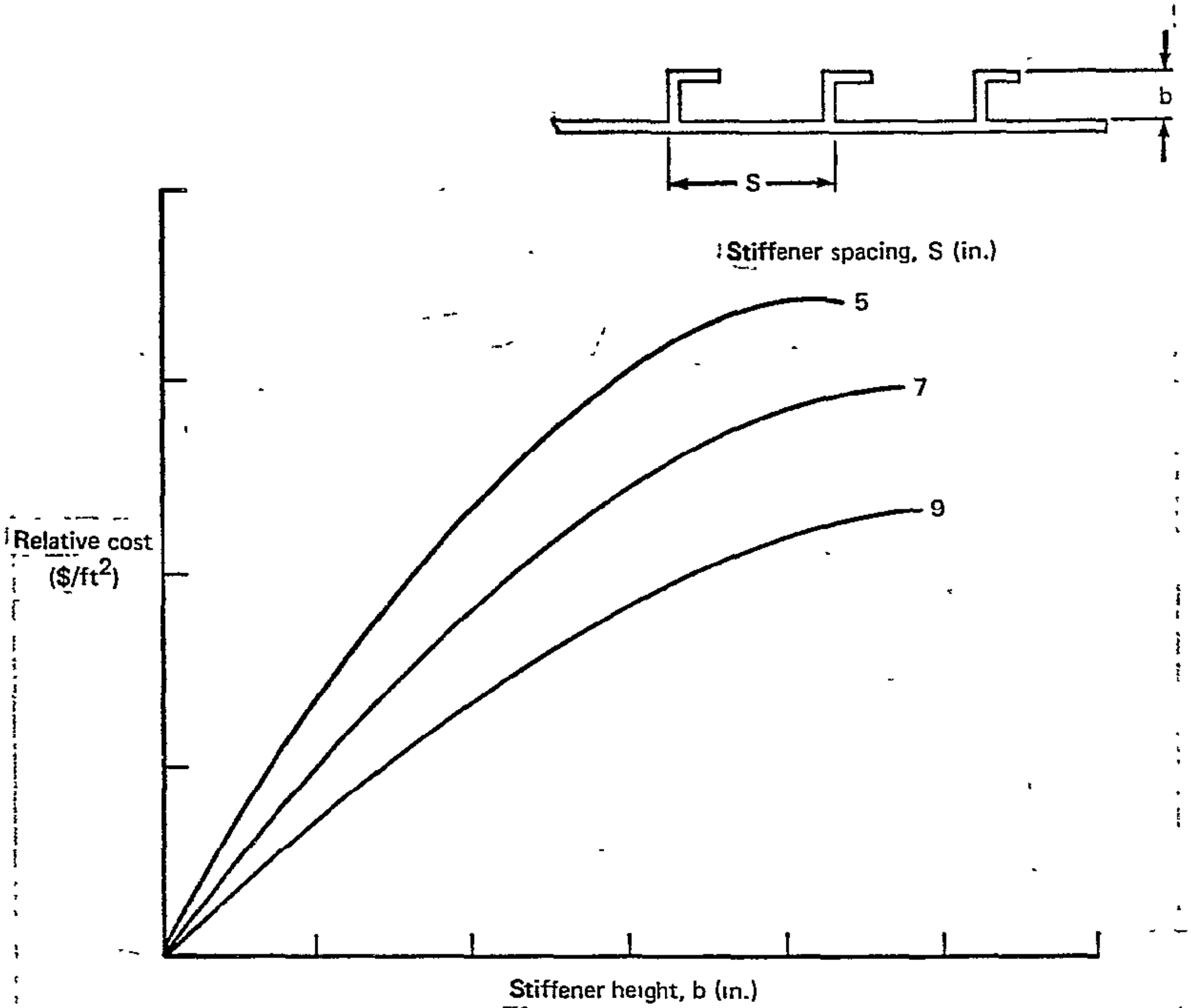


Figure 3-5.—Cost/Merit Function of Integrally Machined Plate

value assessment parameter relating weight, vehicle performance, and vehicle cost is the dollars that can be spent for weight savings to cost effectively improve the vehicle's functional transportation cost.

For most transportation systems the value parameter, or worth in dollars, of a pound of weight saved can be established by evaluating the change in vehicle system cost, including out-the-door and O&M costs as the design weight is changed. The value parameter is the ratio of the change in cost to the change in weight or the maximum permissible slope of the cost/weight curve to reach the breakeven point (fig. 3-6). To exceed this slope is not economically justifiable, since spending more dollars per pound would result in net higher cost to the vehicle system. Spending at a rate less than this maximum breakeven slope results in a net dollar saving to the overall program or system Once the results of saving weight have been determined for a particular vehicle, the foundation has been prepared for the cost-effectiveness analysis of the vehicle's structural concepts.

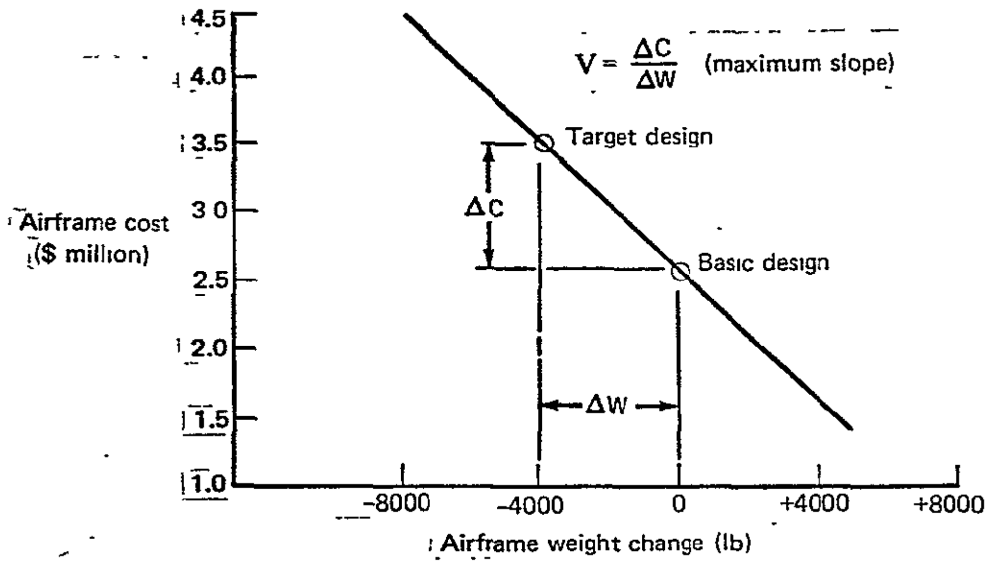


Figure 3-6.—Value Parameter Represented Graphically

Since the study of any vehicle's total system cost is a significant effort in itself, it is suggested that the cost/weight trades established be evaluated over a large range of value parameters (dollars per pound of weight saved). This range of values will allow the study data to be used for spacecraft, military, and commercial aircraft. As noted in section 3.2.1, this can be easily done by use of a log plot (see fig. 3-7).

The weight/cost effectiveness merit functions selected are as follows:

$$M_{cw} = W_i + C_i/V_j$$

$=$ dollars per pound, cost/weight-merit function

W_i = pounds = weight of concept studies (ith concept weight)

C_i = dollars = cost of concept studies (ith concept cost)

V_j = dollars per pound = value parameter (jth vehicle value parameter)

The cost (C_i) analysis of the structural concepts to be studied will be of the preliminary design or conceptual study depth and procedure. The key is to keep all concept cost estimates as close as possible to the same depth. The tendency is to provide more depth to the more familiar, which leads to a conservative factor for the less familiar structure cost estimates. One way to alleviate this is to produce equal depth preliminary design layouts for cost estimating. By keeping the layout at the preliminary design level and by using equal depth, simple manufacturing planning for each concept, a reasonably comparable set of cost estimates can be obtained. Also, by closely monitoring and reviewing the cost ratioing and

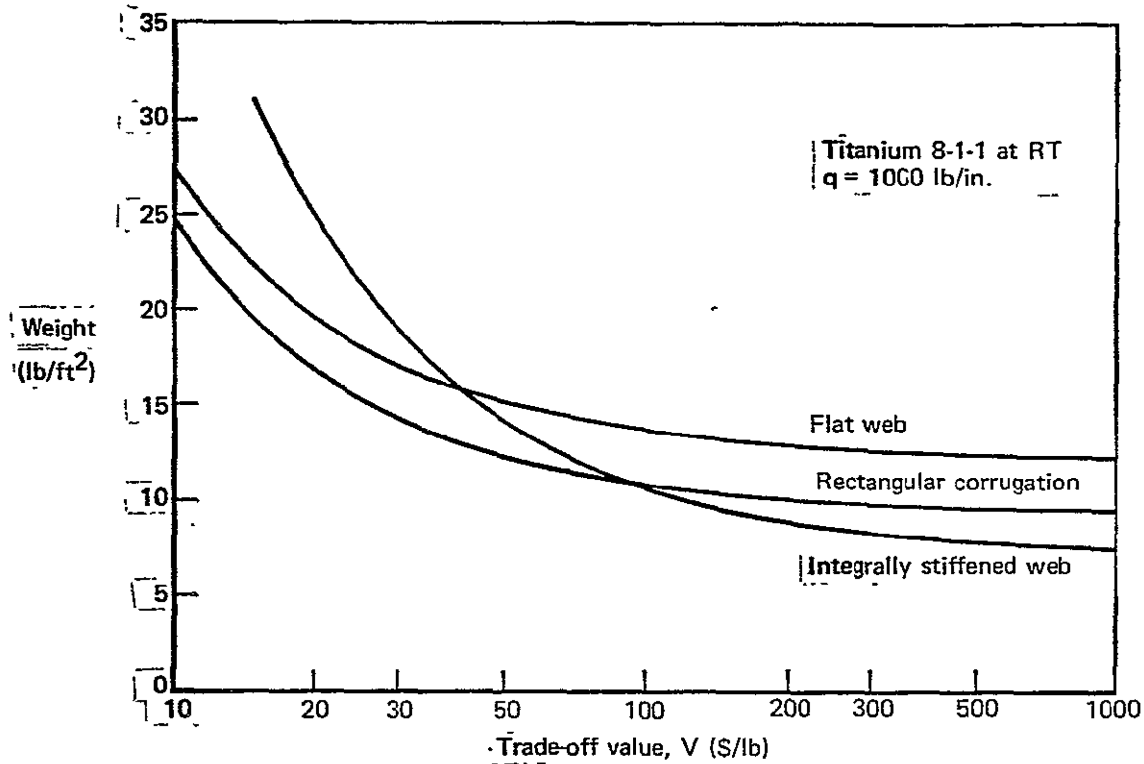


Figure 3-7. —Shear Web Designs—Weight/Cost Effectiveness

cost complexity factor recommended by manufacturing, consistency can be controlled. The estimate will be based on the cost criteria established early in the program. (See sec. 3.2.1 on criteria.)

The one factor affecting all approaches to establishing cost in terms of dollars is the continuing change in the dollar value and the variation of rates from one producer to another. Therefore, this study will attempt to keep costs in terms of equivalent man-hours or a similar parameter. Each user of the data generated from this study can use it with the most current dollar value and his own man-hour rate. Since both recurring and nonrecurring costs contain both man-hours and material dollars, the material dollars portion of each will have to convert to the equivalent man-hour parameter.

