Development of Advanced Methods of Structural and Trajectory Analysis for Transport Aircraft

Annual Report

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Santa Clara University Santa Clara, CA 95053

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Introduction

This report summarizes work accomplished under NASA Grant NCC2-5167, "Development of Advanced Methods of Structural and Trajectory Analysis for Transport Aircraft," during the first year of the grant October 1, 1995 - September 30, 1996. The effort was in two areas: (1) development of advanced methods of structural weight estimation, and (2) development of advanced methods of flight path optimization.

During the Spring of 1996 both graduate student research assistants working on the project, H.C. Chou and Mark Chambers, resigned to take positions in industry. This required assigning three new Santa Clara people to the project: Dr. Lee Hornberger, Associate Professor of Mechanical Engineering; Robert Windhorst, graduate student research assistant; and Frank Dickerson, undergraduate student. These new people inevitably required time to learn the ACSYNT code and the nature of the ongoing research. The result is that some of the tasks in the work statement have not been completed at this time, but will be at the completion of the Grant.

Dr. H. Miura and M. Moore were the NASA collaborators on the Grant.

Review of Results in Structural Weight Estimation

A report that was prepared under a previous grant was published in May 1996 ("Analytical Fuselage and Wing Weight Estimation of Transport Aircraft," by M. Ardema, M. Chambers, A. Patron, A. Hahn, H. Miura, and M. Moore, NASA TM 110392). A paper that summarizes this report has been accepted for presentation at the World Aviation Congress to be held in October 1996; a copy of this paper appears in Appendix A.

Throughout the year, integration of the structural weight computer code, PDCYL, into ACSYNT has continued. Input variables used by PDCYL but already in ASCYNT have been removed from PDCYL. Infrequently used input variables have been defaulted. Data transfer has been modified so that optimization runs with ACSYNT can be done with PDCYL as an integral part of the code.

The major effort to the first year of the grant was to develop an improved method of estimating the weight of wing and fuselage structures made from composite materials. This involved an extensive literature search, the coding of a composite materials subroutine, and demonstrating the code. This work is discussed in detail in Appendix B.

Previously in ACSYNT, the weight of composite material structures was estimated assuming quasi-isotropic materials, maximum stress failure theory, and smeared structural elements. The capability being developed accounts for realistic lay-ups of unidirectional fiber/matrix composites and uses a bi-axial strain failure theory. The new composite routine has been implemented for the fuselage weight calculation and will be implemented for the wing weight calculation in the second year of the grant.

A user's manual for the new composite subroutine may be found in Appendix B. As a check case for the new subroutine, the weight of a composite fuselage of the ASA 2150 has been estimated.

The final effort in the structures area has been support of the project to design and analyze a 150 passenger advanced transport airplane, the ASA 2150. PDCYL has been used as an integral part of ACSYNT to estimate the fuselage and wing weights of this aircraft. Appendix C gives the details of the weight calculations for both Aluminum and Graphite/Epoxy fuselage versions of the ASA 2150.

Appendix C shows that at a gross take-off weight of 152,181 pounds, the ASA 2150 is estimated to have a wing weight of 10,315 pounds and a fuselage weight of 15,652 pounds when made of Aluminum. Figure 1 shows the ASA 2150 fuselage bending moment distribution. The critical loading condition for most of the fuselage is either the landing condition (L) or the runway bump condition (B), with a small portion governed by the maneuver condition (M). The shell unit weight distribution is shown on Figure 2. Approximately the first half of the fuselage is sized by minimum gage, with most of the rest yield strength critical.

When the fuselage is made of composite material, the weight is estimated to be 15,375 pounds, a weight savings of about 2% relative to aluminum. The composite material is a uni-directional tape made from Hercules AS4 carbon fiber in Fiberite 12K/938 resin. The reason for this relatively low weight savings is that for relatively small and lightly loaded aircraft such as the ASA 2150, the fact that the composite material thickness must be in integer thicknesses of the basic stack thickness means that the structure is in many places considerably overdesigned. The basic stack used was a quasi-isotropic lay-up of eight unidirectional plys. Also, the nonoptimun factor used for the composite was 17% higher than that for the Aluminum design. As for the Aluminum design, the composite fuselage was sized by minimum gage and yield strength.

Review of Results in Trajectory Optimization

Because of the unexpected loss of a senior, experienced research assistant, the analysis of the altitude jumps in energy climb paths could not be completed. Rather, a new, basic look was taken at energy-state and related approximations. This analysis is related in Appendix D and will result in an important new addition to the trajectory optimization in ACSYNT.

Previously, the approach was to minimize a weighted sum of time and fuel consumption:

$$\mathbf{J} = \int_0^d \left(K_1 + K_2 CT \right) dt$$

Under energy-state approximation this may be written

$$\mathbf{J} = \int_{O}^{d} \left(\frac{K_1 + K_2 CT}{P} \right) dE$$

where

C

$$P=\frac{V(T-D)}{W}$$

Thus the optimal climb path is given by

$$\frac{\max\left(\frac{P}{K_1+K_2CT}\right)}{h}\Big|_{E=const}$$

This analysis, however, is for range not specified. When range is specified, the optimal path is determined by

$$\frac{\max}{h} \left(\frac{P}{K_1 + K_2 CT - \lambda_X V} \right) \Big|_{E=const.}$$

Where λ_x is the adjoint variable associated with the range equation. It is determined from

$$\lambda_X = \frac{K_1 + K_2 C_C T_C}{V_C}$$

Where C_c , T_c , and V_c are associated with the optimal cruise point, obtained by maximizing the Brequet factor within the flight envelope

$$\max_{h,E}\left(\frac{V(L/D)}{C}\right)$$

.

A search for the optimal cruise point, and the evaluation of λ_x , is being added to the ACSYNT code.





Figure 2 , Shell Unit Weight Distribution (ASA 2150)

Body station (ft)

Appendix A

"Fuselage and Wing Weight of Transport Aircraft"

by

M. Ardema, M. Chambers, A. Patron, A. Hahn, H. Miura, and M. Moore

to be presented at

World Aviation Congress October 22 - 24, 1996 Los Angeles, California

October 1996

Fuselage and Wing Weight of Transport Aircraft

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Andrew S. Hahn, Hirokazu Miura, and Mark D. Moore NASA Ames Research Center

> 1996 World Aviation Congress October 21-24, 1996 Los Angeles, CA



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Fuselage and Wing Weight of Transport Aircraft

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ABSTRACT

A method of estimating the load-bearing fuselage weight and wing weight of transport aircraft based on fundamental structural principles has been developed. This method of weight estimation represents a compromise between the rapid assessment of component weight using empirical methods based on actual weights of existing aircraft, and detailed, but time-consuming, analysis using the finite element method. The method was applied to eight existing subsonic transports for validation and correlation. Integration of the resulting computer program, PDCYL, has been made into the weightscalculating module of the AirCraft SYNThesis (ACSYNT) computer program. ACSYNT has traditionally used only empirical weight estimation methods; PDCYL adds to ACSYNT a rapid, accurate means of assessing the fuselage and wing weights of unconventional aircraft. PDCYL also allows flexibility in the choice of structural concept, as well as a direct means of determining the impact of advanced materials on structural weight.

INTRODUCTION

A methodology based on fundamental structural principles has been developed to estimate the load-carrying weight of the fuselage and basic box weight of the wing for aircraft, and has been incorporated into the AirCraft SYNThesis program (ACSYNT). This weight routine is also available to run independently of ACSYNT, and is a modification of a collection of previously developed structural programs.¹⁻⁴ The main subroutine called by ACSYNT is PDCYL. This study has concentrated on modern transport aircraft because of the detailed weight information available, allowing the weights output from PDCYL to be compared to actual structural weights. The detailed weight statements also allow nonoptimum factors to be computed which, when multiplied by the load-bearing structural weights calculated by PDCYL, will give good representative total structure weight estimates. These nonoptimum factors will be computed through a regression analysis of a group of eight transport aircraft.

PDCYL is able to model both skin-stringer-frame and composite sandwich shell fuselage and wing box constructions. Numerous modifications were made to PDCYL and its associated collection of subroutines. These modifications include the addition of detailed fuselage shell geometry calculations; optional integration of a cylindrical fuselage midsection between the nose and tail sections; addition of landing and bump maneuvers to the load cases sizing the fuselage; ability to introduce an elliptical spanwise lift load distribution on the wing; variation of wing thickness ratio from tip to root; ability to place landing gear on the wing to relieve spanwise bending loads; distribution of propulsion system components between wing and fuselage; and the determination of maximum wingtip deflection.

BRIEF DESCRIPTION OF ACSYNT

The Aircraft Synthesis Computer program, ACSYNT, is an integrated design tool used in the modeling of advanced aircraft for conceptual design studies.⁵ ACSYNT development began at NASA Ames Research Center in the 1970s and continues to this day. The ACSYNT program is quite flexible and can model a wide range of aircraft configurations and sizes, from remotely piloted high altitude craft to the largest transport.

The ACSYNT program uses the following modules, not necessarily in this order: Geometry, Trajectory, Aerodynamics, Propulsion, Stability, Weights, Cost, Advanced Aerodynamic Methods, and Takeoff. An ACSYNT run would normally progress as follows: the Geometry module is called to define the aircraft shape and configuration; the Trajectory module then runs the vehicle through a specified mission; finally the Weight and Cost modules are executed. To determine the performance of the vehicle at each mission point, the Trajectory module will call the Aerodynamics and Propulsion modules.

^{*} Work of the first two authors was supported by NASA Ames Research Center Grant NCC2-5068.

After the mission is completed, the calculated weight of the aircraft may be compared with the initial estimate and an iteration scheme run to converge upon the required aircraft weight. This process is necessarily iterative as the aircraft weight ACSYNT calculates is dependent upon the initial weight estimate.

ACSYNT is able to perform a *sensitivity analysis* on any design variable, such as aspect ratio, thickness-to-chord ratio, fuselage length or maximum fuselage diameter. Sensitivity is defined as (change in objective function/value of objective function) divided by (change in design variable/design variable). As an example, if gross weight is the objective function and decreases when the wing thickness-to-chord ratio increases, then the sensitivity of thickness-to-chord ratio is negative. It is important to note that while this increase in thickness-to-chord ratio lowers the gross weight of the aircraft, it may also have a detrimental effect on aircraft performance.

ACSYNT is also able to size multiple design variables by optimizing the objective function. The objective function represents the interactions between design disciplines such as structures, aerodynamics and propulsion. The automated sizing of design variables during the optimization process is accomplished using the gradient method. Two types of constraints may be imposed during the optimization process. These are performance-based constraints such as runway length or maximum roll angle, and side constraints on design variables such as limitations on wing span or fuselage length. ACSYNT never violates constraints during the optimization process so that each iteration produces a valid aircraft.

METHODS OF WEIGHT ESTIMATION

Two methods are commonly available to estimate the load-bearing fuselage weight and wing box structure weight of aircraft. These methods, in increasing order of complexity and accuracy, are empirical regression and detailed finite element structural analysis. Each method has particular advantages and limitations which will be briefly discussed in the following sections. There is an additional method based on classical plate theory (CPT) which may be used to estimate the weight of the wing box structure.

EMPIRICAL

The empirical approach is the simplest weight estimation tool. It requires knowledge of fuselage and wing weights from a number of similar existing aircraft in addition to various key configuration parameters of these aircraft in order to produce a linear regression. This regression is a function of the configuration parameters of the existing aircraft and is then scaled to give an estimate of fuselage and wing weights for an aircraft under investigation. Obviously, the accuracy of this method is dependent upon the quality and quantity of data available for existing aircraft. Also, the accuracy of the estimation will depend on how closely the existing aircraft match the configuration and weight of the aircraft under investigation. All of the empirical regression functions currently in the ACSYNT program give total fuselage weight and total wing weight.

FINITE ELEMENT

Finite element analysis is the matrix method of solution of a discretized model of a structure. This structure, such as an aircraft fuselage or wing, is modeled as a system of elements connected to adjacent elements at nodal points. An element is a discrete (or finite) structure that has a certain geometric makeup and set of physical characteristics. A nodal force acts at each nodal point, which is capable of displacement. A set of mathematical equations may be written for each element relating its nodal displacements to the corresponding nodal forces. For skeletal structures, such as those composed of rods or beams, the determination of element sizing and corresponding nodal positioning is relatively straightforward. Placement of nodal points on these simple structures would naturally fall on positions of concentrated external force application or joints, where discontinuities in local displacement occur.

Continuum structures, such as an aircraft fuselage or wing, which would use some combination of solid, flat plate, or shell elements, are not as easily discretizable. An approximate mesh of elements must be made to model these structures. In effect, an idealized model of the structure is made, where the element selection and sizing is tailored to local loading and stress conditions.

The assembly of elements representing the entire structure is a large set of simultaneous equations that, when combined with the loading condition and physical constraints, can be solved to find the unknown nodal forces and displacements. The nodal forces and displacements are then substituted back into the each element to produce stress and strain distributions for the entire structural model.

CLASSICAL PLATE THEORY

CPT has been applied to wing structure design and weight estimation for the past 20 years. Using CPT a mathematical model of the wing based on an equivalent plate representation is combined with global Ritz analysis techniques to study the structural response of the wing. An equivalent plate model does not require detailed structural design data as required for finite element analysis model generation and has been shown to be a reliable model for low aspect ratio fighter wings. Generally, CPT will overestimate the stiffness of more flexible, higher aspect ratio wings, such as those employed on modern transport aircraft. Recently, transverse shear deformation has been included in equivalent plate models to account for this added flexibility. This new technique has been shown to give closer representations of tip deflection and natural frequencies of higher aspect ratio wings, although it still overestimates the wing stiffness. No fuselage weight estimation technique which corresponds to the equivalent plate model for wing structures is available.

NEED FOR BETTER, INTERMEDIATE METHOD

Preliminary weight estimates of aircraft are traditionally made using empirical methods based on the weights of existing aircraft, as has been described. These methods, however, are undesirable for studies of unconventional aircraft concepts for two reasons. First, since the weight estimating formulas are based on existing aircraft, their application to unconventional configurations (i.e., canard aircraft or area ruled bodies) is suspect. Second, they provide no straightforward method to assess the impact of advanced technologies and materials (i.e., bonded construction and advanced composite laminates).

On the other hand, finite-element based methods of structural analysis, commonly used in aircraft detailed design, are not appropriate for conceptual design, as the idealized structural model must be built off-line. The solution of even a moderately complex model is also computationally intensive and will become a bottleneck in the vehicle synthesis. Two approaches which may simplify finite-element structural analysis also have drawbacks. The first approach is to create detailed analyses at a few critical locations on the fuselage and wing, then extrapolate the results to the entire aircraft, but this can be misleading because of the great variety of structural, load, and geometric characteristics in a typical design. The second method is to create an extremely coarse model of the aircraft, but this scheme may miss key loading and stress concentrations in addition to suffering from the problems associated with a number of detailed analyses.

The fuselage and wing structural weight estimation method employed in PDCYL is based on another approach, beam theory structural analysis. This results in a weight estimate that is directly driven by material properties, load conditions, and vehicle size and shape, and is not confined to an existing data base. Since the analysis is done station-bystation along the vehicle longitudinal axis, and along the wing structural chord, the distribution of loads and vehicle geometry is accounted for, giving an integrated weight that accounts for local conditions. An analysis based solely on fundamental principles will give an accurate estimate of structural weight only. Weights for fuselage and wing secondary structure, including control surfaces and leading and trailing edges, and some items from the primary structure, such as doublers, cutouts, and fasteners, must be estimated from correlation to existing aircraft.

The equivalent plate representation, which is unable to model the fuselage structure, is not used in PDCYL.

METHODS

OVERVIEW

Since it is necessary in systems analysis studies to be able to rapidly evaluate a large number of specific designs, the methods employed in PDCYL are based on idealized vehicle models and simplified structural analysis. The analyses of the fuselage and wing structures are performed in different routines within PDCYL, and, as such, will be discussed separately. The PDCYL weight analysis program is initiated at the point where ACSYNT performs its fuselage weight calculation. PDCYL first performs a basic geometrical sizing of the aircraft in which the overall dimensions of the aircraft are determined and the propulsion system, landing gear, wing, and lifting surfaces are placed.

Fuselage

The detailed fuselage analysis starts with a calculation of vehicle loads on a station-by-station basis. Three types of loads are considered-longitudinal acceleration (applicable to high-thrust propulsion systems), tank or internal cabin pressure, and longitudinal bending moment. All of these loads occur simultaneously, representing a critical loading condition. For longitudinal acceleration, longitudinal stress resultants caused by acceleration are computed as a function of longitudinal fuselage station; these stress resultants are compressive ahead of the propulsion system and tensile behind the propulsion system. For internal pressure loads, the longitudinal distribution of longitudinal and circumferential (hoop) stress resultants is computed for a given shell gage pressure (generally 12 psig). There is an option to either use the pressure loads to reduce the compressive loads from other sources or not to do this; in either case, the pressure loads are added to the other tensile loads.

The following is a summary of the methods used; the details may be found in Ref. 6.

Longitudinal bending moment distributions from three load cases are examined for the fuselage. Loads on the fuselage are computed for a quasi-static pull-up maneuver, a landing maneuver, and travel over runway bumps. These three load cases occur at user-specified fractions of gross takeoff weight. Aerodynamic loads are computed as a constant fraction of fuselage planform area and are considered negligible for subsonic transports. For pitch control there is an option to use either elevators mounted on the horizontal tail (the conventional configuration) or elevons mounted on the trailing edges of the wing. The envelope of maximum bending moments is computed for all three load cases and is then used to determine the net stress resultants at each fuselage station.

After the net stress resultants are determined at each fuselage station, a search is conducted at each station to determine the amount of structural material required to preclude failure in the most critical condition at the most critical point on the shell circumference. This critical point is assumed to be the outermost fiber at each station. Failure modes considered are tensile yield, compressive yield, local buckling, and gross buckling of the entire structure. A minimum gage restriction is also imposed as a final criterion. It is assumed that the material near the neutral fiber of the fuselage (with respect to longitudinal bending loads) is sufficient to resist the shear and torsion loads transmitted through the fuselage. For the shear loads this is a good approximation as the fibers farthest from the neutral axis will carry no shear. Also, for beams with large fineness ratios (fuselage length/maximum diameter) bending becomes the predominant failure mode.

The maximum stress failure theory is used for predicting yield failures. Buckling calculations assume stiffened shells behave as wide columns and sandwich shells behave as cylinders. The frames required for the stiffened shells are sized by the Shanley criterion. This criterion is based on the premise that, to a first-order approximation, the frames act as elastic supports for the wide column.⁷

There are a variety of structural geometries available for the fuselage. There is a simply stiffened shell concept using longitudinal frames. There are three concepts with Z-stiffened shells and longitudinal frames; one with structural material proportioned to give minimum weight in buckling, one with buckling efficiency compromised to give lighter weight in minimum gage, and one a buckling-pressure compromise. Similarly, there are three truss-core sandwich designs, two for minimal weight in buckling with and without frames, and one a buckling-minimum gage compromise.

It is assumed that the structural materials exhibit elastoplastic behavior. Further, to account for the effects of creep, fatigue, stress-corrosion, thermal cycling and thermal stresses, options are available to scale the material properties of strength and Young's modulus of elasticity. In the numerical results of this study, all materials were considered elastic and the full room-temperature material properties were used.

Composite materials can be modeled with PDCYL by assuming them to consist of orthotropic lamina formed into quasi-isotropic (two-dimensionally, or planar, isotropic) laminates. Each of the lamina is assumed to be composed of filaments placed unidirectionally in a matrix material. Such a laminate has been found to give very nearly minimum weight for typical aircraft structures.

Wing

The wing structure is a multi-web box beam designed by spanwise bending and shear. The wing-fuselage carrythrough structure, defined by the wing-fuselage intersection, carries the spanwise bending, shear, and torsion loads introduced by the outboard portion of the wing.

The load case used for the wing weight analysis is the quasi-static pull-up maneuver. The applied loads to the wing include the distributed lift and inertia forces, and the point loads of landing gear and propulsion, if placed on the wing. Fuel may also be stored in the wing, which will relieve bending loads during the pull-up maneuver.

The wing weight analysis proceeds in a similar fashion to that of the fuselage. The weight of the structural box is determined by calculating the minimum amount of material required to satisfy static buckling and strength requirements at a series of spanwise stations. The covers of the multi-web box are sized by buckling due to local instability and the webs by flexure-induced crushing. Required shear material is computed independently of buckling material. Aeroelastic effects are not accounted for directly, although an approximation of the magnitude of the tip deflection during the pull-up maneuver is made. For the carrythrough structure, buckling, shear, and torsion material are computed independently and summed.

As for the fuselage, there are a variety of structural geometries available. There are a total of six structural concepts, three with unstiffened covers and three with trussstiffened covers. Both cover configurations use webs that are either Z-stiffened, unflanged, or trusses.

GEOMETRY

Fuselage

The fuselage is assumed to be composed of a nose section, an optional cylindrical midsection, and a tail section. The gross density and fineness ratio are defined as

$$\rho_B = \frac{W_B}{V_B} \tag{1}$$

$$R_{fin} = \frac{l_B}{D} \tag{2}$$

where W_B is the fuselage weight (W_B = gross takeoff weight excluding the summed weight of the wing, tails, wingmounted landing gear, wing-mounted propulsion, and fuel if stored in the wing), V_B is the total fuselage volume, l_B is the fuselage length, and D is the maximum fuselage diameter. The fuselage outline is defined by two power-law bodies of revolution placed back-to-back, with an optional cylindrical midsection between them (Fig. 1). (For the present study, all eight transports used for validation of the analysis used the optional cylindrical midsection.)

The horizontal tail is placed according to its quarter chord location as a fraction of the fuselage length.

Propulsion may be either mounted on the fuselage or placed on the wing. In the case of fuselage mounted propulsion, the starting and ending positions of the propulsion unit are again calculated from their respective fractions of fuselage length

Similarly, the nose landing gear is placed on the fuselage as a fraction of vehicle length; the main gear, on the other hand, may be placed either on the fuselage as a single unit, also as a fraction of fuselage length, or on the wing in multiple units.

Wing

The lifting planforms are assumed to be tapered, swept wings with straight leading and trailing edges. The planform shape is trapezoidal as the root chord and tip chord are parallel. The wing is placed on the fuselage according to the location of the leading edge of its root chord, determined as a fraction of the fuselage length (Fig. 2). It is assumed that specified portions of the streamwise (aerodynamic) chord are required for controls and high lift devices, leaving the remainder for the structural wing box. The intersection of this structural box with the fuselage contours determines the location of the rectangular carrythrough structure. The width of the carrythrough structure is defined by the corresponding fuselage diameter.



Fig. 1 The body configuration.

For the transports in the present study, all the fuel is carried within the wing structure. An option is also available to carry the fuel entirely within the fuselage, negating any bending relief in the wing.

LOADS

Fuselage

Fuselage loading is determined on a station-by-station basis along the length of the vehicle. Three types of fuselage loads are considered—longitudinal acceleration, tank pressure, and bending moment. In the present study, all three load types are assumed to occur simultaneously to determine maximum compressive and tensile loads at the outer shell fibers at each station.

Bending loads applied to the vehicle fuselage are obtained by simulating vehicle pitch-plane motion during a quasi-static pull-up maneuver; a landing; and movement over a runway bump. Simplified vehicle loading models are used where it is assumed that: (1) fuselage lift forces (nominally zero for subsonic transports) are distributed uniformly over the fuselage plan area; (2) wing loading, determined independently, is transferred by a couple of vertical force and torque through the wing carrythrough structure; (3) fuselage weight is distributed uniformly over fuselage volume; (4) control surface forces and landing gear reactions are point loads; and (5) the propulsion system weight, if mounted on the fuselage, is uniformly distributed. A factor of safety (nominally 1.5) is applied to each load case. The aircraft weight for each case is selected as a fraction of gross takeoff weight. All fuselage lift forces are assumed to be linear functions of angle of attack. Longitudinal bending moments

are computed for each of the three loading cases and the envelope of the maximum values taken as the design loading condition. The bending moment computation is given in detail in Ref. 4 and will only be summarized here.