The following are some examples of how the suggested merit function will be applied to make cost/weight trades during the study. The first example shows how it can be applied over a range of the value parameter and be used to pick one concept from a group of concepts. The second example shows how it can be used to select detailed structural arrangement for the optimum cost/weight relationship.

Example 1 (Spars)—Several design possibilities for shear webs with varying weight and cost figures can be examined to determine an optimum cost/weight concept. For this example, a design load level of q = 1000 lb/in. and titanium 8-1-1 material were assumed.

Three types of shear webs were considered: flat web, integrally stiffened web, and rectangular corrugations. The weight and cost data associated with each concept are given in table 3-2.

Table 3-2.—Shear Web Designs—Titanium 8-1-1 at RT, q = 1000 Lb/In.

Design Concept	Web thickness (in.)	Weight[a] (lb/ft^2)	Cost,[a] 200th unit (\$/ft^2)
Flat web	0.452	12.33	151.42
Integrally stiffened web	0.263	7.23	359.43
Rectangular corrugation web	0.262	9.46	153.35

[a] Weight and cost data include spar caps and attachment to upper and lower wing surfaces—panels 20 by 100 in.

The least-weight concept is the integrally stiffened web construction, but it is also the most expensive of the three designs. The least-cost design (flat web) is heaviest, and the rectangular corrugation concept is competitive from a cost standpoint. Applying the weight/cost effectiveness yardstick to this problem results in the data shown in figure 3-7.

Example 2 (Compression Cover)—The compression cover of a typical high-aspect-ratio wing box can be analyzed for cost/weight effectiveness using a combined optimization and producibility approach. The wing box upper surface panel example will be designed for an ultimate axial load level of 8000 lb/in. A Z-stiffened cover has been selected for the load-carrying requirement, with a fixed rib support spacing of 20 in. An efficiency equation of the form:

$$\frac{N_x}{L\bar{n}E} = \xi \frac{\bar{t}}{L}^2$$

is used to determine the effective $\bar{t}$ and associated design dimensions of the cross section. By systematically varying the efficiency factor (ξ) from maximum efficiency to other off-optimum values, a variation in cover panel weight and spacing of the Z-stringers results. This variation in weight and stringer spacing is essential in a cost/weight analysis. Minimizing the number of pieces and lines of attachment is a firm guideline for cost reductions. By increasing the spacing of the Z-stringers, the total number of details and lines of attachment are reduced, thereby reducing the cost of manufacturing and assembly. The pertinent dimensional, weight, and cost data resulting from this analysis are shown in table 3-3.

Note that increased stringer spacing drives costs downward, with small variations in the weight penalty. A tradeoff possibility is suggested by the increasing weight and decreasing costs shown in table 3-3. Since weight and cost cannot be added directly due to dissimilar dimensions, a third parameter must be introduced—the value of saving weight. Dividing the cost of each design by the value parameter (assumed at \$50 per pound for this example)

Table 3-3.—Z-Stiffener, Multirib, Aluminum 7075-T6 at RT

Configuration	Stringer[a] spacing	Weight (lb/ft^2)	Cost (\$/ft^2)
1	1.538	5.23	87.51
2	1.764	5.25	83.79
3	2.049	5 28	78.90
4	2.351	5.33	75.24
5	2.770	5.58	72.87
6	3.217	5.97	71 21

[a] Note the effect of increased stringer spacing driving costs downward with small variations.

allows computation of the merit function, M_{CW}. By performing this summation at discrete values of the variable b_s, a "bucket" curve is produced that indicates the optimum stringer spacing for the particular vehicle under study (see fig. 3-8). Note that the least-weight design is not the best choice for structural cost effectiveness in this example. As in example 1, a graph of the merit function versus value could also have been constructed to illustrate the optimum design choice for any assigned worth of saving weight.

The depth and utility of this cost approach as well as others will be reviewed. The approach suggested here offers a simple preliminary design approach to cover the usually complex cost/weight trade.

3.2.5 Practical Considerations

The selection and optimization of a structural concept involves the evaluation of practical design features. Qualitative factors such as design simplicity, durability, manufacturability, and serviceability must be considered in the final design. The feasibility of constructing actual hardware from the optimized structural elements may lead to conclusions that invalidate analysis results for certain ranges of the structural loading parameter. For example, structures optimized primarily for compression loading conditions may require considerable additional material to meet torsional rigidity, shear strength, and other secondary loading conditions.

Included in practical considerations are the nonoptimizable weight items that normally render a design useful. These items must be accounted for in the final weight predictions of optimized structural concepts. Certain parameters such as minimum-gage, pad-ups, adhesive and core weight, and standard skin gages may be handled relatively easily as a geometric constraint in the computerized optimization tasks. The majority, however, cannot be expressed as design variables and must be added to the theoretical weight of the optimized structural design. Items to be considered in this regard include primary joint details and mechanical fasteners. Items such as cutouts and attachment fittings would normally occur in a production program but are not considered to be items important to this exploratory program. Examples of practical design considerations for sandwich panels are shown in figure 3-9.

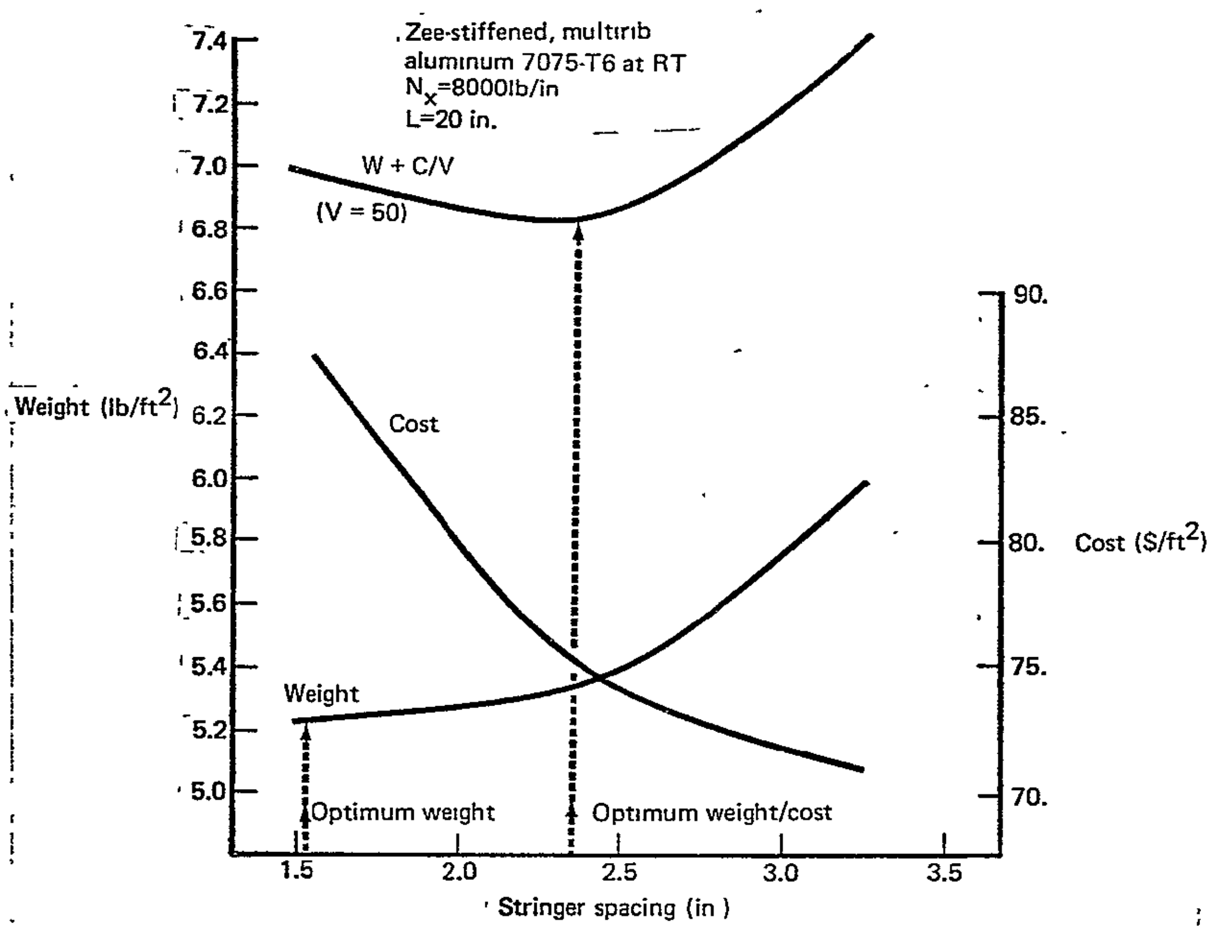


Figure 3-8. —Optimum Weight/Cost Stringer Spacing

Repairability and inspectability can have a significant impact on spacecraft structure, particularly when refurbishment is an operational requirement. Mechanically fastened, open-section structural designs are easily repaired and accessible to visual inspection techniques. Adhesive-bonded structures pose more difficult repair problems and require more sophisticated nondestructive inspection techniques. These practical considerations will be given to the respective structural forms. Possible design constraints due to repairability and inspectability will be identified if they could affect optimum weight and cost.

3.2.6 Data Documentation

Coordinated documentation will be prepared to present the form data banks, results of the configuration optimization and stability analyses, design detail studies, and performance/cost index studies.

The data presentation will be in a format such that a designer having the applicable design conditions may readily determine a low-cost, least-weight or near-least-weight preliminary structural design satisfying the design conditions. The data presentation is expected to produce a basis for assembly material selection, structural form selection, and

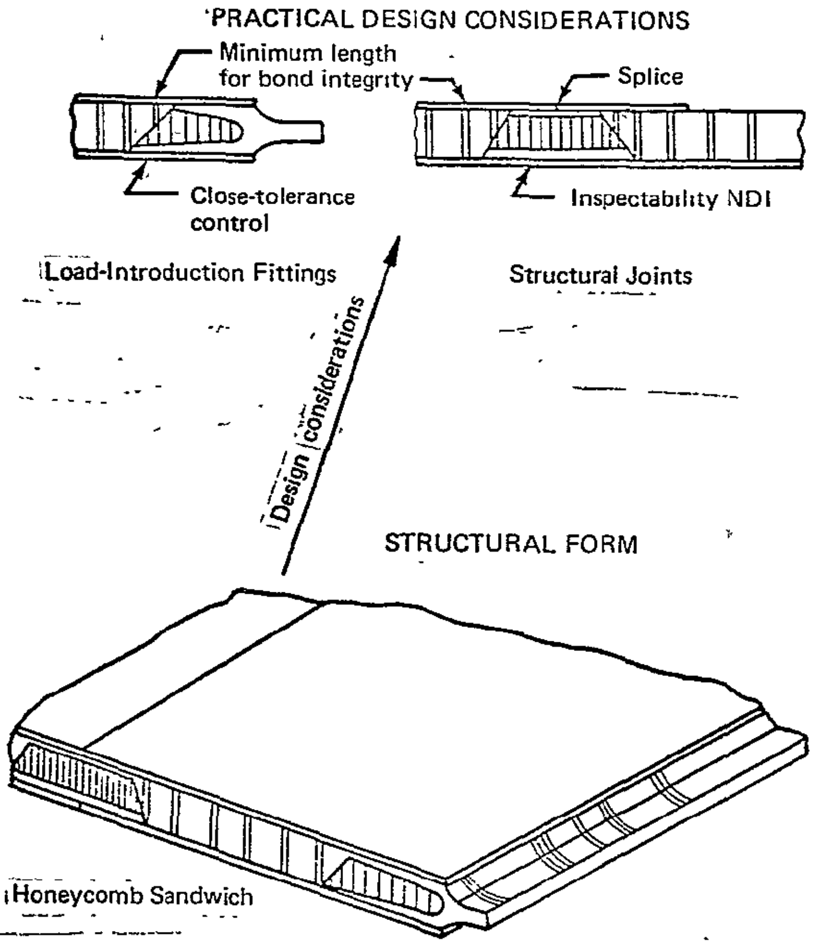


Figure 3-9 —Sandwich Forms and Practical Design Considerations

structural element geometry. The presentation will allow determination of penalties associated with nonoptimum selections made by cost or other practical considerations In all cases, the data assembly will strive to employ parameters that tend to reduce the number of variables, thereby increasing data generality.