Considering first the pull-up maneuver loading, the motion is assumed to be a quasi-static pitch-plane pull-up of given normal load factor n (nominally 2.5 for transport aircraft). The vehicle is trimmed with the appropriate control surface (a horizontal tail for all eight transport used for validation in the present study), after which the angle of attack is calculated.

Landing loads are developed as the aircraft descends at a given vertical speed after which it impacts the ground; thereafter the main and nose landing gears are assumed to exert a constant, or optionally a $(1 - \cos(\omega t))$, force during its stroke until the aircraft comes to rest. The vehicle weight is set equal to the nominal landing weight. Wing lift as a fraction of landing weight is specified, which reduces the effective load the landing gear carries. Likewise, the portion of total vehicle load the main gear carries is specified. No pitch-plane motion is considered during the landing.

Runway bump loads are handled by inputting the bump load factor into the landing gear. Bump load factor is applied according to Ref. 8. This simulates the vehicle running over a bump during taxi. In a similar fashion to the landing, the wing lift as a fraction of gross takeoff weight is specified, as is the portion of effective load input through the main gear. No pitch-plane motion is considered during the bump.





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Wing

For the wing, only a quasi-static pull-up maneuver condition at load factor n is considered for determining loads. At each spanwise station along the quarter chord, from the wingtip to the wing-fuselage intersection, the lift load, center of pressure, inertia load, center of gravity, shear force, and bending moment are computed. For the inertia load, it is assumed that the fuel weight is distributed uniformly with respect to the wing volume.

There is an option for either a trapezoidal or a Schrenk⁹ lift load distribution along the wingspan; the trapezoidal distribution represents a uniform lift over the wing area (which has a trapezoidal planform) while the Schrenk distribution is an average of the trapezoidal distribution with an elliptical distribution, where the lift is zero at the wingtip and maximum at the wing-fuselage intersection. Prandtl has shown that a true elliptical lift load distribution will have a minimum induced drag, but a combination of the elliptical and trapezoidal distributions will give a better representation of actual aircraft loading.⁹

STRUCTURAL ANALYSIS

Fuselage

Weight estimating relationships are now developed for the load-carrying fuselage structure. In addition, the volume taken up by the fuselage structure is also determined.

Considering first the circular shell, the stress resultants in the axial direction caused by longitudinal bending, axial acceleration, and pressure at a fuselage station x are

$$N_{xB} = \frac{Mr}{l'_{y}} \tag{3}$$

$$N_{x_A} = \frac{N_x W_S}{P} \tag{4}$$

$$N_{xp} = \frac{AP_g}{P} \tag{5}$$

respectively, where r = D/2 is the fuselage radius, $A = \pi r^2$ is the fuselage cross-sectional area, and $P = 2\pi r$ is the fuselage perimeter. In EQ (3), $I'_y = \pi r^3$ is the moment of inertia of the shell divided by the shell thickness. In EQ (4), for the case of fuselage-mounted propulsion, W_s is the portion of vehicle weight ahead of station x if x is ahead of the inlet entrance, or the portion of vehicle weight behind x if x is behind the nozzle exit. In EQ (5), P_g is the limit gage pressure differential for the passenger compartment during cruise. The total tension stress resultant is then

$$N_{x}^{+} = N_{xR} + N_{xP} \tag{6}$$

if x is ahead of the nozzle exit, and

$$N_{x}^{+} = N_{xB} + N_{xP} + N_{xA} \tag{7}$$

if x is behind it. Similarly, the total compressive stress resultant is

$$N_{x}^{-} = N_{xB} + N_{xA} - \begin{cases} 0, & \text{if not pressure stabilized} \\ N_{xP}, & \text{if stabilized} \end{cases}$$
(8)

if x is ahead of the inlet entrance, and

$$N_{x}^{-} = N_{xB} - \begin{cases} 0, & \text{if not pressure stabilized} \\ N_{xP}, & \text{if stabilized} \end{cases}$$
(9)

if x is behind it. These relations are based on the premise that acceleration loads never decrease stress resultants, but pressure loads may relieve stress, if pressure stabilization is chosen as an option. The stress resultant in the hoop direction is

$$N_y = r P_g \dot{K_P} \tag{10}$$

where K_p accounts for the fact that not all of the shell material (for example, the core material in sandwich designs) is available for resisting hoop stress.

The equivalent isotropic thicknesses of the shell are given by

$$\bar{t}_{S_C} = \frac{N_x^-}{F_{CY}} \tag{11}$$

$$\bar{\iota}_{S_T} = \frac{1}{F_{\iota u}} \max \left(N_x^+, N_y \right) \tag{12}$$

$$\bar{i}_{S_G} = K_{mg} t_{mg} \tag{13}$$

for designs limited by compressive yield strength (F_{Cy}) , ultimate tensile strength (F_{lu}) , and minimum gage, respectively. In EQ (13), t_{mg} is a specified minimum material thickness and K_{mg} is a parameter relating \bar{t}_{SG} to t_{mg} which depends on the shell geometry.

A fourth thickness that must be considered is that for buckling critical designs, \bar{i}_{S_B} , which will now be developed. The nominal vehicles of this study have integrally stiffened shells stabilized by ring frames. In the buckling analysis of these structures, the shell is analyzed as a wide column and the frames are sized by the Shanley criteria.⁷ Expressions are derived for the equivalent isotropic thickness of the shell required to preclude buckling, \bar{i}_{S_B} , and for the *smeared* equivalent isotropic thickness of the ring frames required to preclude general instability, \bar{i}_F . The analysis will be restricted to the case of cylindrical shells. The major assumptions are that the structural shell behaves as an Euler beam and that all structural materials behave elastically.

For the stiffened shell with frames concept, the common procedure of assuming the shell to be a wide column is adopted. If the frame spacing is defined as d and Young's

modulus of the shell material is defined as E, the buckling equation is then

$$\frac{N_x^-}{dE} = \varepsilon \left(\frac{\bar{t}_{S_B}}{d}\right)^2 \tag{14}$$

or, solving for \bar{I}_{SR}

$$\bar{\iota}_{S_B} = \sqrt{\frac{N_x^- d}{E\varepsilon}}$$
(15)

Fuselage structural geometry concepts are presented in Table 1; values of the shell efficiency ε for the various structural concepts are given in Table 2. The structural shell geometries available are simply stiffened, Z-stiffened, and truss-core sandwich. We next size the frames to prevent general instability failure. The Shanley criterion is based on the premise that the frames act as elastic supports for the wide column; this criterion gives the smeared equivalent thickness of the frames as

$$\tilde{\iota}_{F_B} = 2r^2 \sqrt{\frac{\pi C_F N_x^-}{K_{FI} d^3 E_F}}$$
(16)

where C_F is Shanley's constant, K_{F1} is a frame geometry parameter, and E_F is Young's modulus for the frame material. (See Ref. 3 for a discussion of the applicability of this criterion and for a detailed derivation of the equations presented here.) If the structure is buckling critical, the total thickness is

$$\bar{t} = \bar{t}_{S_R} + \bar{t}_{F_R} \tag{17}$$

Minimizing \overline{i} with respect to d results in

$$\bar{\iota} = \frac{4}{27^{1/4}} \left(\frac{\pi C_F}{K_{F1} \varepsilon^3 E_F E^3} \right)^{\frac{1}{8}} \left(\frac{2r^2 \rho_F (N_x^-)^2}{\rho} \right)^{\frac{1}{4}}$$
(18)
$$\bar{\iota}_{S_B} = \frac{3}{4} \bar{\iota}$$
(19)
$$\bar{\iota}_{F_B} = \frac{1}{4} \bar{\iota}$$
(20)
$$d = \left(6r^2 \frac{\rho_F}{\rho} \sqrt{\frac{\pi C_F \varepsilon E}{K_{F1} E_F}} \right)^{\frac{1}{2}}$$
(21)

where ρ_F is the density of the frame material and ρ is the density of the shell material, so that the shell is three times as heavy as the frames.

Frameless sandwich shell concepts may also be used. For these concepts, it is assumed that the elliptical shell buckles at the load determined by the maximum compressive stress resultant N_x^- on the cylinder. The buckling equation for these frameless sandwich shell concepts is

$$\frac{N_x^-}{rE} = \varepsilon \left(\frac{\bar{\iota}_{SB}}{r}\right)^m \tag{22}$$

where *m* is the buckling equation exponent. Or, solving for I_{S_R}

$$\bar{\iota}_{S_B} = r \left(\frac{N_x^-}{rE\varepsilon} \right)^{\frac{1}{m}}$$
(23)

This equation is based on small deflection theory, which seems reasonable for sandwich cylindrical shells, although it is known to be inaccurate for monocoque cylinders. Values of mand ε may be found, for example in Refs. 10 and 11 for many shell geometries. Table 2 gives values for sandwich structural concepts available in PDCYL, numbers 8 and 9, both of which are truss-core sandwich. The quantities N_x^- , r, and consequently \tilde{i}_{S_R} , will vary with fuselage station dimension x.

At each fuselage station x, the shell must satisfy all failure criteria and meet all geometric constraints. Thus, the shell thickness is selected according to compression, tension, minimum gage, and buckling criteria, or

$$\bar{i}_{S} = \max\left(\bar{i}_{SC}, \bar{i}_{ST}, \bar{i}_{SG}, \bar{i}_{SB}\right)$$
(24)

If $\bar{t}_S = \bar{t}_{SB}$, the structure is buckling critical and the equivalent isotropic thickness of the frames, \bar{t}_F , is computed from EQ (20). If $\bar{t}_S > \bar{t}_{SB}$, the structure is not buckling critical at the optimum frame sizing and the frames are resized to make $\bar{t}_S = \bar{t}_{SB}$. Specifically, a new frame spacing is computed from EQ (15) as

$$d = \frac{E\varepsilon \bar{I}_S^2}{N_r}$$
(25)

and this value is used in EQ (16) to determine \bar{t}_F .

The total thickness of the fuselage structure is then given by the summation of the smeared weights of the shell and the frames

$$\vec{t}_B = \vec{t}_S + \vec{t}_F \tag{26}$$

The shell gage thickness may be computed from

 $\bar{i}_g = \bar{i}_S / K_{mg}$. The *ideal* fuselage structural weight is obtained by summation over the vehicle length

$$W_{I} = 2\pi \sum \left(\rho \bar{i}_{S_{i}} + \rho_{F} \bar{i}_{F_{i}}\right) r_{i} \Delta x_{i}$$
⁽²⁷⁾

where the quantities subscripted i depend on x.

Since the preceding analysis gives only the ideal weight, W_I , the *nonoptimum* weight, W_{NO} (including fasteners, cutouts, surface attachments, uniform gage penalties,

KCON sets concept number	
2	Simply stiffened shell, frames, sized for minimum weight in buckling
3	Z-stiffened shell, frames, best buckling
4	Z-stiffened shell, frames, buckling-minimum gage compromise
5	Z-stiffened shell, frames, buckling-pressure compromise
6	Truss-core sandwich, frames, best buckling
8	Truss-core sandwich, no frames, best buckling
9	Truss-core sandwich, no frames, buckling-minimum gage-pressure compromise

Structural concept (KCON)	m	3	K _{mg}	Kp	K _{th}	
2	2	0.656	2.463	2.463	9.0	
3	2	0.911	2.475	2.475	0.0	
4	2	0.760	2.039	1.835	0.0	
5	2	0.760	2.628	1.576	0.0	
6	2	0.605	4.310	3.965	0.459	
8	1.667	0.4423	4.820	3.132	0.405	
9	1.667	0.3615	3.413	3.413	0.320	

Table 2 Fuselage structural geometry parameters

Table 1 Fuselage structural geometry concepts

manufacturing constraints, etc.) has yet to be determined. The method used will be explained in a later section.