Both tabular and graphic data will be presented. Graphic presentations that clearly demonstrate trends are preferred to those sufficiently complex to obscure the physical significance of the data accumulation. As discussed in section 3.2.3, the form design data banks will contain a wealth of detailed design data that will be useful in future preliminary design studies of point-loaded structure. The data bank information will also contain structural analysis methods, equations, and references.

3.2.7 Space Tug Applications

The current orbit-to-orbit systems schemes are comprised basically of two tankages in tandem surrounded by insulation and an outer shell, a forward skirt area for payload docking/retrieval attachment, the intertank shell, the aft skirt area for adapter interfacing, the thrust structure, and engine installation. Three industry concepts are depicted in figure 1-1; all have similarities.

The areas of particular interest for application of the generated methods are the forward skirt, the aft skirt, and the thrust structure. The forward and aft skirts are particularly suited to analysis since they are basically structure that distributes concentrated loads into tank shell structure. The proposed outer structural shells could be especially suited to a minimum-weight structural design method. Typical load ranges will be studied to encompass the space tug structural loading.

3.2.8 Fabrication of Test Panels

Two composite honeycomb sandwich panels (12 by 12 in.) will be fabricated to verify the analysis and design techniques developed in the phase 1 study. Conceptual sketches of two test panels are shown in figures 3-10 and 3-11. The test panels will be constructed with a load-introduction member and edge support suitable for mechanical loading. These two test panels are representative structure for diffusing a concentrated load to a uniform-compression or a uniform-shear load reaction. Test-quality panels will be established using previously developed process specifications and nondestructive testing techniques. As an additional assurance of design integrity, small control specimens will be fabricated and tested to verify the laminate strength and bonding quality. Flexure, short-beam, and flatwise tension specimen tests are sufficient to establish the panel quality.

3.3 PHASE 2—FEASIBILITY DEMONSTRATION PANELS

In the second phase of the proposed program, the design techniques developed in phase 1 will be applied to a specific structural component of the space tug vehicle. The current version of the space tug structural configurations will be reviewed and a specific component selected for redesign. The structural form will be selected by applying the weight/strength and cost/weight design parameters developed in phase 1. Redesign will be governed by strength/weight/cost merit functions and aided by the element data banks established for the structural form of interest. The feasibility of the design procedures will be demonstrated by fabricating a panel representative of the redesigned space tug structure. Producibility and cost evaluation studies will be conducted concurrently with fabrication. This phase of the program will be of 6 months' duration.

3.3.1 Selection of a Space Tug Structural Component

The current space tug configuration consists basically of two tankages, arranged in tandem and connected by an intertank structure or shell. This structure is surrounded by thermal insulation, meteoroid shield, and/or an outer structural shell. Payload docking and retrieval system loads are transmitted by a forward structural skirt of stiffened shell construction. A similar aft skirt—and in one configuration, the intertank shell structure—is used to transmit the space shuttle adapter interfacing loads. The engine installation and thrust loads are reacted by a combination of space truss and shell structure.

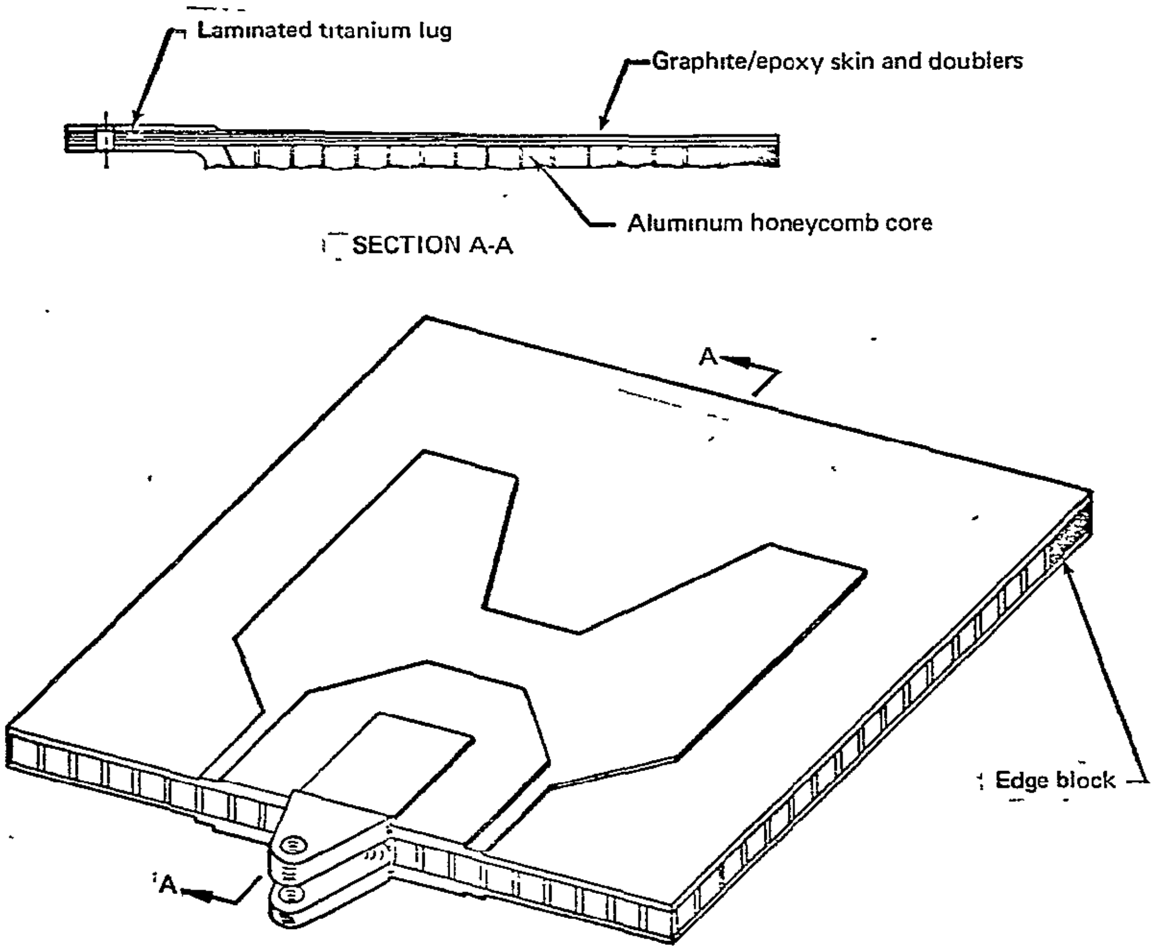


Figure 3-10.—Graphite/Epoxy Honeycomb Sandwich Compression Panel

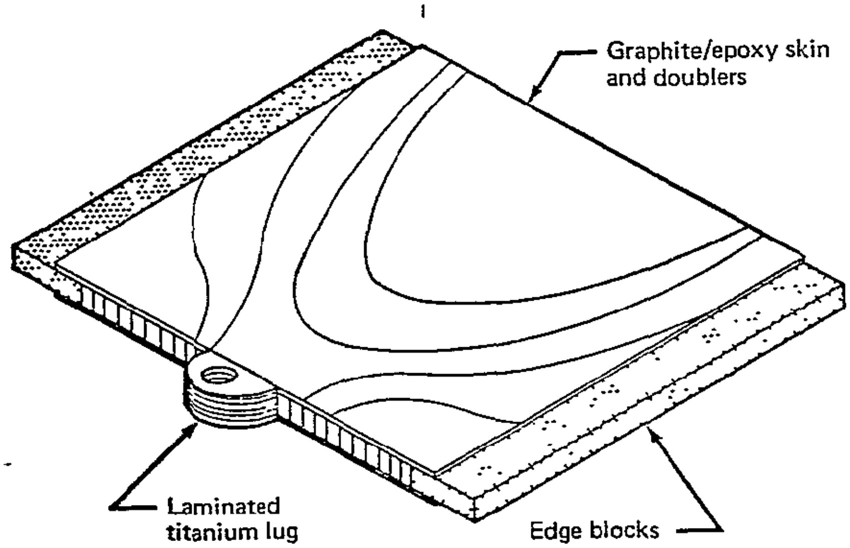


Figure 3-11.—Graphite/Epoxy Honeycomb Sandwich Shear Panel

A structural component for the phase 2 study will be selected from one of the following areas:

a) Forward and aft structural skirt

b) Intertank structural shell

c) Engine thrust structure

The component selected would be represented by a lightweight panel design to react the applied concentrated loads by either uniform compression or uniform shear. Further application of phase 1 developed design techniques to the selected sandwich panel component form for the applicable load environment will determine the desirable material and structural arrangement.

Approximately 40% of the weight of a typical space tug configuration is contained in the primary load-carrying structure. Of this portion of structural weight, approximately half is attributed to the forward skirts, intertank, and aft skirt structure. A significant payoff in weight reduction may be realized from the application of advanced lightweight structure technology to these areas of the space tug.

3.3.2 Analysis and Design of the Selected Structural Form for Space Tug Application

A unique analysis and design technique, aided by the study results and methods developed in phase 1, will be used for the design of the demonstration panels. From application of the weight/strength and cost/weight parameters, the selected structural form will be known to be optimum for the loading environment of interest. It is further known that if general instability considerations are ignored, a panel designed to uniformly achieve a predetermined maximum allowable stress will approach minimum weight for a given load condition when represented by an equivalent variable-thickness plate. This equivalent plate will satisfy the minimum-weight criterion by providing minimum area. Analysis and design will proceed in the following manner:

a) The selected sandwich panel form will be idealized as an equivalent plate.

b) The stress analysis mode of computer program TES-222 will be used to conduct a stress analysis of the plate for the load environment of interest.

c) Stress levels and minimum area requirements will be established for the plate elements. Superimposing these data on conceptual sketches will give added visibility to the design.

d) General instability checks will be conducted using STAGS. Modifications will be incorporated to satisfy general and local instability requirements.

e) The appropriate optimized sandwich element will be used to match the stress, area, and stiffness requirements for individual areas of the panel with the optimum structural arrangement and geometric proportions.

f) The structural assembly will be reanalyzed subject to buckling constraints. The initial instability check will be made with respect to the primary load axis; secondary analysis will be for the minor load axis and shear. Successive searches will be conducted as necessary to converge on the minimum weight assembly.

g) Panel producibility will be monitored continually during design convergence to ensure a realistic and practical design.

h) The evolving design will be monitored continually to achieve optimum cost/weight relation in the final design.