Wing

Using the geometry and loads applied to the wing developed above, the structural dimensions and weight of the structural box may now be calculated. The wing structure is assumed to be a rectangular multi-web box beam with the webs running in the direction of the structural semispan. Reference 10 indicates that the critical instability mode for multi-web box beams is simultaneous buckling of the covers due to local instability and of the webs due to flexure induced crushing. This reference gives the solidity (ratio of volume of structural material to total wing box volume) of the least weight multi-web box beams as

$$\Sigma = \varepsilon \left(\frac{M}{Z_S t^2 E}\right)^e \tag{28}$$

where ε and e depend on the cover and web geometries (Table 3), M is the applied moment, t is the thickness, E is the elastic modulus, and Z_S is obtained from Ref. 10. The solidity is therefore

$$\Sigma = \frac{W'_{BEND}(y)}{\rho Z_S t}$$
(29)

where W_{BEND} is the weight of bending material per unit span and ρ is the material density. W_{BEND} is computed from EQS (28) and (29). The weight per unit span of the shear material is

$$W'_{SHEAR}(y) = \frac{\rho F_S}{\sigma_S}$$
(30)

where F_S is the applied shear load and σ_S is the allowable shear stress. The optimum web spacing is computed from²

$$d_{W} = t \left[\frac{(1 - 2e_{C})}{(1 - e_{C})\sqrt{2\varepsilon_{W}}} \left(\frac{M}{Z_{S}t^{2}E} \right)^{\frac{2e_{C} - 3}{2e_{C}}} \varepsilon_{C}^{\frac{3}{2e_{C}}} \right]^{\frac{2e_{C}}{4e_{C} - 3}}$$
(31)

where subscripts W and C refer to webs and covers, respectively. The equivalent isotropic thicknesses of the covers and webs are

Table 3 Wing stru	ictural coefficients	and expone	nts
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	Waha		,	3	ec	Ew	Kgc	K _{gw}
Covers	webs	<u>د</u>				0.605	1 000	0.407
Unstiffened	Truss	2.25	0.556	3.62	3	0.005	1.000	0.505
Unstitlened	Linflanged	2.21	0.556	3.62	3	0.656	1.000	0.505
Unsuiteneu	Uninangee	2.05	0 556	3.62	3	0.911	1.000	0.405
Unstiffened	Z-suitened	2.05	0.550				0.546	0.407
T	Tenes	2.44	0.600	1.108	2	0.605	0.540	0.407
Truss	Truss	2.40	0.600	1.108	2	0.656	0.546	0.505
Truss	Unitangeo	2.40	0.000	1 109	2	0.911	0.546	0.405
Truss	Z-stiffened	2.25	0.600	1.100				

$$\bar{\iota}_C - d_W \left(\frac{M}{Z_{S} \iota E \varepsilon_C d_W}\right)^{\frac{1}{\epsilon_C}}$$
(32)

$$\bar{\iota}_{W} = \iota \sqrt{\left(\frac{M}{Z_{S}\iota^{2}E}\right)^{2-\frac{1}{e_{C}}}} \left(\frac{\varepsilon_{C}d_{W}}{\iota}\right)^{\frac{1}{e_{C}}} \left(\frac{2}{\varepsilon_{W}}\right)$$
(33)

respectively, and the gage thicknesses are

$$t_{BC} = K_{BC} \bar{t}_C \tag{34}$$

$$I_{gW} = K_{gW} \bar{I}_W \tag{35}$$

Values of ε , e, ε_C , E_C , ε_W , K_{g_W} , and K_{g_C} are found in Table 3 for various structural concepts.¹⁰ If the wing structural semispan is divided into N equal Tength segments, the total ideal weight of the wing box structure is

$$W_{BOX} = \frac{2b_S}{N} \sum_{i=1}^{N} \left(W'_{BEND_i} + W'_{SHEAR_i} \right)$$
(36)

The wing carrythrough structure consists of torsion material in addition to bending and shear material. The torsion material is required to resist the twist induced due to the sweep of the wing. The bending material is computed in a similar manner as that of the box except that only the longitudinal component of the bending moment contributes. Letting $t_0 = t(y = 0)$ and $M_0 = M(y = 0)$,

$$\Sigma_C = \epsilon \left(\frac{M_0 \cos(\Lambda_S)}{\iota_0^2 C_{SR} E} \right)^e$$
(37)

The weight of the bending material is then

$$W_{BEND_C} = \rho \Sigma_C C_{SR^{i_0} WC}$$
(38)

where w_C is the width of the carrythrough structure. (When the wing-fuselage intersection occurs entirely within the cylindrical midsection, as is the case with all eight transport used for validation in the present study, $w_C = D$.) The quantities d_W , t_W , and t_C are computed in the same manner as for the box. The weight of the shear material is

$$W_{SHEAR_C} = \rho \frac{F_{S_0}}{\sigma_S} w_C \tag{39}$$

where
$$F_{S_0} = F_S(0)$$
.

The torque on the carrythrough structure is

$$T = M_0 \sin(\Lambda_S) \tag{40}$$

and the weight of the torsion material is then

$$W_{TORSION_C} = \frac{\rho T (t_0 + C_{SR}) w_C}{t_0 C_{SR} \sigma_S}$$
(41)

Finally, the ideal weight of the carrythrough structure is computed from a summation of the bending shear and torsion material, or

$$W_C = W_{BEND_C} + W_{SHEAR_C} + W_{TORSION_C}$$
(42)

(12)

As in the case of the fuselage structural weight, nonoptimum weight must be added to the ideal weight to obtain the true wing structural weight. The method used will be discussed below.

REGRESSION ANALYSIS

Overview

Using fuselage and wing weight statements of eight subsonic transports, a relation between the calculated loadbearing structure weights obtained through PDCYL and the actual load-bearing structure weights, primary structure weights, and total weights is determined using statistical analysis techniques. A basic application which is first described is linear regression, wherein the estimated weights of the aircraft are related to the weights calculated by PDCYL with a straight line, y = mx + b, where y is the value of the estimated weight, m is the slope of the line, x is the value obtained through PDCYL, and b is the y-intercept. This line is termed a regression line, and is found by using the method of least squares, in which the sum of the squares of the residual

errors between actual data points and the corresponding points on the regression line is minimized. Effectively, a straight line is drawn through a set of ordered pairs of data (in this case eight weights obtained through PDCYL and the corresponding actual weights) so that the aggregate deviation of the actual weights above or below this line is minimized. The estimated weight is therefore dependent upon the independent PDCYL weight.

Of key importance is the degree of accuracy to which the prediction techniques are able to estimate actual aircraft weight. A measure of this accuracy, the correlation coefficient, denoted R, represents the reduction in residual error due to the regression technique. R is defined as

$$R = \sqrt{\frac{E_t - E_r}{E_r}} \tag{43}$$

where E_t and E_r refer to the residual errors associated with the regression before and after analysis is performed, respectively. A value of R = 1 denotes a perfect fit of the data with the regression line. Conversely, a value of R = 0 denotes no improvement in the data fit due to regression analysis.

There are two basic forms of equations which are implemented in this study. The first is of the form

$$y_{est} = mx_{calc} \tag{44}$$

The second general form is

$$y_{est} = mx_{calc}^{a} \tag{45}$$

Fuselage

The analysis above is used to develop a relationship between weight calculated by PDCYL and actual wing and fuselage weights. The data were obtained from detailed weight breakdowns of eight transport aircraft¹²⁻¹⁶ and are shown in Table 4 for the fuselage. Because the theory used in the PDCYL analysis only predicts the load-carrying structure of the aircraft components, a correlation between the predicted weight and the actual load-carrying structural weight and primary weight, as well as the total weight of the fuselage, was made.

Structural weight consists of all load-carrying members including bulkheads and frames, minor frames, covering, covering stiffeners, and longerons. For the linear curve-fit, the resulting regression equation is

$$W_{actual} = 1.3503 W_{calc}$$
 $R = 0.9946$ (46)

This shows that the *nonoptimum* factor for fuselage structure is 1.3503; in other words, the calculated weight must be increased by about 35 percent to get the actual structural weight. For the alternative power-intercept curve fitting analysis, the resulting load-carrying regression equation is

$$W_{actual} = 1.1304 W_{calc}^{1.0179} \quad R = 0.9946$$
 (47)

To use either of these equations to estimate total fuselage weight, nonstructural weight items must be estimated independently and added to the structural weight.

Primary weight consists of all load-carrying members as well as any secondary structural items such as joints fasteners, keel beam, fail-safe straps, flooring, flooring structural supplies, and pressure web. It also includes the lavatory structure, galley support, partitions, shear ties, tie rods, structural firewall, torque boxes, and attachment fittings. The linear curve fit for this weight yields the following primary regression equation

$$W_{actual} = 1.8872 W_{calc}$$
 $R = 0.9917$ (48)

The primary power-intercept regression equation is

$$W_{actual} = 1.6399 W_{calc}^{1.0141} \quad R = 0.9917$$
 (49)

Table 4	Fuselage	weight	hreakdowns	for eight	transport	aircraft
	LUSCIAKC	ACIATI	DICALUOWIIS	лог сіхш	ualisiaau	ancian

		Weight, lb					
Aircraft	PDCYL	Load-carrying structure	Primary structure	Total structure			
B-720	6545	9013	13336	19383			
B-727	5888	8790	12424	17586			
B-737	3428	5089	7435	11831			
B-747	28039	39936	55207	72659			
DC-8	9527	13312	18584	24886			
MD-11	20915	25970	34999	54 936			
MD-83	7443	9410	11880	16432			
L-1011	21608	28352	41804	52329			

The total fuselage weight accounts for all members of the body, including the structural weight and primary weight. It does not include passenger accommodations, such as seats, lavatories, kitchens, stowage, and lighting; the electrical system; flight and navigation systems; alighting gear; fuel and propulsion systems; hydraulic and pneumatic systems; the communication system; cargo accommodations; flight deck accommodations; air conditioning equipment; the auxiliary power system; and emergency systems. Linear regression results in the following total fuselage weight equation

$$W_{actual} = 2.5686 W_{calc}$$
 $R = 0.9944$ (50)

This shows that the nonoptimum factor for the total fuselage weight is 2.5686; in other words, the fuselage structure weight estimated by PDCYL must be increased by about 157 percent to get the actual total fuselage weight. This nonoptimum factor is used to compare fuselage structure weight estimates from PDCYL with total fuselage weight estimates from the Sanders and the Air Force equations used by ACSYNT.

The total fuselage weight power-intercept regression equation is

$$W_{actual} = 3.9089 W_{calc}^{0.9578} R = 0.9949$$
 (51)

Plots of actual fuselage component weight versus PDCYL-calculated weight, as well as the corresponding linear regressions, are shown in Figs. 3-5. Wing

The same analysis was performed on the wing weight for the sample aircraft and is shown in Table 5. The wing box, or load-carrying structure, consists of spar caps, interspar coverings, spanwise stiffeners, spar webs, spar stiffeners, and interspar ribs. The wing box linear regression equation is

$$W_{actual} = 0.9843 W_{calc}$$
 $R = 0.9898$ (52)

so that the nonoptimum factor is 0.9843. Power-intercept regression results in

$$W_{actual} = 1.3342 W_{calc}^{0.9701}$$
 $R = 0.9902$ (53)

Wing primary structural weight includes all wing box items in addition to auxiliary spar caps and spar webs, joints and fasteners, landing gear support beam, leading and trailing edges, tips, structural firewall, bulkheads, jacket fittings, terminal fittings, and attachments. Linear regression results in

$$W_{actual} = 1.3442 W_{calc}$$
 $R = 0.9958$ (54)

Power-intercept regression yields

$$W_{actual} = 2.1926 W_{calc}^{0.9534}$$
 $R = 0.9969$ (55)



Fig. 3 Fuselage load-carrying structure and linear regression.



Fig. 4 Fuselage primary structure and linear regression.



Fig. 5 Fuselage total structure and linear regression.