This analysis and design plan will require the minimum of automated analysis methods and will converge rapidly to a near-optimum design. Incorporated in the resulting design will be practical features and cost constraints, producing a structural concept directly applicable to a preliminary design and/or conceptual study.

3.3.3 Fabrication of Feasibility Demonstration Panel

Feasibility of the proposed advanced design/analysis methods for concentrated load diffusion will be demonstrated through the fabrication of two lightweight test panels representative of space tug structural applications. Conceptual design drawings, sufficiently detailed to permit fabrication of the panel, will be produced. Fabrication of the panel will be controlled by the Boeing Manufacturing Research and Development organization. During design evaluation, the coordinated efforts of production planning, tooling, and fabrication organizations will ensure a realistic demonstration panel design.

Conceptual sketches of two honeycomb panel configurations are shown in figures 3-10 and 3-11. It is anticipated that the structural configurations chosen for fabrication will incorporate advanced filamentary composite to utilize the high specific strength and stiffness offered by these materials. Cost consideration and processability would favor the graphite/epoxy materials for this application. The panels are expected to be fabricated as a one-part item without the necessity of expensive, complex tooling.

3.3.4 Producibility and Cost Evaluation

Producibility and cost evaluation studies will be conducted concurrently with the design, fabrication, and assembly of the feasibility panels. True production costs cannot be determined from the manufacture of one item, however, the fabrication processes required may be assessed and cost projections for production application determined with reasonable accuracy. These cost projections will be compared with the cost/weight/strength trade study data generated in phase 1 of the program and correlation of results will be documented.

4.0 PROGRAM ORGANIZATION AND PERSONNEL QUALIFICATIONS

The relationship of the proposed program to the corporate structure of The Boeing Company is presented in figure 4-1. The program functional organization is defined in figure 4-2. Resumes of the technical personnel who will contribute to the proposed program are provided in the following pages. Emphasis has been placed on selecting key personnel who are especially well qualified in structural analysis, cost estimating and evaluation, and advanced composite technology.

Mr. J. E. McCarty, the program manager, will have overall responsibility and authority for all aspects of the program. This will involve planning, directing, and controlling the program, as well as ensuring the completeness and accuracy of the program results. He will have authority to draw upon all company organizations for program support.

The program technical leader, Mr. G. W. Belleman, will be the principal technical investigator and will be responsible for all program documentation and reporting.

J. E. McCARTY—Program Manager

BS, aeronautical engineering, University of Kansas, 1949

Mr. McCarty is currently technical manager of the Advanced Structural Concepts group (structural bonding development and composites) and the Durability group. In this capacity he is familiar with the engineering aspects of applying advanced technology to aircraft structures. He was also program manager on the recently completed Air Force ADP on advanced cargo/tanker structures, contract AF33615-72-C-1893.

In a previous assignment, he headed Boeing's structural bonding program to develop adhesive bonding for primary structural application. In this capacity he was responsible for all areas of bonding technology including materials research, allowables, hardware development and test, and the development of compatible repair techniques.

Mr. McCarty's background includes both production and research experience in structural analysis and design. He was employed for 7 years by Douglas Aircraft Company in their commercial aircraft division and participated in the design and analysis of the DC-6B, DC-7, and DC-8 aircraft. He has 13 years' experience with Boeing, which includes 4 years in the Aerospace Division on the Dyna-Soar program, and 1-1/2 years with the Vertol Division as supervisor of the Stress Research group.

For the past 9 years, Mr. McCarty has been a member of the Commercial Airplane Group. His experience in this division includes stress analysis support for the 707 and 737 airplanes, as well as several preliminary design programs He has extensive experience in the development and application of finite-element methods to structural analysis problems.

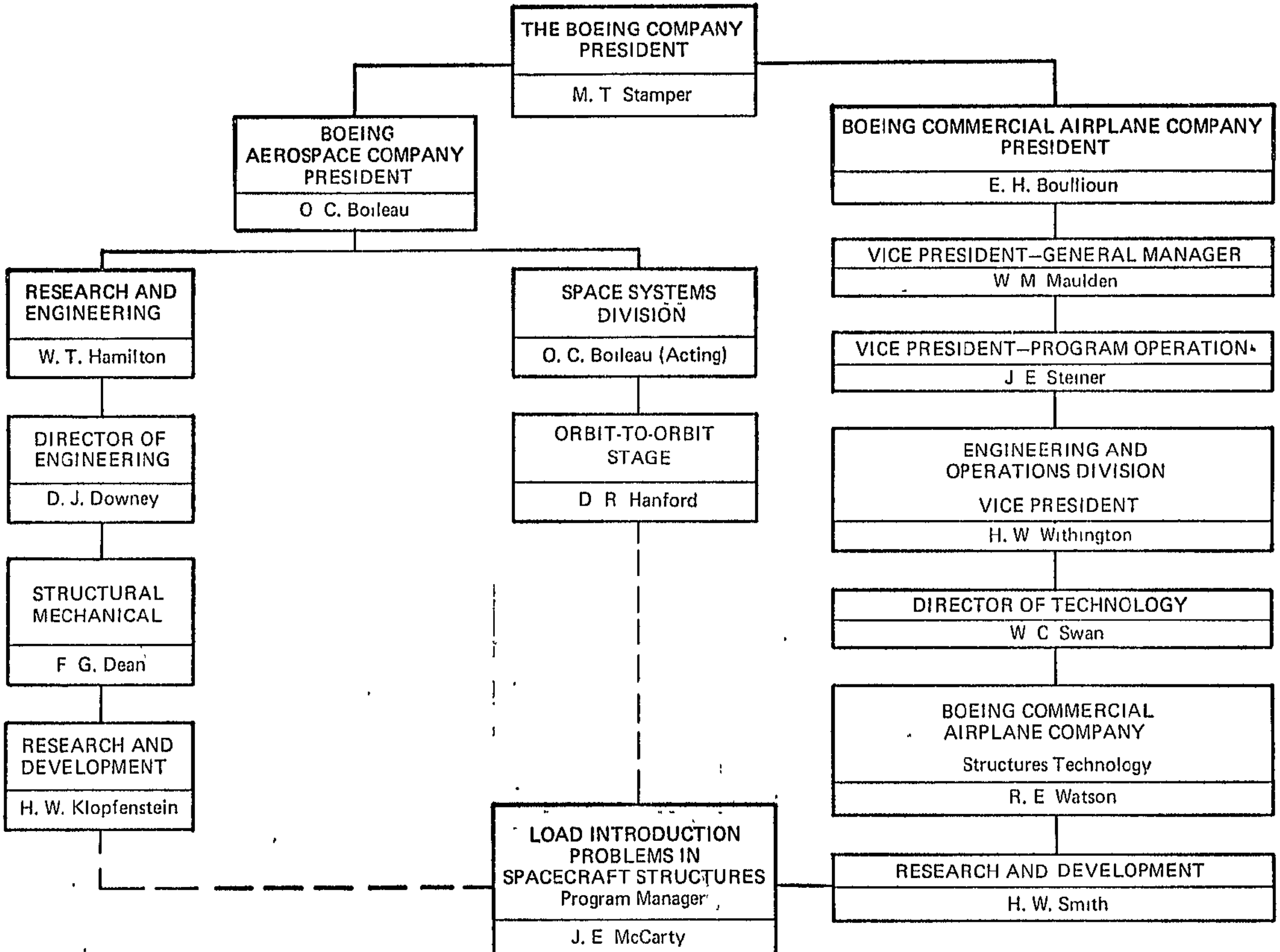


Figure 4-1 —The Boeing Company Organization

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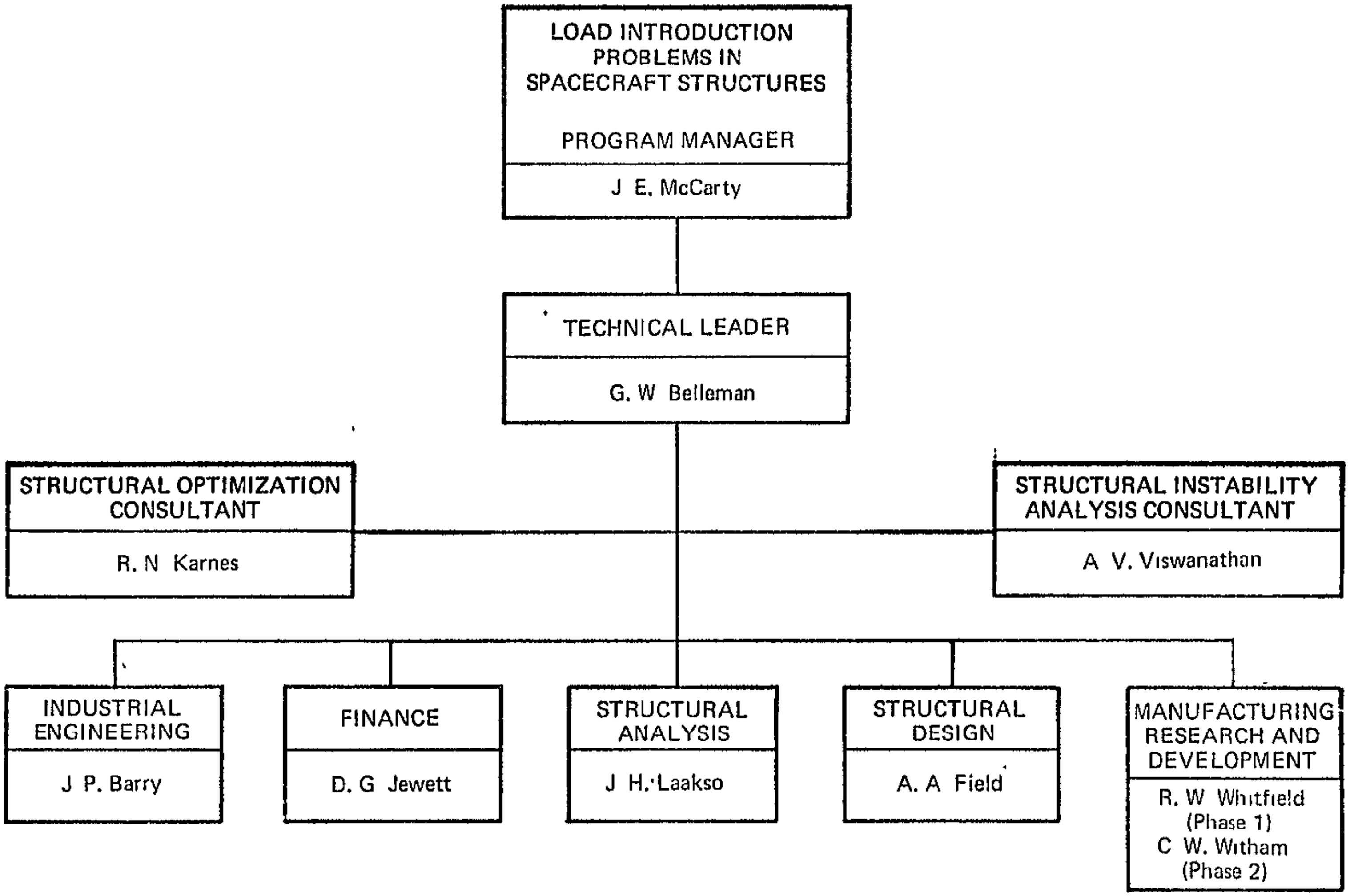

Figure 4-2.—Program Functional Organization

G. W. BELLEMAN–Technical Leader

BA, mathematics, University of Washington, 1956

Mr. Belleman has had over 15 years' engineering experience in aerospace and aircraft structures. He spent 2 years as a stress analyst and fabrication monitor on a research program requiring fabrication and test of a bonded aircraft fuselage section. He has served 3 years as lead stress engineer and lead design engineer for handling equipment required by Minuteman intercontinental ballistic missile launches at the Air Force Test Range. He has 5 years' experience in materials allowables generation requiring statistical analysis of elements and component tests. He has experience in material thermal properties generation on metallics and nonmetallics. He has 5 years' experience in materials and processing engineering technology. In addition, he has served for several years as Boeing representative to the Military Handbook 23 Industry Advisory Committee. His most recent assignments have been as technical leader on NASA contract NAS1-11767, "Design and Fabrication of an Aeroelastic Flap Element for STOL Aircraft," and as a member of the stress group on the Arrow Wing Study (NAS1-12287).