	Weight, lb					
	PDCVI	Load-carrying structure	Primary structure	Total structure		
Aircraft	FLCTL		18914	23528		
B-720	13962	11747	10210	17860		
- 	8688	8791	12388	10687		
D-/2/	6717	5414	7671	10087		
B- 737	5/1/	60205	68761	88202		
B-747	52950	30393				
	22080	19130	27924	35330		
DC-0	22000		47614	62985		
MD-11	33617	35157	47017	15830		
MD 83	6953	8720	11553	1.0.00		
1410-00	0710		36101	46233		
L-1011	25034	28355	.0101			

- -

Table 5. Wing weight breakdowns for eight transport aircraft

The total wing weight includes wing box and primary weight items in addition to high-lift devices, control surfaces, and access items. It does not include the propulsion system, fuel system, and thrust reversers; the electrical system; alighting gear; hydraulic and pneumatic systems; anti-icing devices; and emergency systems. The resulting total weight linear regression equation is

$$W_{actual} = 1.7372 W_{calc}$$
 $R = 0.9925$ (56)

This shows that the nonoptimum factor for the total wing weight is 1.7372; in other words, the wing box weight estimated by PDCYL must be increased by about 74 percent to get the actual total wing weight. This nonoptimum factor is used to compare wing box weight estimates from PDCYL with total wing weight estimates from the Sanders and the Air Force equations used by ACSYNT.

The power-intercept equation for total wing weight is

$$W_{actual} = 3.7464 W_{calc}^{0.9268} R = 0.9946$$
 (57)

Plots of actual wing component weight versus PDCYLcalculated weight, as well as the corresponding linear regressions, are shown in Figs. 6-8.

Discussion

Both fuselage and wing weight linear and power regressions give excellent correlation with the respective weights of existing aircraft, as evidenced by the high values of the correlation coefficient, R. It should be noted that even though the power-based regressions give correlations equal to or better than the linear regressions their factors may vary distinctly from the linear cases. This is due to their powers not equaling unity.

Because estimates of non-load-bearing primary structure are generally not available at the conceptual design stage, and because nonprimary structure is probably not well estimated by a nonoptimum factor, EQS (48) and (54) are recommended for estimating the primary structural weights of the respective transport fuselage and wing structures (Figs. 4 and 7).

A comparison may be made between weight estimates from weight estimating relationships currently used by ACSYNT, PDCYL output, and actual aircraft component weights. Figure 9(a) shows a comparison between fuselage weight estimated from the Sanders equation, the Air Force equation, and PDCYL output with the actual fuselage weight of the 747-21P. Figure 9(b) shows a similar comparison for the wing weight.



Fig. 6 Wing load-carrying structure and linear regression.



Fig. 7 Wing primary structure and linear regression.



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Fig. 8 Wing total structure and linear regression.



Fig. 9(a) Fuselage weight estimation comparison for 747-21P.



Fig. 9(b) Wing weight estimation comparison for 747-21P.

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Designing Composite Transport Aircraft

by

L. Hornberger, M. Ardema, and F. Dickerson

DESIGNING COMPOSITE TRANSPORT AIRCRAFT by Mark Ardema, Frank Dickerson and Lee Hornberger Mechanical Engineering Department Santa Clara University

ADDING COMPOSITES TO PDCYL

Light weight materials such as fiber reinforced plastics (composites) and bonded honeycomb sandwiches have become more and more common in airplanes in the last two decades (1). Designers value the unique properties of these materials, particularly their high stiffness to weight ratios. They must, however, balance these assets against the additional cost of these materials and their manufacture. To aid designers with this analysis, a composites subroutine has been added to the PDCYL structures weight calculation code. This subroutine sizes the thickness of a particular composite necessary to withstand the required aircraft loads, and provides this information to PDCYL which calculates the resultant weight of the aircraft.

TYPICAL AIRCRAFT COMPOSITES

The selection and use of composites on transport aircraft is an evolving process. A variety of composites have been tested in both military and commercial aircraft in the last 25 years (1). These composites typically consist of a strong, stiff fiber such as glass, graphite or kevlar, and a protective, adhering, inexpensive plastic matrix such as polyester or epoxy.

Glass fibers embedded in a polyester matrix have been the dominate composite for military and civil aircraft in the past. Currently, the aircraft industry prefers the stiffer and higher temperature composites made from carbon fiber in an epoxy matrix. However, the grade of carbon fiber and epoxy seems to change from year to year and from airplane manufacturer to airplane manufacturer. The current favored carbon fibers are AS4 (Hercules/Hexcel), IM6 and IM7 (Hercules/Hexcel). The AS4 is an economical, high-strength carbon fiber and the IM6 &7 are high-modulus expensive fibers. These three carbon fibers have been used on military aircraft and in research, but are not on commercial vehicles. The T-800 fiber (a Toray equivalent to IM7) has recently been used in some commercial applications (1-6).

Epoxies, particularly the 350°F curing systems, are the least expensive high temperature options for matrix materials. Several epoxy systems have been developed and tested for use with specific fibers. There is a current trend to use rubber modified epoxies such as 8552 and 3900 to increase the toughness of the composite system and its resistance to impact. Fiber-resin combinations currently in use by airplane designers and researchers are:

AS4/938 (ICI Fiberite) -Boeing Advanced Composites Program Door Panel(2)

AS4/8552 (Hexcel/Hercules), -Boeing Adv. Comp Fuselage (6-7) AS4/8551 (Hexcel/Hercules) (6)

AS4/3501-6 (Hercules) -McDonnell Douglas Adv. Technology Composite Wing program (8)

AS4/3502 (Hercules) Military Aircraft (6)

COMPOSITE STRUCTURAL ANALYSIS

Composite materials were originally added to the options in the PDCYL program in 1995. This was done by simulating these materials by homogeneous structures with uniform mechanical properties (strength and modulus of elasticity) in every direction. This approach limits the code to evaluation of only the simplest and weakest type of composites called random mat¹. Random mat composites are made by stacking the reinforcing fiber in all direction throughout the thickness of the material. In this type of composite the elastic properties and strength of the layup are roughly the same in every direction but the fiber density and reinforcement is low in any specific direction.

Random mat composites are not favored by aircraft designers because of their low strength to weight ratios. The preferred type of composite for these applications are ones in which the properties of the material are customized to meet the specific directions and magnitudes of the structural loads. This yields the minimum weight composite for the job. To accomplish this, composite designers specify a layup pattern for a composite laminate relative to a major axis of loading.

¹ See Appendix A for definition of composite terms

A typical composite laminate is made of a stack of 4-16 plys. A ply is a single layer of parallel reinforcing fibers embedded in a partially cured matrix of plastic. The location of each ply in the stack is defined relative to the angle its fibers makes with a major axis, such as the x-axis. For instance. a 0/90/90/45/0 layup is one in which the fibers of the outer and inner layers are parallel to the x-axis, the next two plies have fibers perpendicular to this axis and the fibers of the third layer are at an angle 45° clockwise to the x axis. This type of composite would have reinforcing fibers to sustain tensile and compressive loads in the x and y directions but would be weakest in the 45° direction. Composites walls for structural parts such as aircraft are often made from stacks of these laminates.

Analysis of a multilayer stack is more complex than that of homogeneous materials such as aluminum or random mat and requires the use of a macromechanics approach to determine elastic properties and strength. The macromechanics approach used in the COMPOS part of the PDCYL code is that presented in most textbooks on composite design (9-11). In this approach the stiffness of a particular laminate is calculated by summing the contributions of each layer (ply) in the stack to the stiffness of the laminate in a particular direction. The composite stiffness in each major direction is then used to calculate the net strain of the composite in that direction due to the applied loads. From the net strain, the strain on each layer (ply) parallel and transverse to its fiber is derived. The resulting strains are then compared to the failure strains of the ply material and from this the potential for the failure of the stack is determined. The details of implementing this approach in PDCYL are described in the following section describing the COMPOS (composites) code addition.

COMPOS CODE

COMPOS is a section of code which has been added to PDCYL program to calculate the minimum laminate thickness required to withstand the forces imposed at each section of the airplane.

Assumptions within COMPOS

The laminate is symmetric and orthotropic. (This type of layup is commonly used in aircraft design to minimize warpage of the layup).
Every ply in the stack is composed of the same resin- fiber material.
The stack is a minimum of 3 plys. (A ply is usually .003-.007 inches thick depending on the material.)

• The modulus of the material is the same in compression and tension. (if the compression modulus is different than its tensile modulus, the smaller of the two values is selected for all calculations.)

Failure of the composite laminate occurs when any single ply fails.
Failure of a ply occurs when it reaches the maximum strain transverse or parallel to the fiber direction in tension, compression or shear (11) Maximum strain theory is invoked in this analysis because it is currently believed to be the most predictive failure theory for composites (3,4,8).
The minimum gage thickness for the composite material is assumed to be the thickness of the initial laminate (a stack of plies).

•All loads are applied in the plane of the ply. This means that there are no z direction loads in tension, compression or shear.

• The buckling equations used in PDCYL to analyze the frames and stringers made from homogeneous materials apply to these heterogeneous materials. For buckling analysis the modulus of the laminate in the direction of load is used. This is a very course assumption and maybe somewhat optimistic for quasi isotropic composites manufactured with adhesive joints but seems highly unlikely for symmetric orthotropic laminates with heterogeneous properties. However, buckling analysis of complex composites structures is still in the developmental stage.(12)

Calculation Procedure

•Calculations for Compressive and Tensile Loads

Once the maximum tensile and/or compressive loads per unit width (Nx and Ny) at any given aircraft section are determined in the PDCYL code, they are transferred to the COMPOS subroutine. The effect of these normalized forces on the composite laminate strain is calculated using the following relationship for an orthotropic symmetric laminate (9) :

Where:

$$[\mathsf{N}] = [\mathsf{A}] \times [\varepsilon^0] \tag{1}$$

[N] = Matrix of forces on the composite section (Nx, Ny and Nxy)

[A] = Stiffness matrix of the composite

 $[\varepsilon^0]$ = strain matrix of the composite (ex, ey, exy)

The components of the stiffness matrix [A] are determined in the code through the following relationship (9):

$$A_{ij} = \sum n_{\kappa=1} (QB_{ij})_k (h_k)$$
(2)

Where:

 QB_{ij} = component of each ply's stiffness in the i and j's directions h = thickness of k ply k= ply number in the laminate

The stiffness contributions, QB values, of each ply are determined from the initial ply properties, E1, E2, v_{12} and the ply angles, θ , specified by the user in the input file for a particular laminate construction. (Here, the "1" direction is taken parallel to the fiber and the "2" direction transverse to the fiber).

Once the average laminate strain is determined from equation (1), this strain is then transferred to each ply and transformed into strain parallel and transverse to each fiber as well as shear strain. These strains are then divided by the mating failure strains for the material (supplied by the user in the input file) to determine the R value of the layup.

 $R_{ij} = alle_{ij} / e_{ij}$

Where:

alle_{ij}=allowable components of strain in principle ply direction e_{ii} = components of strain in principle ply directions

If the R value for all plys in all the principle directions is more than 1, the laminate thickness is adequate to support the load and is left unaltered. If R is less than one on any ply in any of the principle directions, the thickness of the laminate is increased by giving it the value of it initial thickness divided by R.

Calculations for Buckling

PDCYL currently determines critical buckling loads from the modulus of elasticity of the material. COMPOS calculates the modulus of the laminate in the direction parallel to the buckling force and passes this value back to PDCYL. As mentioned in the assumptions portion of this report, the buckling calculation of PDCYL may not be valid for composites as they were developed for isotropic materials. *Little research has been done on composites in buckling so the authors advise caution in interpreting this result particularly with non-isotropic layups.*

NON-OPTIMUM FACTORS

Unfortunately, few all composite planes have been built so it is difficult to find planes to use as checks for the composite section of the code (8). The all composite planes listed in the literature (8) are:

- Windecker Eagle in 1967 which was glass fiber reinforced
- Learfan in 1981 which used glass, carbon and kevlar fibers
- Piaggio Avanti in 1986 with carbon fiber parts
- · Beech Starship in 1986 with carbon fiber
- · Grob GF-200- all composite
- Slingsby T-3A Firefly -all composite

A literature search and personal interviews failed to turn up much information directly useful in determining non-optimum factors. (These factors are used to multiply the results of theoretical calculations to get weights of practical structures.)