J. H. LAAKSO–Structural Analysis

BSCE, Purdue University, 1961
MSCE, engineering mechanics, University of Washington, 1967

Mr. Laakso has been employed by The Boeing Company since 1962 in the aerospace structures technology field. He was initially involved with static and dynamic structural analyses in support of the company's advanced missile system programs. From October 1966 to October 1967, he was the department representative at Kirtland AFB coordinating technical data for the Minuteman missile system program and advanced-missile system survivability research programs.

Since 1967, Mr. Laakso has been engaged in structural development research on advanced composite material applications. This work involved the development and use of computer-aided design and analysis methods for preliminary design of various structural components. In addition, he was directly involved with subsequent fabrication and structural testing of numerous structural elements and components.

Mr. Laakso is currently in the Structural Development and Analysis Methods group and is technical leader on NASA-Langley contract NAS1-10860, "Evaluation of Metal Shear Webs Selectively Reinforced with Filamentary Composites for Space Shuttle Application." In this contract he has directed the development of a stiffened titanium-clad boron/epoxy shear web concept. He was directly involved with computer-aided design, analysis, element and component testing, and test data correlation activities.

R. A. KARNES—Structural Optimization Analysis

BCE, Cornell University, 1962
Postgraduate studies in structural engineering, Cornell University, 1962-1964 and engineering mechanics, University of Washington, 1968-1969

Mr. Karnes is a structural computing specialist in the Analysis Systems unit of Boeing Computing Services, Inc., where he has had extensive structural computing experience. He was a structural engineer with The Boeing Company for 7 years prior to joining BCS. After 1-1/2 years on the SST Structures staff, he joined the Stress Analysis Research group, where he first began his investigations in automated structural design. In 1966-1967 he worked with Dr. J. L. Tocher in the development of Boeing's first structural optimization digital computer program.

In 1968-1969 Mr. Karnes and Dr. Tocher again collaborated to write Boeing's "second-generation" automated structural design program. Of particular value is a new built-in mathematical feature to find an initial feasible design.

In 1971-1972 Mr. Karnes added buckling and thermal load capability to the optimization program and applied it to the automated design of a boron-reinforced space shuttle frame under NASA-Langley contract NAS1-10797.

Mr. Karnes is also experienced in system design. He spent 2 years (1969-1971) as lead engineer over Boeing's computerized preliminary design project, where he was responsible for the development of a computer software system for preliminary design and analysis of jet transport aircraft.

A. V. VISWANATHAN—Structural Instability Analysis

BS, physics, Madras University, India 1949
DMIT (aero), diploma in aeronautical engineering, Madras Institute of Technology, India, 1952
PhD, aeronautical engineering, Technical University of Norway, Norway, 1964

Since joining The Boeing Company in May 1969, Dr. Viswanathan has been with the Stress Analysis Research group. During this period he has worked on various structural stability problems. Dr. Viswanathan was technical leader for the buckling analysis work done under contract NAS1-8858, "Analytical and Experimental Investigation of Aircraft Metal Structures Reinforced With Filamentary Composites." Computer programs BUCLASP-2 and BUCLASP-3, for the buckling of composite stiffened panels subjected to biaxial mechanical loads and thermal loads, resulted from this work. Also, he was the technical leader for contract NAS1-11879, "Combined Load Buckling Analysis of Composite Stiffened Plates: Phase I." Computer program BUCLAP-2, for buckling of laminated composite flat and curved plates subjected to combined inplane normal and shear loads, was developed under this contract. He is currently lead engineer with responsibility for development of analysis methods for stress and stability of structures.

Previous experience between 1952 and 1967 includes: stress engineer, Handley Page (Reading), Ltd., England; research assistant in aeronautics, Technical University of Norway, Trondheim, Norway; assistant professor in aeronautics, Madras Institute of Technology, India and the Technical University of Norway, Trondheim, Norway; officiating professor in aeronautics, Technical University of Norway, Trondheim, Norway; professor in aeronautical engineering, Punjab Engineering College, India.

J. P. BARRY–Industrial Engineering

AA, Green River Community College, 1973

Mr. Barry joined the Boeing Commercial Airplane Company in early 1966. Since then he has had experience in fabrication, in bluestreak and jig building, and in Industrial Engineering as a methods analysis in shop work load control and estimating.

Mr. Barry is currently assigned to Operations Technology Support to Product Development as R&D proposal coordinator and estimator, and for 2 years has supported the Structures Technology organization in producibility and cost trade studies on programs such as the NASA-funded Advanced Technology Transport Program.

D. G. JEWETT–Cost Analysis

Mr. Jewett has been with Boeing for 23 years. He has 19 years' experience in finance estimating; 3 years as chief estimator on the SST; 3 years as chief of spares estimating: 4 years as chief estimator–new technology estimating; 5 years in major proposal estimating on C-5A, AWACS, navigator trainer, B-1 bomber; and 4 years on commercial program estimating and trade study evaluation. Mr. Jewett has estimated new design technology studies in depth on bonded honeycomb fuselage, bonded titanium wing box, composites, and related technological advancements. He was recently responsible for all trade studies on terminal area compatibility (TAC).

A. A. FIELD–Structural Design

Inter. BSc, engineering, University of London, England, 1936

Mr. Field's current assignment is design of brazed titanium sandwich structures.

Mr. Field joined Boeing in 1968 as a senior design engineer and has been engaged in research and development of advanced designs of fuselage structures. In this capacity he has closely supported the development of bonded honeycomb and composite-reinforced structures, evaluating the potential application of these forms of construction. He supplied design support for preparation of Boeing document D6-24800, "Graphite-Resin Composites, Feasibility Study, Reinforcement of Conventional Structures"; for NASA contract NAS1-11162, "Application of Filamentary Composites in a Commercial Jet Aircraft Fuselage"; and the Air Force contract F33615-72-1893, "Preliminary Design for a Critical Cargo/Tanker Component."

Prior to joining Boeing, Mr. Field was a supervisor at Short Bros. and Harland, Belfast, Northern Ireland, where he was responsible for fuselage structures of many projects from prototype design to production development.

C. W. WITHAM—Advanced Composites, Manufacturing Research and Development

BS, chemical engineering, University of Washington, 1961

Mr. Witham is currently assigned as supervisor of the Manufacturing Research and Development organization for adhesive bonding, structural glass, and advanced composites. Prior to this assignment, he supervised the Structural Fiberglass, High-Modulus Composites, and Acoustic Attenuation group, and the Nonmetals and Machining Development group for the Boeing SST program.

Mr. Witham's assignments prior to the SST program covered such technical areas as organic and inorganic finishing, aluminum and titanium cleaning, chemical milling and conversion coatings, sealants, high-temperature adhesive bonding, and titanium machining. Specific highlights include the implementation of an automated process line for aluminum cleaning and priming, and development and implementation of a stripping recovery system for acrylic-lacquer-coated panels.

Mr. Witham was program manager for Air Force contract F33615-72-C-1235, "Manufacturing Processes for Advanced Composite Substructural Shapes." Other contracts with which Mr. Witham is associated are listed as follows:

a) A Study of Effects on Long-Term Ground and Flight Environment Exposure on the Behavior of Graphite/Epoxy Spoilers, NAS1-11668

b) Application Study of Filamentary Composites in a Commercial Jet Aircraft Fuselage, NAS1-11162

c) Design and Fabrication of an Aeroelastic Flap Element for a Short Takeoff and Landing (STOL) Aircraft

d) Advanced Composite, High Altitude Drone, AFML contract F33615-872-2135

R. W. WHITFIELD—Manufacturing Research and Development

AB, engineering sciences, Dartmouth College, 1962
BE, ME, and DE (doctor of engineering), with dual majors in advanced materials/processes and management of technology, Thayer School of Engineering, an associated graduate school of Dartmouth College, 1964, 1968, and 1971

Since returning to Boeing, Dr. Whitfield has been involved in the development of low-cost aircraft structure and manufacturing research in support of new structural concepts. Previous experience with Boeing included a lead engineering position in developing and implementing fiberglass-reinforced assemblies and metal spraying on the 737 and 747 programs, and protective coatings development work on the Saturn program.

Dr. Whitfield left Boeing in 1967 to manage Thermal Dynamics Corporation's development of plasma arc processes and equipment for metal cutting, welding, and spraying. His engineering doctorate program at Dartmouth was done under NASA and NSF traineeships. Development work for the doctorate included efforts in the fields of computer simulation, plastic blow molding, rotogravure roll manufacture, and plasma arc welding processes. For the 2 years prior to returning to Boeing, Dr. Whitfield was involved in the development of new metal-casting processes and technoeconomic studies of advanced composites as director of Group Technical Development for Howmet Corporation.

5.0 APPLICABLE COMPANY EXPERIENCE

5.1 ADVANCED TECHNOLOGY PROGRAMS

The Boeing Company has the required background and directly related experience to successfully perform the proposed program. Boeing has been aware of the potential for improvement in structural efficiency to be achieved through the use of advanced material and design methods. Since 1964, the company has continuously sponsored major in-house research programs in these areas. In addition, numerous contracts related to these technologies have been performed. Table 5-1 lists a number of contracts recently completed or currently in work. A few of the programs directly related to this study are summarized.