One reference was found which had this type of data (12). In this reference, a theoretical analysis gave 8640 pounds as the weight of a composite wing box whereas the actual wing was estimated to weigh 11,284 pounds giving a non-optimum factor of 1.306. Using the non-optimum factors for aluminum structures (13) this number can be used to estimate non optimum factors for carbon fiber-epoxy structures. If it is assumed that the non-optimum factors for the fuselage primary structure increase in the same proportion as wing structure relative to aluminum, and that the increments for secondary structure and non-structural are the same for graphite-epoxy composites and aluminum, then the following non-optimum factors for the composite result:

	Primary Structure	Primary &Secondary Structure	Total Assembly
Fuselage	1.792	2.329	3.010
Wing	1.306	1.666	2.059

There are many composite components in commercial and military structure as well as some from research on advanced composites. It may be possible to compare these components to predictions of the code.

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APPENDIX 1 COMPOSITES TERMINOLOGY

Random Mat- equal fibers in every direction

<u>Balanced</u>- equal fibers in orthotropic directions yield a composite with identical properties in 2 principal directions.

<u>Symmetric</u>-A symmetric laminate is one in which for each ply above the center of the stack there is and identical one at an equal distance below the center. For instance, a 0/-45/90/90/-45/0 is a symmetric layup but a 0/-45/90/ 0/-45/90 is not.

<u>Quasi-Isotropic</u>- Layups which are designed to have only two independent elastic constants, the modulus of elasticity and Poisson's ratio. These materials have the same values in every inplane direction. To meet this criteria fiber (ply) layups must have the following conditions:

•Total number of plies must be 3 or more

•Individual plies must have identical stiffness [Q] matrices and thickness

•Layers must be oriented at "equal" angles (if total number of layers is n, than each layer is pi/n relative to the next). If the laminate is constructed from several groups of laminates, the condition must be satisfied for each laminate group Typical laminates which satisfy these rules : [0/60/-60], [0/45/-45/90]
Appendix C

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Weights of ASA 2150 With Aluminum and Graphite/Epoxy Fuselage

ALUMINUM

I

~ ~ Calling Module # Calling Module # TAKEOFF

0 = 0.3242445E+04 $W = 0.1489386E+00$	2 = 0.2259222E+03 SMN2 = 0.2034092E+0	0 = 0.000000E+00 SFC0 = 0.100000E+0		
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WFTO1 = 0.3242445E+04	CLS = 0.1769708E+01	TN2 = 0.0000000E+00	SFCAVE = 0.1000000E+01	
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ASA ZIJO Aluminum

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UNITW COVEF LB/F	0.790	0.790	0.790	0.790	0.790	0.790	0.790	0.790	0.790	0.790	0.790	0.825	0.922	1.017	1.112	1.204	1.293	1.380	1.465	1.546	1.625	1.701	1.774
WGAGE THICK IN	0.02000	0.02000	0.02000	0.02000	0.02000	0.02000	0.02000	0.02000	0.02000	0.02000	0.02000	0.02000	0.02000	0.02000	0.02000	0.02000	0.02000	0.02000	0.02000	0.02000	0.02000	0.02000	0.02000
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WEB THICK IN	0.03960	0.03960	0.03960	0.03960	0.03960	0.03960	0.03960	0.03960	0.03960	0.03960	0.03960	0.03960	0.03960	0.03960	0.03960	0.03960	0.03960	0.03960	0.03960	0.03960	0.03960	0.03960	0.03960
COVER THICK IN	0.0543	0.0543	0.0543	0.0543	0.0543	0.0543	0.0543	0.0543	0.0543	0.0543	0.0543	0.0567	0.0634	0.0700	0.0765	0.0828	0.0889	0.0949	0.1007	0.1064	0.1118	0.1170	0.1220
WEB SPACE IN	0.1830	0.3699	0.5195	0.6522	0.7751	0.8915	1.0032	1.1114	1.2169	1.3202	1.4217	1.5217	1.6204	1.7179	1.8146	1.9104	2.0054	2.0997	2.1934	2.2865	2.3790	2.4711	2.5626
BEND MOM FT-LBS	130.	1713.	5798.	12923.	23489.	37831.	56236.	78955.	106209.	138198.	175102.	217079.	264273.	316811.	374806.	438357.	507548.	582452.	663128.	749622.	841966.	940182.	1044280.
LENGTH PRIME FT	2.4504	2.6341	2.8179	3.0017	3.1855	3.3692	3.5530	3.7368	3.9206	4.1043	4.2881	4.4719	4.6557	4.8395	5.0232	5.2070	5.3908	5.5746	5.7583	5.9421	6.1259	6.3097	6.4934
LENGTH FT	2.4504	2.6341	2.8179	3.0017	3.1855	3.3692	3.5530	3.7368	3.9206	4.1043	4.2881	4.4719	4.6557	4.8395	5.0232	5.2070	5.3908	5.5746	5.7583	5.9421	6.1259	6.3097	6.4934
CHORD FT	5.4034	5.7636	6.1238	6.4840	6.8442	7.2044	7.5646	7.9249	8.2851	8.6453	9.0055	9.3657	9.7259	10.0861	10.4463	10.8065	11.1667	11.5269	11.8871	12.2473	12.6075	12.9677	13.3279
WING STATION FT	50.764	49.495	48.226	46.957	45.688	44.419	43.150	41.881	40.611	39.342	38.073	36.804	35.535	34.266	32.997	31.728	30.459	29.190	27.920	26.651	25.382	24.113	22.844

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346 18.0	1105 8.	.8825	8.8825	2894486	3.7066	0.1678	00650.0	2100.0			T// 0.	09/5
076 18.3	1707 9.	0663	9.0582	3059202	. 3.7871	0.1693	0.03960	0 0623	0 0200			00/0
807 18.7	309 9.	.2501	7.3616	3227873	. 3.8673	0.1706	0.03960	0.0628	0.0200	0 2.48	814 0.	5760
538 19.0	911 9.	4339	5.6651	3398988	. 3.9467	0.1718	0.03960	0.0632	0.0200	00 2.45	980 0.	5760
269 19.4	1513 9.	. 6177	3.9685	3571425	. 4.0251	0.1727	0.03960	0.0635	0.0200	10 2.51	112 0.	5760
9.61 000	3115 9.	.8014	2.2720	3743631	. 4.1021	0.1733	0.03960	0.0638	0.0200	0 2.52	204 0.	5760
1 CLINT	CLINTE	P LBOX	WBO	X TBO	ГN X	W WEBSB	TORK	ΓΨΟ	TRCOV			
FΤ	FT	ΕT	ΕŢ	FΤ		ΕŪ	FT-LRS	NI	IN			
3 49.955	59.861	06.6 1	58 11.9	334 2.8	92 5	0.3353	586064.	7 0.01	34 0.0438	-		
AR WE L	JEND BS	WWING LBS	зц	VSHBOX ,BS	WBDBOX LBS	WTOBOX LBS	WWBOX LBS	WWINGT	WPOD	DELTIP		
.50 2	532.25	9046	.48	31.93	1177.02	59.63	1268.58	10315.06	3272.93	3.849		
ROL AREA FT2. 14.99	STF	RUCTURE A FT2. 719.62	AREA	SPLA FT2 1450	ч · ·							
TS W	то м 2181.	VBOD 83574.	WWING 52804	WPROP . 654	. WTAIL 6. 2983	CG · 53.255	RG 0.000					
PROP VOL ETERS 9	UME DEN 348. 16.	4SITY 2789	CL1 58.915	FIN RAT 9.3665	LENGTH W 117.830 1	IDTH A	30D ASUR 14.9 3596.8	CLP1 0.00	CLP2 0.00			
AT ETERS 2	AIL CL 02. 115.	ד 47										
ETERS	WEIGHT 152181.	АLРНА 7.00	DEFLE -11.5	C LIF	TB LIFTW 2. 166974	LIFTT -14925	CLAQW 16.45	CLAQB S 0.02	TAMA 4.72	СGМ 53.3		
VER ETERS	SLFM 2.50	ALPHA 17.50	DEFLEC -28.75	LIFTB 330.	LIFTW 417435.	LIFTT -37312.				-		
Y 5 2.60	BEND MOM -0.3974E	IENT 1+03	WSAV(219.	I) BMBI 39	W BM	BL 1 2.	зми в 0.	MP 0.	BMT 0.	BMG 0.	MAX MOMI 0.3974E-	ENT + 03

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0.2581E+04	0.7710E+04	0.1676E+05	0.3060E+05	0.5006E+05	0.8179E+05	0.1240E+06	0.1739E+06	0.2323E+06	0.2998E+06	0.3771E+06	0.4646E+06	0.5629E+06	0.6718E+06	0.7915E+06	0.9218E+06	0.1063E+07	0.1215E+07	0.1374E+07	0.1547E+07	0.1731E+07	0.1925E+07	0.3277E+29	0.2343E+07	0.2611E+07	0.3071E+07	0.3595E+07	0.4021E+07	0.4188E+07	0.4006E+07	0.3778E+07	0.3557E+07	0.3345E+07	0.3140E+07	0.2943E+07	0.2754E+07	0.1274E+30	0.2397E+07	0.2229E+07	0.2067E+07	0.1912E+07	0.1763E+07	0.1620E+07	0.1482E+07	0.1350E+07	0.1223E+07	0 1101E-07
0	0.	0.	0.	0.	-5915.	-15157.	-24399.	-33642.	-42884.	-52126.	-61368.	-70610.	-79852.	-89095.	-98337.	-107579.	-116821.	-126063.	-135306.	-144548.	-153790.	-163032.	-172274.	-203004.	-333668.	-522706.	-713745.	-850414.	-896168.	-926973.	-957778.	-988583.	-1019388.	-1050193.	-1080997.	-1111802.	-1142607.	-1173412.	-1204217.	-1235022.	-1265827.	-1296631.	-1327436.	-1358241.	-1389046.	
0	.0	.0	.0	0.	0.	0.	0.	0.	0.	.0	.0	0.	.0	0.	0.	0.	0.	0.	0	0.	0.	0.	0.	0.	0.	0.	.0	.0	0.	0.	0.	.0	0.	.0	.0	.0	0.	.0	0.	0	0.	0	.0	0	.0	
0.	0.	0.	0.	0.	0.	0.	.0	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	-1154.	-8748.	-23094.	-43795.	-70454.	-101961.	-134099.	-166236.	-198374.	-230511.	-262649.	-294786.	-326923.	-359061.	-391198.	-423336.	-455473.	-487611.	-519748.	-551886.	-584023.	-616160.	
0.	0.	0.	0.	0.	.0	0.	.0	0.	0.	0.	0.	0.	0.	0.	.0	.0	0.	0.	.0	0.	.0	0.	0.	-20922.	-119059.	-206138.	-177886.	69967.	589028.	1149554.	1710081.	2270608.	2831135.	3391663.	3952190.	4512717.	5073245.	5633772.	6194300.	6754827.	7315354.	7875882.	8436409.	8996936.	9557463.	
.6	24.	47.	80.	123.	176.	241.	318.	407.	509.	624.	753.	896.	1053.	1224.	1409.	1608.	1821.	4895.	5038.	5176.	5307.	5432.	5551.	5662.	5767.	5866.	5957.	6043.	6122.	6195.	6262.	6325.	6382.	6435.	6484.	6529.	6571.	6610.	6647.	6683.	6718.	6752.	6787.	6822.	6860.	
-2590.	-7734.	-16806.	-30683.	-50178.	-76054.	-109036.	-149820.	-199073.	-257442.	-325555.	-404015.	-493168.	-593034.	-703614.	-824907.	-956914.	-1099635.	-1253067.	-1417161.	-1591746.	-1776630.	-1971622.	-2176535.	-2391175.	-2615345.	-2848849.	-3091491.	-3343067.	-3603380.	-3872219.	-4149384.	-4434665.	-4727849.	-5028724.	-5337075.	-5652680.	-5975320.	-6304769.	-6640800.	-6983183.	-7331679.	-7686053.	-8046055.	-8411446.	-8781966.	
712.	1417.	2309.	3372.	4595.	5970.	7489.	9147.	10939.	12860.	14907.	17068.	19250.	21432.	23614.	25797.	27979.	30161.	32342.	34498.	36615.	38694.	40734.	42734.	.0	0.	0.	0.	0.	32660.	30944.	29269.	27638.	26050.	24505.	23005.	21549.	20139.	18775.	17458.	16189.	14967.	13795.	12673.	11602.	10583.	
-0.2581E+04	-0.7710E+04	-0.1676E+05	-0.3060E+05	-0.5006E+05	-0.8179E+05	-0.1240E+06	-0.1739E+06	-0.2323E+06	-0.2998E+06	-0.3771E+06	-0.4646E+06	-0.5629E+06	-0.6718E+06	-0.7915E+06	-0.9218E+06	-0.1063E+07	-0.1215E+07	-0.1374E+07	-0.1547E+07	-0.1731E+07	-0.1925E+07	-0.2129E+07	-0.2343E+07	-0.2611E+07	-0.3071E+07	-0.3595E+07	-0.4021E+07	-0.4188E+07	-0.4006E+07	-0.3778E+07	-0.3557E+07	-0.3345E+07	-0.3140E+07	-0.2943E+07	-0.2754E+07	-0.2572E+07	-0.2397E+07	-0.2229E+07	-0.2067E+07	-0.1912E+07	-0.1763E+07	-0.1620E+07	-0.1482E+07	-0.1350E+07	-0.1223E+07	
3.31	3.81	4.21	4.56	4.86	5.12	5.37	5.59	5.80	6.00	6.18	6.29	6.29	6.29	6.29	6.29	6.29	6.29	6.28	6.22	6.17	6.11	6.05	5.99	5.93	5.87	5.81	5.74	5.68	5.61	5.54	5.47	5.40	5.33	5.25	5.18	5.10	5.01	4.93	4.84	4.75	4.66	4.56	4.46	4.35	4.24	
3.93	5.89	7.86	9.82	11.78	13.75	15.71	17.67	19.64	21.60	23.57	25.53	27.49	29.46	31.42	33.39	35.35	37.31	39.28	41.24	43.20	45.17	47.13	49.10	51.06	53.02	54.99	56.95	58.91	60.88	62.84	64.81	66.77	68.73	70.70	72.66	74.63	76.59	78.55	80.52	82.48	84.44	86.41	88.37	90.34	92.30	