Table 5-1.—Related Contracts

Contract	Title	Value ($)
NAS1-11668	A Study of the Effects of Long-Term Ground and Flight Environment Exposure on the Behavior of Graphite Spoilers	718,000
NAS1-8858	Analytical and Experimental Investigation of Aircraft Metal Structures Reinforced With Filamentary Composites	1,055,000
NAS1-10797	Evaluation of Metal Structures Reinforced With Filamentary Composites for Space Shuttle Application (Fuselage Frames)	275,000
NAS1-10860	Evaluation of Metal Structures Reinforced With Filamentary Composites for Space Shuttle Application (Shear Webs)	209,000
F33(615)67-C-1641	Development of Carbon-Composite Structural Elements for Missile Interstage Application	193,000
NAS1-10749	Design and Testing of Advanced Structural Panels	900,000
F33(615)-70-C-1541	Development of Fabrication Techniques for Boron/Aluminum Aircraft Structure	175,000
NAS1-11162	Application Study of Filamentary Reinforcement in a Commercial Jet Aircraft Fuselage	97,000

5.1.1 Model 737 Graphite/Epoxy Spoilers

Graphite/epoxy flight spoilers for the 737 have been developed for flight evaluation. The basic design consists of aluminum honeycomb core, stabilized graphite/epoxy skins, fiberglass closure ribs, and the same aluminum spar and hinge fittings used in the all-metal production spoiler. The graphite/epoxy design represents a 15% weight savings compared to the conventional aluminum spoiler.

FAA certification was granted in late 1971, and two spoilers were installed on a commercial 737 airplane under Boeing funding. To date, these components have each accumulated almost 3000 hr of flight time. This flight evaluation program is scheduled to continue for a minimum of 2 years. In 1972, NASA awarded contract NAS1-11668 to Boeing to fabricate and install 108 spoilers on Boeing 737 commercial airplanes. Of these components, 76 are now in commercial service and have accumulated over 50,000 flight-hours.

5.1.2 Analytical and Experimental Investigations of Aircraft Metal Structures Reinforced With Filamentary Composites

This program (NASA contract NAS1-8858) was conducted in three phases and demonstrated the feasibility of reinforcing metallic structure with filamentary composites; developed practical reinforcing concepts; investigated the creep, fatigue, and damage containment capabilities of this type of construction; and culminated in the fabrication and structural test of large-scale fuselage components using the reinforcing concepts developed. Aluminum and titanium alloys were reinforced with boron/epoxy and boron/polyimide composites.

5.1.3 Design and Testing of Advanced Structural Panels

The Boeing Company, under contract NAS1-10749, has evaluated structural concepts that use curved, cross-sectional elements to develop lightweight structural panels. The curved elements exhibit high, local-buckling strength, which leads to highly efficient structural concepts that may be applied where lightly beaded external surfaces are aerodynamically acceptable.

The design shapes were determined with the aid of an optimization computer code, which iterates geometric parameters to satisfy strength, stability, and weight constraints. The optimized design concepts offer 25% to 40% weight savings compared to conventional stiffened sheet construction.

5.1.4 Evaluation of a Metal Fuselage Frame Selectively Reinforced With Filamentary Composites for Space Shuttle Application (NAS1-10797)

This program is being performed to evaluate a metal fuselage frame selectively reinforced with filamentary composites for space shuttle application. In phase 1 the efforts were directed toward the development of the technology necessary to apply this design concept to a space shuttle airframe and, in particular, to the development of a full-scale, all-titanium frame design and a boron/epoxy-reinforced titanium frame design to establish a base for performing weight and cost studies. These results showed that the reinforced concept is 25.4% lighter than an equivalent all-metal (titanium) concept and that weight savings can be accomplished at a cost of approximately $200/lb. Structural elements, including subscale curved beams, were fabricated to demonstrate the reinforced concepts. Several structural element tests were performed to evaluate critical design areas. They were tested statically at room and elevated temperatures and under cyclic load and temperature conditions. The test results showed that the frame design studies were realistic.

Subsequently, a design of a one-third-scale model frame was developed. This frame was fabricated and tested in a manner that simulated critical design conditions. The frame exceeded the ultimate design load by 8%.

For the remaining work in this program, full-scale 8-ft curved beams are being fabricated and tested at room temperature and elevated temperature. The material systems and maximum use temperature that are being evaluated are: boron/epoxy/aluminum at 250° F, boron/epoxy/titanium at 375° F, and boron/polyimide/titanium at 600° F.

5.1.5 Evaluation of a Metal Shear Web Selectively Reinforced With Filamentary Composites for Space Shuttle Application

An advanced composite shear web design concept was developed in this program (NASA-Langley contract NAS1-10860). The concept resulted from an extensive computer-aided trade study (using OPTRAN) of numerous design candidates and is a practical configuration: titanium-clad ±45° boron/epoxy web plate with vertical boron/epoxy-reinforced aluminum stiffeners and a longitudinal aluminum stiffener. The weight savings offered by the development concept is 24% compared to all-metal construction (based on study of detailed drawings of flightweight hardware). Small-scale test elements were used to substantiate the design details in critical areas. A low-cost (minimum tooling) fabrication method was developed to produce future flight hardware. Three large-scale shear web components were fabricated and tested. Data from component tests were correlated with analysis methods (coded in the OPTRAN code for this concept) that provide a basis for preliminary design of the shear web concept in other applications.

5.1.6 Large-Space Telescope (LST) Metering Truss Module Development

A program has been initiated to develop a graphite/thermoplastic LST metering truss design and to fabricate and test a representative module of this design. This structure is very critical to the successful operation of the LST system. It controls the distance between the primary and secondary mirrors. Once the system begins to function, this distance must be held within ultrasmall tolerances. These small tolerances require that the truss structure remain relatively insensitive to temperature changes experienced in space. The graphite composites are very suitable for this application. The fibers have a coefficient of thermal expansion close to zero. Also, by proper orientation of the fibers in the truss elements, their coefficients of thermal expansion can be balanced such that the truss has a near-zero vertical growth due to temperature changes. Thermoplastic materials are being considered as the composite matrixes to provide a material that is inexpensive and potentially easy to fabricate when compared to epoxy composites.

Initially, structural configuration studies of the three-bay truss and its basic elements will be made in conjunction with theoretical parametric and supporting structural analysis to determine the best approaches for attaining thermal stability and stiffness requirements. Also, studies will be made to determine the designs most suitable for manufacturing. A conceptual sketch of the upper portion of the LST metering truss considered under this study is shown in figure 5-1.

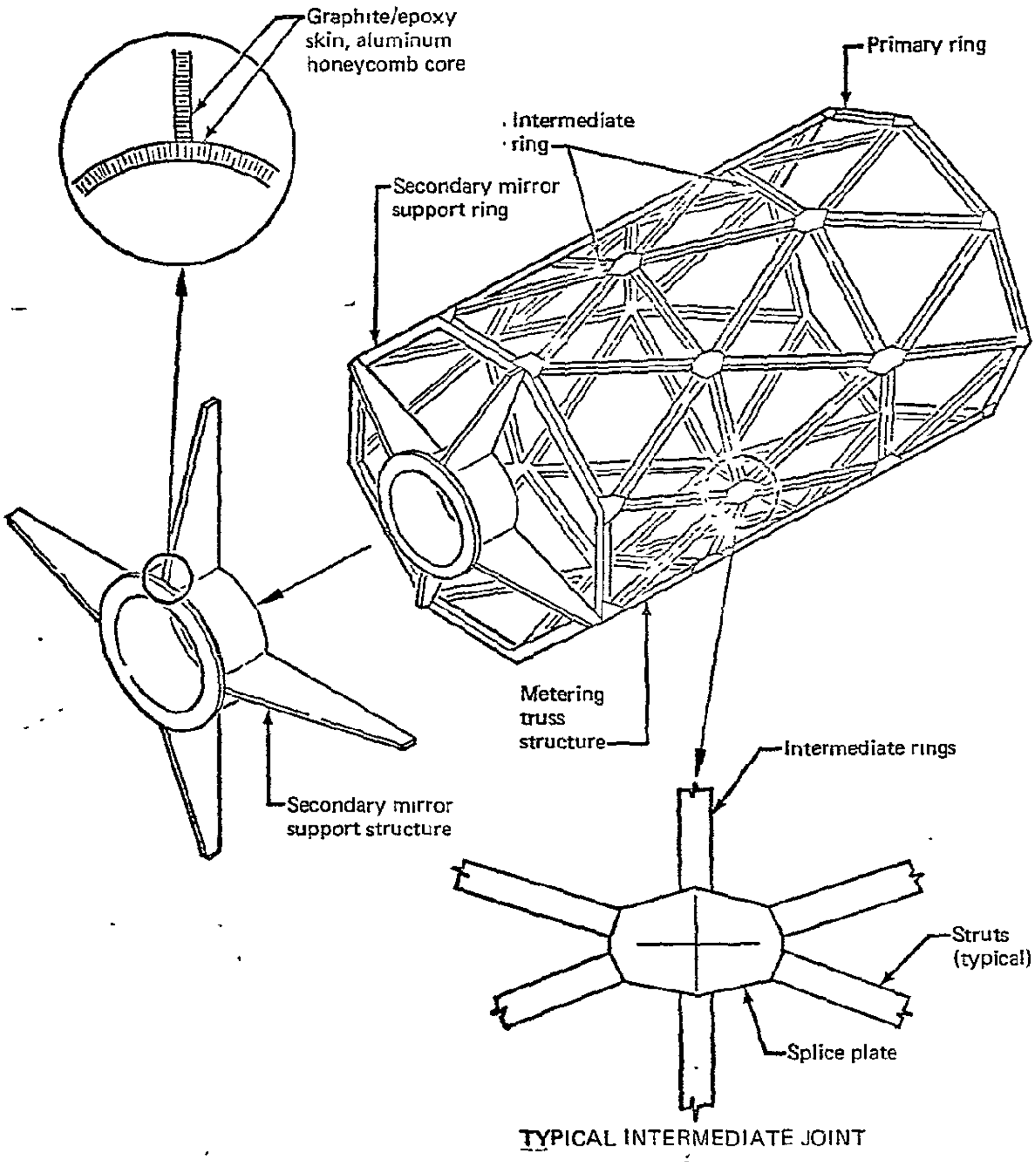

Figure 5-1.—LST Metering Truss Structural Configuration

5.2 OTHER RELATED PROGRAMS

Several automated structural design and optimization programs have been developed and used at Boeing. Programs capable of handling composite-reinforced structures as well as all-metal structural concepts are available. In addition to large-scale programs capable of analyzing and optimizing total airframe structure over a complete flight loading spectrum, small programs designed to analyze detailed design problems unique to aerospace structures have been developed. A brief summary of some of the pertinent programs is presented.

5.2.1 Automated Structural Design Program (TES-222)

The automated structural design program is a powerful engineering tool for designing structures of minimum weight that perform satisfactorily under a variety of design conditions. The program basically consists of a high-speed stress analysis combined with a general-purpose optimization program.

Given an initial structural design that satisfies a specific design criterion, TES-222 will determine a minimum-weight design that satisfies the same criterion. Starting with a geometric definition of the structure, a set of loading conditions, and material properties, the program will select the proper gages and cross-sectional areas to produce a minimum-weight design. It will not alter the basic geometric configuration of the structure, nor will it exceed the allowable stresses and displacements. Some design criteria, such as compression instability and thermal variations effects, are not considered but may be compensated for by artificially altering the material properties.

The TES-222 program was first applied at Boeing in the face sheet design for a 747 honeycomb sandwich window belt panel, illustrated in figure 5-2. The structure was idealized with a total of 300 nodes and 600 finite elements. Further optimization of a smaller portion of the panel between two adjoining windows was carried out. The final face sheet thickness distribution, shown in figure 5-3 combined with honeycomb sandwich, represents a 5-lb-per-window weight savings over the basic conventional design.

A second application was in the design of a typical frame of a supersonic bomber air inlet duct. This titanium frame, illustrated in figure 5-4, was subjected to a nonconstant internal pressure distribution, and its design was controlled by both stress and deflection limits. This continuous, wide-flange frame was idealized with 55 rod elements and 28 shear webs. The beam depths were fixed by design considerations, however, the flange areas could vary linearly along the sides. The TES-222 analysis and optimization of this frame permitted the removal of 15 lb of structural weight, representing a 14.3% weight reduction from the original frame design.

5.2.2 Space Shuttle Frame Analysis

A computer design and optimization program was developed in support of NASA contract NAS1-10797, "Evaluation of Metal Fuselage Frame Selectively Reinforced With Filamentary Composites for Space Shuttle Application." The frame analysis is achieved in two stages.

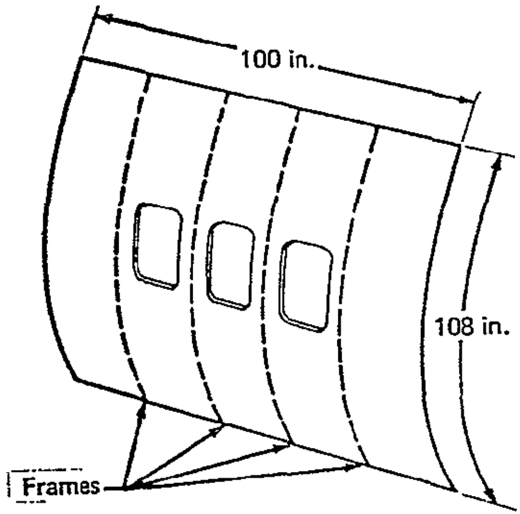

Figure 5-2.—747 Aircraft Window Belt Test Panel

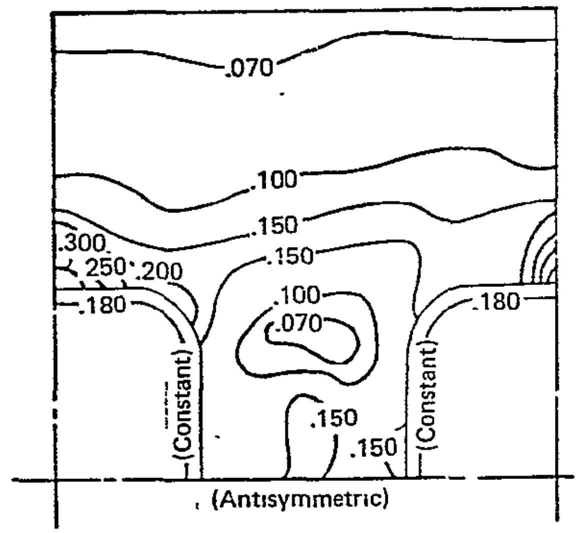

Figure 5-3.—Optimum Thickness Contours for Window Belt

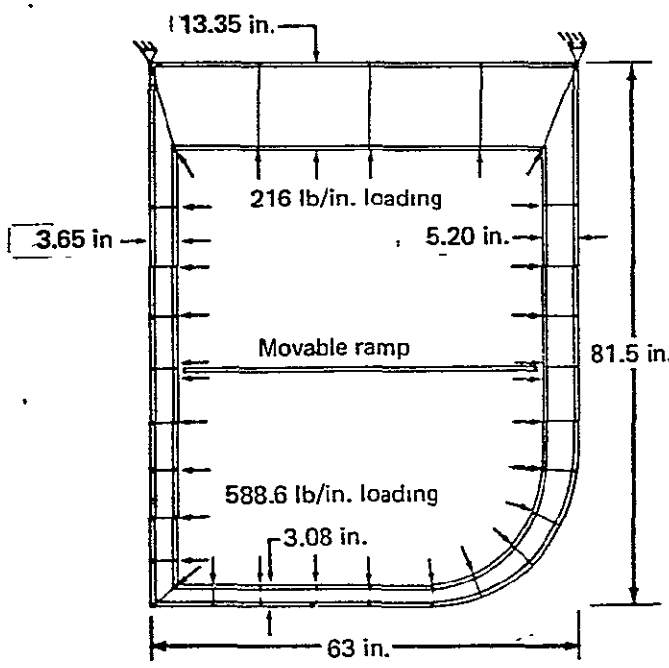

Figure 5-4.—Element Idealization of Inlet Duct Frame

Initially, the buckling program BUCLASP is used to compute tables of buckling strain as a function of geometric parameters for each frame flange configuration. These tabulated data could then be used as inputs to the structural optimization program TES-222 to produce a quasi-buckling constraint for each flange cross-sectional configuration. The complete structural frame is then optimized taking into account the variability of the component compression allowable stresses and component material properties.

For the typical flange cross section shown in figure 5-5, buckling allowables are determined for various ratios of metal area to composite area for a variety of stringer spacings. The frame web thickness and sandwich skin thicknesses remain constant. The allowable buckling strain for each geometry is then determined as the minimum of all possible failure modes, whether local, general, or both.

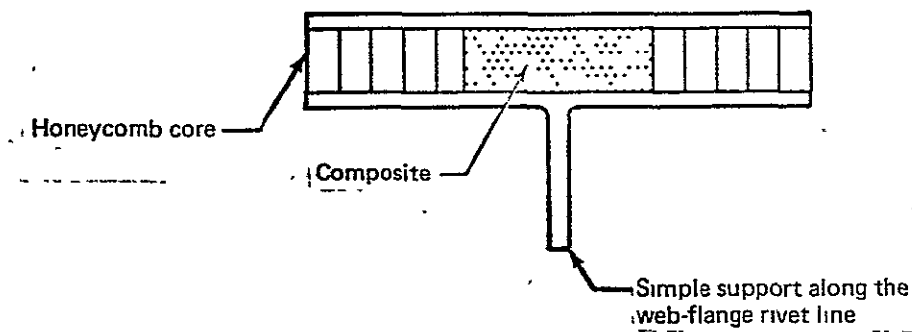

Figure 5-5.—Typical Space Shuttle Composite-Reinforced Frame Concept

The TES-222 program designs on the basis of stress margins; that is, the difference between the actual stress and the allowable stress. In computing the stress margins, the program first calculates the stress in each element. If the stress is tensile, the stress margin is computed directly. However, if the stress is compressive, the element must be checked for a critical buckling allowable. The influence of coincident elements is also evaluated. The compression stress margin is then based on the minimum critical buckling stress.

5.2.3 747 Composite-Reinforced Window Belt Panel—Finite-Element Analysis

A finite-element program, based on a forced-displacement, plane stress analysis concept, was developed to analyze a boron/epoxy-reinforced 747 window belt panel. An orthotropic plate element was developed for use in the finite analysis.

The analysis was conducted in three stages:

a) Solution for three basic distortion modes of unit amplitude (circumferential extension, longitudinal extension, and shear deformation) for an infinitely long panel.

b) Extraction of a three-window panel from the infinitely long panel for analysis of critical design load cases by proper scaling and superposition of the unitary distortion cases.

c) A fine-grid analysis, performed on a quarter window section.

The output from this analysis consists of the following:

a) Nodal displacements and nodal forces, including reactions.

b) Strain values at nodal points and at the center of each grid element. The orthogonal x,y strain components, the principal strains, and the maximum shear strains are computed. Subroutines may be used to resolve the strains in a particular fiber reinforcement direction.

c) Computations of the corresponding orthogonal stress components, principal stresses, and maximum principal shear stresses. Orthotropic, elastic, material properties are used in these computations.

5.2.4 BUCLASP-2—Stiffened-Panel Buckling Load Program

This program calculates critical biaxial compressive buckling loads on stiffened plates built up of one or more orthotropic laminated flat or curved plate elements and beam elements. Buckling analyses of corrugated plates, integral panels, and truss-core sandwich panels, for example, can be performed. The placement of stiffeners in panels generally has a repetitive nature. This feature is recognized by the program, so the required amount of input data is kept at a minimum. A panel, in general, consists of rectangular plate elements interconnected and stiffened by beam elements along their longitudinal direction. Simple-support boundary conditions are assumed along loaded plate-element edges and at the ends of beam elements that are loaded. Along each interelement boundary, displacement compatibility and equilibrium are satisfied. A choice of free, simply supported, or clamped boundary conditions is available for the exterior panel edges that are not loaded. The program can handle a maximum of 25 plate elements with up to 25 laminas and 10 beam elements with up to 35 laminas.

The analytical method used to calculate the buckling loads is based on the direct-stiffness finite-element theory. The stiffness matrices used in the eigenvalue equations to determine the buckling loads are derived from the exact sinusoidal lengthwise variation of displacements. The buckled-pattern wavelength is inherent in the resulting stiffness matrices, but this dependence is of a very simple form since all stiffness coefficients contain the buckling wavelengths as a common factor. The buckling panel load is defined as the lowest load level that causes any plate element buckling (local mode) or the simultaneous buckling of several plate elements (general mode). By suitable modeling techniques, it is possible to differentiate between a bonded and a riveted stiffener Additionally, the program can be used to develop design curves for stiffened plates with laminated composite reinforcements.

This program was developed for NASA-Langley under contract NAS1-8858, "Analytical and Experimental Investigation of Aircraft Metal Structures Reinforced With Filamentary Composites."

5.2.5 OPTRAN

OPTRAN (optimization by random design search algorithm) is a general multivariable search optimization code developed by Boeing specifically for structural component optimization. This code was derived from experience with the Boeing-developed AESOP program in optimizing panel and cylinder designs. Program AESOP was developed by Boeing for NASA-Ames and has also been used at NASA-Langley, NASA-MSFC, NASA-Lewis, and Air Force Flight Dynamics Laboratory. OPTRAN will optimize an objective function of up to 100 variables subject to as many as 20 constraint functions. The optimization problem can be treated as a totally constrained or as a quasi-unconstrained problem. Synthesis tasks using SPEED and OPTRAN codes may be specialized with code modules for particular structural forms. Weight, stiffness, and failure-mode analyses may be treated for a specified structural form The OPTRAN coding is currently capable of handling the structural configurations listed in table 5-2, which may consist of metal, composite, or composite-reinforced metal structure. Figure 5-6 shows the SPEED (synthesis procedure for establishing element data) code organization (ref. 6). The OPTRAN code, used in numerous NASA contracts (refs. 2, 4, and 8), is employed as a subroutine to the SPEED code to generate an optimum design for a specified load/length/concept case. The SPEED code is executed for a number of load cases and finally outputs an optimum design data bank for a specific length/concept case.