0 8697E+D6	0.7604E+06	0.65508+06	0.55308+06	0 4541E+06	0 35808+06	D 2645FL06	00+31571 0	0.11/JIE+00	0.1254E+05						WAY NOVENII	INTERIOR VER	CU 12/215.0	PU-21/07.0	0.17375404	C0+3/2/T.0	0 51855405	0 8447F+05	0.12775+06	0.17918+06	0.23918+06	0.3086E+06	0.3882E+06	0.4785E+06	0.5798E+06	0.6921E+06	0.8156E+06	0.9501E+06	0.1096E+07	0.1252E+07	0.1420E+07	0.1599E+07	0.1789E+07	0.1989E+07	0.2199E+07	0.2420E+07	0.2611E+07	0.3071E+07	0.3595E+07	0.4556E+07	0.6198E+07
-1481460.	-1512265.	-1543070	-1573875.	-1604680	-1635485	-1666290	-1697094	-1777890	-1758704.		CGM	51.1			UNE	0000						-5878 -5878	-15063	-24248.	-33433.	-42618.	-51803.	-60988.	-70173.	-79358.	-88543.	-97728.	-106913.	-116098.	-125283.	-134468.	-143653.	-152838.	-162023.	-171208.	158344.	2063313.	4888512.	7745263.	9744895.
0.		.0	0.	.0	. 0		c		-17587.		STAMA	6.55			TMA		. c		ċc	o c			0.	.0		. 0	0.	0.0	0.	0.	0.	0.	0.	0.	0	0.	0	0.	.0	0.	.0	0.	0.	0.	.0
712573.	744710.	776848.	308985.	341123.	373260.	905398.	37535.	969672	01810.		CLAQB	0.02			RMP	0		. c				. 0	.0	.0	.0	.0	0.	0.	0.	0.	.0	0.	0.	.0	0	0.	0.	.0	0.	0.	-1192.	-9037.	23857.	45242.	72782.
9045	95737	0100.	6278	11558	16828	209		32640	179110		CLAQW	16.45	FGEAR	184619.	MM	0	. 0	.0	. 0	. 0	. 0	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	.0	.0	0.	.0	0	0.	0.	934.	770.	469	455	845
. 11239	. 11799	. 12360	. 1292(. 13481	. 14041	. 14602	. 15162	1572	. 16283		LIFTT	153032.	LIFT	153032.	ц															_											ŋ	33	58	20	-19
6993	7047	7109	7178	7258	7349	7454	7575	7716	7881		LIFTW	-54239.	LIFTW	-54239.	BMBL	0	1	- n	191	-10	-16	-23	-31	-41	-53	-66	-81	-98	-116	-137	-159.	-183	-209.	-237.	-636.	-655.	-672.	-690.	-706.	-721.	-736.	-749.	-762.	-774.	-785.
-9921700.	10310093.	10702260.	11097895.	11496694.	11898332.	12302474.	12708749.	13116777.	13526116.		LIFTB	-43.	LIFTB	-43.	BMBW	-412.	-2676.	-7990.	-17361.	-31697.	-51836.	-78567.	-112639.	-154770.	-205650.	-265948.	-336311.	-417364.	-509462.	-612628.	-726861.	-852162.	-988530.	-1135967.	-1294469.	-1463984.	-1644337.	-1835330.	-2036764.	-2248447.	-2470179.	-2701755.	-2942975.	-3193633.	-3453522.
7851.	7053	6316	5640	5028	4483	4010	3613	3299			DEFLEC	-149.10	DEFLEC	****	WSAV(I)	219.	712.	1417.	2309.	3372.	4595.	5970.	7489.	9147.	10939.	12860.	14907.	17068.	19250.	21432.	23614.	25797.	27979.	30161.	32342.	34498	36615.	38694.	40734.	42734.	.0	•	0		0
E+06	E+05	E+05		ALPHA	7.00	ALPHA	7.00	MENT	E+03	E+04	E+04	E+05	E+05	E+05	E+05	E+06	E+06	E+07	101	101 101	E+07	1.0+3	E+07	E+07	E+07	E+07	E+06	E+07	E+07	E+07															
-0.8697	-0.7604	-0.6550	-0.5530	-0.4541	-0.3580	-0.2645	-0.1731	-0.8337	-0.1254		LUSTAM	109722.	SLFM	2.58	BEND MO	-0.4127	-0.2677	-0.7993	-0.1737	-0.3171	-0.5185	-0.8447	-0.1277	-0.1791	-0.2391	-0.3086	-0.3882	-0.4785	-0.5798	-0.6921	-0.8156	-0.9501	-0.1096	2621.0-	-0.1420	-0.1599	-0.1.84	-0.1989	-0.2199	-0.2420	-0.23081	-0.61451	0.1979	0.4556	0.61981
3.87	3.73	3.58	3.42	3.24	3.04	2.81	2.54	2.21	1.73			ERS		ERS	Υ	2.60	3.31	3.81	4.21	4.56	4.86	5.12	5.37	5.59	5.80	6.00	6.18	6.29	6.29	6.29	6.29	6.29	0.29	6.49	0.28	27.9	0.1/	6.11	6.05	5.99	5.93	5.87	5.81	5.74	5.68
98.19	100.16	102.12	104.08	106.05	108.01	109.97	111.94	113.90	115.87	T AND TWO	SNITCHT	PARAMET	LANDING	PARAMETI	×	1.96	3.93	5.89	7.86	9.82	11.78	13.75	15.71	17.67	19.64	21.60	23.57	25.53	27.49	29.46	31.42	33.39 75 75	(), (), (), (), (), (), (), (), (), (), (), (),	10.00	87.45	41.24	43.20	45.17	47.13	49.10	51.06	53.02	54.99	56.95 10	58.91

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0.3545E+06	478253.	о.	°.	0.		-194	-123572.	12860.	E+06	C14CE.0	6.00	21.60
0.13CK12.0	.101C/C							10757.	00+J0		0.00	17.04
0.2001E+06	272109.	. 0		. 0		-121	-71913.	9147.	E+06	0.2001	5.59	17.67
0.1277E+06	169037.	.0	.0	.0		-92	-52337.	7489.	E+06	0.1166	5.37	15.71
0.8447E+05	65966.	0.	0.	0.		-67	-36506.	5970.	E+05	0.2939	5.12	13.75
0.5185E+05	0.	0	0.	0.		-47	-24086.	4595.	E+05	-0.2413	4.86	11.78
0.3171E+05	0.	.0	0.	0.		-30	-14728.	3372.	E+05	-0.1476	4.56	9.82
0.1737E+05	0.	0.	0.	0		-18	-8067.	2309.	E+04	-0.8085	4.21	7.86
0.7993E+04	0.	.0	0.	0		6-	-3712.	1417.	E+04	-0.3722	3.81	5.89
0.2677E+04	0.	.0	.0	0		-4	-1243.	712.	E+04	-0.1247	3.31	3.93
0.4127E+03	.0	.0	.0	0.		-	-192.	219.	E+03	-0.1923	2.60	1.96
MAX MOMENT	BMG	BMT	BMP	BMW	щ	BMBL) BMBW	WSAV(I	MENT	BEND MC	¥	×
				FGEAR 182465.	LIFTT 159419.	JEFLTW 159141.	LIFTB D -126	DEFLEC -57.45	АLРНА 7.00	SLFM 1.20	IRS	BUMP PARAMETE
	53.3	4.72	0.02	. 16.45	159419.	-159141.	-126.	-57.45	7.00	152181.	IRS	PARAMETE
	CGM	STAMA	CLAQB	CLAQW	LIFT	LIFTW	LIFTB	DEFLEC	ALPHA	WEIGHT		BUMP
0.1254E+05	19572002.	57091.	034910.	37151	4618	-1024	-13973018.	98.	E+04	0.1426	1.73	115.87
0.2287E+06	19241262.		001710.	97281	-4459	-1003	-13550155.	3299.	E+06	0.2287	2.21	113.90
0.7928E+06	18579784.	. 0	935312. 555533	[753	4141	-968	-12708947.	4010.	E+06	0.7928	2.81	109.97
0.1072E+07	18249046.	0.	902113.	2766	3982	-955	-12291453.	4483.	E+07	0.1072	3.04	108.01
0.1348E+07	17918306.	0.	868913.	6178	3823	-943	-11876544.	5028.	E+07	0.1348	3.24	106.05
0.1622E+07	17587568.	.0	835714.	1211		-933	-11464569.	5640.	E+07	0.1622	3.42	104.08
0.2158E+07	17756876	. c	/69316. 802515		-3505-	976- 976-	-11055867	, 1035. A15A	医+0/ 8+07	9C12.0	57.5 82.5	100.15
0.2421E+07	16595349.	0.	736116.	7830	3187	606	-10249512.	7851.	.E+07	0.2421	3.87	98.19
0.2679E+07	16264611.	0.	702917.	3842	3028	-902	-9852477.	8706.	E+07	0.2679	4.00	96.23
0.2933E+07	15933872.		669718.	9855	-2865	-897	-9459920.	9617.	E+07	0.2933	4.13	94.26
0 31838+07	15603133		616518 636518	1868 -	J116-	198-	-9072122	10583	E+07	12#C.U	00 T	40.06 40 30
0.3666E+07	14941654.		570120. 22222		-2392	-882	-8311897.	12673.	E+07	0.3666	4.46	88.37
0.3899E+07	14610915.	0.	536921.	906	2233	-877	-7939999.	13795.	E+07	0.3899	4.56	86.41
0.4127E+07	14280176.	0.	503721.	8181	2074	-873	-7573917.	14967.	E+07	0.4127	4.66	84.44
0.4348E+07	13949437.	.0	470522.		-1915	-868	-7213907.	16189.	E+07	0.4348	4.75	82.48
0.4563E+07	13618698		437323		-1756	-864	-6860212.	17458.	E+07	0.4563	4.84	80.52
0 4772F+07	13787959		404174.	- · · · · · · · · · · · · · · · · · · ·	1597	470- 658-	-6513078	18775.	E+07	P124.0	10.0	78 55
0.168E+U7	12626481.		337725.		5/71	- 848	-5839444.	21549.	E+U/	891C.U	01.0 10	76.03
0.5356E+07	12295742.	0.	304526.		-1120	-842	-5513412.	23005.	E+07	0.5356	5.18	72.66
0.5536E+07	11965003.	.0	271326.	2007	962	-836	-5194873.	24505.	E+07	0.5536	5.25	70.70
0.5708E+07	11634264.	.0	238127.	1020	803	-829	-4884057.	26050.	E+07	0.5708	5.33	68.73
0.5873E+07	11303525.	0.	204928.	1032.	-644	-822	-4581186.	27638.	E+07	0.5873	5.40	66.77
0.6029E+07	10972785.		171729.		-485-	-814	-4286480.	29269.	E+07	0.6029	5.47	64.81
0.6176E+07	10642047		138529.		-326-	-805	-4000157.	30944.	E+07	0.6176	10.0	62.84
0 63168-07	00511501	c	105230	1 2 0 1	-167	-795	2580075-	17660	2073	3153 0	5 £1	60 09