Table 5-2.—Structural Configuration Applications of OPTRAN

Configuration	Loading	Construction concept	
Panels	Compression	Honeycomb sandwich Truss-core sandwich Web-core sandwich Stiffened monocoque Corrugated monocoque	
Cylindrical shells	Compression External pressure	Honeycomb sandwich Truss-core sandwich Web-core sandwich Stiffened monocoque Corrugated monocoque	With or without ring stiffeners
Cylindrical compression	Internal pressure External pressure	Stiffened monocoque Corrugated monocoque	With or without overwrap and ring stiffeners
Tubular strut	Compression	Monocoque—homogeneous or laminated	

OPTRAN establishes candidate designs by random selection of dimensional variable values from user-specified input search ranges The program then checks for weight savings relative to the best preceding design. If the candidate design offers weight savings, then the failure mode constraints are checked in succession. A design that survives all of the failure mode checks then becomes the best current design. However, if a violation of a failure mode constraint occurs, a new random design is established without wasting time on further

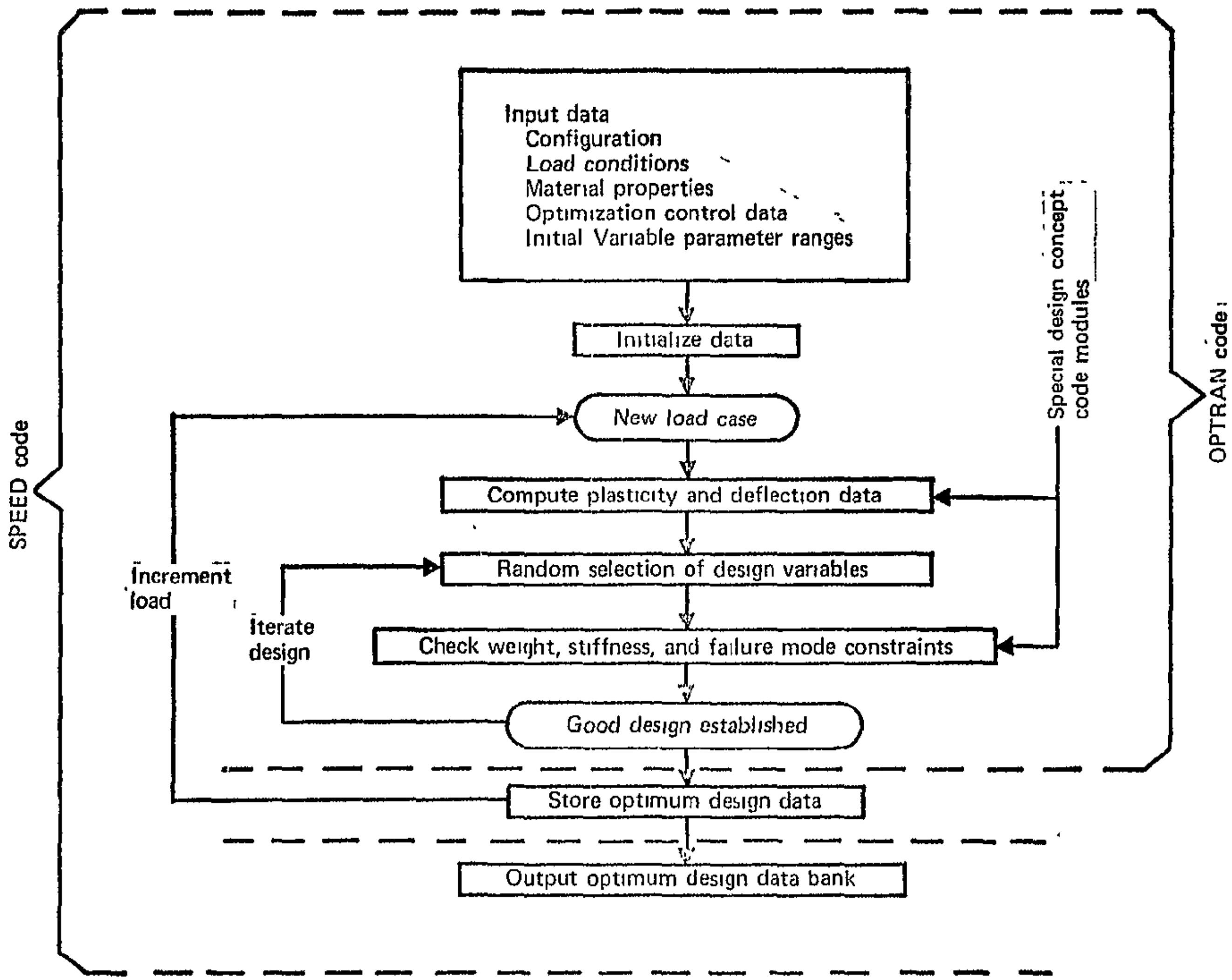

Figure 5-6.–SPEED Code Organization

failure mode checks. Ordering of the analyses so that governing constraints are treated first decreases run time. After a certain number of good designs have been found, the parameter search ranges are compressed in a manner that is adaptive to the history of the best previous designs, and then another search cycle is conducted. If one variable shows greater variation from cycle to cycle, its range is made broader to increase the probability of directing the design to a true minimum. A large number of good designs (e.g., five) in the first search cycle expedites establishment of minimum-weight designs in subsequent cycles. The search cycles are terminated after the search ranges converge to a desired minimum size. The use of discrete variables (standard structural sections, number of composite laminate plies, etc.) presents no difficulties to this search method. Additional discussion concerning the optimization strategy used in OPTRAN may be found in references 2 and 9.

5.2.6 Skin/Stringer Column Analysis and Optimization Program

Boeing has developed a computer program for quick preliminary analysis and optimization of skin/stringer columns. This capability is particularly valuable for the comparison of structural concepts during preliminary design when consistent analysis methods, criteria, and allowables are necessary for rapid evaluation of the various concepts. Detailed weight and strength data can be generated early in a program in order to evaluate the impact of using nonoptimum structures or revising material allowables.

Analysis capabilities include a wide variety of skin/stringer configurations including riveted, bonded, integral machined, sandwich skin, padded skin, and composite-reinforced configurations. The program has the ability to handle both isotropic and orthotropic materials.

The program serves a dual purpose in that it has the capability to perform a buckling analysis of a given structural configuration or to optimize the stringer elements for a given skin thickness and stringer spacing. It is possible to optimize each stringer element width and thickness, or to optimize either the element width or thickness separately. The constrained optimization condition may arise as a result of consideration of minimum gages, manufacturing techniques, or other constraints. The basic constraints on the geometry and the elements that may be imposed are as follows:

a) Minimum thickness

b) Minimum length

c) Constant thickness

d) Constant length

e) Stringer flange length-to-height ratio

f) Skin-thickness-to-smeared-thickness ratio

If the structural configuration is symmetrical, the structural allowables are calculated based on Johnson-Euler buckling. If the structure is not symmetrical, the program will calculate an allowable based on the Argyris coupled flexure-torsion buckling mode. The isotropic material element crippling stresses are calculated using either the equation that can be generated from the input data or actual test data that can be input. Orthotropic plate buckling theory is used to calculate the allowables for the orthotropic elements using the data specified in the materials input section.

5.2.7 Laminated Composite Analysis and Optimization Computer Program (COOP)

A computer program has been developed to analyze and optimize laminated composite plates and honeycomb sandwich panels subject to strength, thermal load, and buckling criteria. The stiffness and strength analysis subroutines may be used for the analysis of symmetrical or unsymmetrical laminates under inplane applied and thermal loads. Buckling equations for simply supported orthotropic plates or honeycomb sandwich panels under biaxial loads, including the effects of interlaminar shear stiffness, are used. A nonlinear stress analysis is performed by dividing the analysis into a series of linear steps. The optimization subroutine uses a modification of the steepest descent method in which sidesteps are made in the direction of minimum weight. Laminate thicknesses, including honeycomb sandwich thickness, are optimized for strength and buckling constraints while holding layer orientations constant.

6.0 FACILITIES AND EQUIPMENT

6.1 MANUFACTURING CAPABILITY

Boeing maintains manufacturing capability in the technical areas of fiberglass lamination, bonded honeycomb sandwich structures, and compression molding, as well as capability in standard metal airframe fabrication to support general production requirements. Research and development capability has additionally been developed to expand the use of filamentary composite materials. Titanium fabrication facilities were developed in support of the United States SST program.

Most advanced composite components are fabricated at the company's Fabrication and Services Division located in Auburn, Washington. The Process Assembly Organization—Advanced Structures Shop of this division is a well-equipped production facility that has been set up especially to fabricate advanced composite hardware. This facility is supplemented by those available in the adjacent manufacturing area and in the Manufacturing Research and Development and Materials Technology Laboratories.

Personnel working in the Advanced Structures Shop have been specially screened and selected for this assignment. The facility provides excellent support to technical staffs in fabricating high-quality development hardware with minimum response time and at reasonable cost. Equipment is available in this shop to obtain actual manufacturing costs at the factory work code level (such as layup, bagging cure, debagging).

6.2 COMPUTER CAPABILITY

Digital computers and associated equipment at Boeing are housed primarily in two locations: the Boeing Computer Center in Seattle and building 10-80 in Renton. Remote computers directly transmit magnetic tape data between outlying facilities and the two computer centers to expedite the flow of input and output information. The digital computers currently in operation include 11 large-scale, third-generation machines (CDC 6600 and IBM 360 models 50, 65, and 75) and six IBM 7000 computers. Remote input terminals are installed at various plants in the Seattle, Renton, Kent, Everett, and Auburn areas for the direct acquisition of data at its source.

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1. CONTROL NUMBER 084112

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BATCH NUMBER 08634

2. SOURCE CODE: NASA

4. ACCESSION NUMBER: N78-78237

2A. RECOGNITION NUMBER: NASA-CR-157506

10. TRANSACTION N NEW D DUPLICATE P PRIOR S SUPERSEDES: N

10A.

6. MGT. CD. C D A R K L: Q X B

7. PROCESS ACCTG. CD.: 0N

7A. REGISTRATION FEE

8. PROD. MGR.: H

9. RECEIPT TYPE 0 N/A 1 RETAIN 2 LAST 3 ERRATA 4 LOAN: 1

9A. LOAN DUE OUT Y Y D D D

11. PRICE 1. STD. 2. EXC.

12. RETURNS 0 N/A 1 PAGES 3 REPRO 4 BDS 5 EDIT 6 MISC: 0

12A. RETURN DATE Y Y M M D D

13. PROCESS 0 STANDARD 1 INVENTORY 2 HIGHLIGHT 3 UNEDIT 4 REANNOUNCE 5 SPECIAL B/B: 1

14. PAT. POTENTIAL 0 NO 1 POOR 2 FAIR 3 GOOD 4 EXCEL.: 0

16. BILL

17. FORM/PRICE: 12001

18. ANNOUNCE GRA WGA D/B

19. PUBLICATION 1: U8014

20. PUBLICATION-2

20A. PUBLICATION-3

21. LIMITATION 0 N/A 1 DISCLAIMER 2 COLOR 3 COMBO: 0

23. PC BIN STOCK: 000 0 N

24A PAGES/SHEETS: 60

PRICE CODE: A04 / X00

DOMESTIC PRICE

FOREIGN PRICE

ACTION CD.: BX / XM

MICROFICHE

37. REL. CD.: A

38. MF PRINT: D

39. ADD. INFO.: A D

40. PRINT PC

43. 43A) ORDER ADD'L STOCK & CODE/ADDR 43B) BIN MSG 43C) STK UNIT COST 43D) SET ID 43E) CAMERA CONTROL 43F) REFER TO ADDR

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