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0.7789E+07 0.7754E+07 0.7754E+07	9484074. 9994344.		-35818. -48941. -64367	-24243. -459183. -896147	-2362	-1729622.	32660. 30944.	0.754E+07 0.7516E+07	5.54	88 3 4
0.4609E+07 0.6591E+07	5829373. 7959901.	. 0 0	-11085. -21022.	160697. 138673.	-2236. -2271.	-1367448. -1483916. -1604677		0.4609E+07 0.6591E+07 0.7789E+07	5.81 5.74 5.68	6 6 F
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0.1420E+07 0.1599E+07	1405898. 1508969.	. 0 . 0	0.0	. 0 0	-1866. -1921.	-601472. -680237.	32342. 34498. 34415	0.8026E+06 0.8268E+06 0.8160E+06	6.28 6.22 6.17	2 7 0
0.1252E+07 0.1252E+07	1199754. 1302826.			0.0	-613. -694.	-459319. -527825.	27979. 30161.	0.7398E+06 0.7743E+06	6.29 6.29	31
0.8156E+06 0.8156E+06 0.9501E+06	993611. 1096683				-467.	-337734. -395955.	23614. 25797.	0.6554E+06 0.7002E+06	6.29 6.29	42
0.5798E+06	684396. 787468. 000520		 		-342.	-236721. -284656.	19250. 21432.	0.5504E+06 0.6055E+06	6.29 6.29	49 46
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29.4575 | 15.7107
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31.4213 | 29.4575
31.4213 | 31.4213 | ; | 33.3852 | 35.3490 | 37.3128 | 39.2767
 | 41.2405 | 43.2043 | 45.1682 | 47.1320 | 49.0958 | 51.0597 | 53.0235
 | 54.9873

 | 56.9512 | 58.9150 1 | 60.8788 | 62.8427 | 64.8065 1 | 66.7703 1 | 68.7342 | 70.6980 | 72.6618
 | 74.6257 | 76.5895 | 78.5533 |

80 5172	7811461 500	0000 0	54000 0000	0 1637		21 6E73	u	1013 11	0000 0		
82.4810	7401658.500	0,0000	54000.0039	0.1611	0.0790	C2C0.42	n ư	2000.07	2000.2 7015 C	0.1412 0	
84.4449	6987453.000	0.000	54000.0000	0.1583	0.0776	23 8314	۰ v	68 1393	2105 0	0 1351	MIR
86.4087	6569014.500	0.0000	53999.9961	0.1552	0.0761	23.3724	പ	65.3170	2.2573	0.1321	BUM
88.3725	6146523.000	0.0000	54000.0000	0.1519	0.0745	22.8765	5	62.4408	2.2094	0.1290	BUM
90.3364	5720152.500	0.0000	54000.0000	0.1483	0.0728	22.3396	ъ	59.5061	2.1575	0.1259	BUM
92.3002	5290087.500	0.0000	54000.0039	0.1445	0.0709	21.7564	ъ	56.5072	2.1012	0.1227	BUM
94.2640	4856512.500	0.0000	54000.0000	0.1402	0.0688	21.1206	<u>م</u>	53.4377	2.0398	0.1196	BUM
96.2279	4419615.000	0.0000	54000.0078	0.1356	0.0665	20.4236	5	50.2898	1.9725	0.1164	BUM
98.1917	3979597.500	0.0000	54000.0039	0.1305	0.0640	19.6549	ъ	47.0541	1.8982	0.1131	BUM
100.1555	3536653.500	0.0000	54000.0000	0.1248	0.0612	18.7997	5	43.7189	1.8157	0.1099	BUM
102.1194	3090988.500	0.0000	53999.9961	0.1185	0.0581	17.8381	S	40.2695	1.7228	0.1067	BUM
104.0832	2642824.500	0.0000	54000.0000	0.1112	0.0545	16.7412	S	36.6869	1.6168	0.1036	BUM
106.0471	2192373.000	0.0000	54000.0000	0.1027	0.0504	15.4648	S	32.9456	1.4936	0.1007	BUM
108.0109	1739874.000	0.0000	54000.0000	0.0926	0.0454	13.9380	ъ	29.0100	1.3461	0.0984	BUM
109.9747	1285563.000	0.0000	54000.0039	0.0799	0.0392	12.0336	ъ	24.8271	1.1622	0.0975	BUM
111.9386	829707.000	0.0000	46374.3086	0.0734	0.0360	12.8718	۴	20.3117	1.0676	0.0640	BUM
113.9024	372576.000	0.0000	27633.7871	0.0734	0.0360	21.6011	m	15.3064	1.0676	0.0171	BUM
115.8662	18816.750	0.0000	2263.7334	0.0734	0.0360	263.6889	٣	9.4367	1.0676	0.0001	MAN
115.8662	0.000	0.0000	0.0000	0.0734	0.0360	263.6889	m	9.4367	1.0676	0.0001	NONE
1	ST	RUCTURAL	WEIGHT SUMMARY								
	WE	IGHT	WEIGHT	ND	IT						
	<u>Г</u>)	BS)	FRACTION	MEI	GHT						
				(LBS/)	FT*FT)						
		* * * *									
SHELL	57	19.80	0.0376	1.5	945						
FRAMES	e	73.81	0.0025	0.1	042						
NONOP	95	58.42	0.0628	2.6	646						
SEC		0.00	0.0000	0.0	000						
TOTAL	156	52.02	0.1029	4.3	634						
VOLPEN		0.00	0.0000	0.0	000						
GRANTO	Т 156	52.02	0.1029	4.3	634						
Surfac	e Area, SQF		3587.15								
Volume	Ratio		1.00000000								
BODY W 1	EIGHT		15652.02441406								
FUSE	BENDING	THIC	SHELL	EQUIV	GAGE	FRAME	ΓN	SECTION	SHELL	FRAME	MAX
STAT	MOMENT		STRESS	THICK	THICK	SPACE		AREA	TWT INU	UNITWT	BENDING
FT	FT LBS	IN	ISI	NI	IN	IN		SQ FT	LB/FT2		
1.9638	619.015	0.0000	33.1421	0.0734	0.0360	18010.9727	en	21.2041	1.0676	0.0000	ILAN
3.9277	4015.657	0.0000	132.5520	0.0734	0.0360	4503.2979	Ē	34.3929	1.0676	0.0000	LAN
5.8915	11989.037	0.0000	298.2244	0.0734	0.0360	2001.5850	m (45.6394	1.0676	0.0000	LAN
7.46	Z6050.969	0.0000	530.1569	0.0734	0.0360	1125.9333	e	55.7851	1.0676	0.0000	LAN

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0.1067	1.7228	40.2695	ŝ	17.8381	0.0581	0.1185	53999.9961	0.0000	3090988.500	102.1194
0.1099	1.8157	43.7189	v N	18.7997	0.0612	0.1248	54000.0000	0.0000	3536653.500	100.1555
0.1131 0.1131	C2/Y.L	872.0541	റഗ	20.4235 19.6549	0,0640	0.1305	6/00.00045	0,0000	000-CTQ4T44	98.1917
0.1196	2.0398	53.4377	ın ı	21.1206	0.0688	0.1402	54000.0000	0.0000	4856512.500	94.2640
0.1227	2.1012	56.5072	ŝ	21.7564	0.0709	0.1445	54000.0039	0.0000	5290087.500	92.3002
0.1259	2.1575	59.5061	Ŋ	22.3396	0.0728	0.1483	54000.0000	0.0000	5720152.500	90.3364
0.1290	2.2094	62.4408	ហ	22.8765	0.0745	0.1519	54000.0000	0.0000	6146523.000	88.3725
0.1321	2.2573	65.3170	ŝ	23.3724	0.0761	0.1552	53999.9961	0.0000	6569014.500	86.4087
0.1351	2.3016	68.1393	ŝ	23.8314	0.0776	0.1583	54000.0000	0.0000	6987453.000	84.4449
0.1382	2.3427	70.9119	Ś	24.2571	0.0790	0.1611	54000.0039	0.0000	7401658.500	82.4810
0.1412	2.3809	73.6382	ŝ	24.6523	0.0803	0.1637	54000.0000	0.0000	7811461.500	80.5172
0.1442	2.4163	76.3216	ŝ	25.0195	0.0815	0.1661	54000.0000	0.0000	8216694.000	78.5533
0.1471	2.4493	78.9646	S	25.3607	0.0826	0.1684	53999.9961	0.0000	8617191.000	76.5895
0.1501	2.4799	81.5699	S	25.6778	0.0836	0.1705	54000.0000	0.0000	9012783.000	74.6257
0.1531	2.5084	84.1396	Ŋ	25.9722	0.0846	0.1725	53999.9961	0.0000	9403313.000	72.6618
0.1561	2.5347	86.6757	5	26.2454	0.0855	0.1743	54000.0000	0.0000	9788623.000	70.6980
0.1591	2.5592	89.1801	ŝ	26.4984	0.0863	0.1760	54000.0000	0.0000	10168552.000	68.7342
0.1620	2.5818	91.6544	S	26.7324	0.0871	0.1775	54000.0000	0.0000	10542948.000	66.7703
0.1650	2.6026	94.1001	Ś	26.9482	0.0878	0.1789	53999.9961	0.0000	10911652.000	64.8065
0.1680	2.6218	96.5186	on o	27.1466	0.0884	0.1803	54000.0000	0.0000	11274519.000	62.8427
0.1785	1985.2	18/2.101	ה ו	26.8083 77 7765	0.08/3	08/1.0	54000.0000	0.000	11683106.000	UCIE.80
0.2209	2.1415	103.6227	'n	22.1738	0.0722	0.1472	54000.0000	0.000.0	9887048.000	56.9512
0.3301	1.4647	105.9439	S	15.1663	0.0494	0.1007	54000.0000	0.0000	6913951.500	54.9873
0.3595	1.0784	108.2432	4	12.6057	0.0364	0.0741	47831.5078	0 0 0 0 0 0	4606577.500	53.0235
0.2622	1.0676	110.5216	e	14.8400	0.0360	0.0734	40223.6992	0.0000	3915888.750	51.0597
0.2208	1.0676	112.7799	e	16.3334	0.0360	0.0734	36546.1406	0.0000	3630565.000	49.0958
0.1788	1.0676	115.0187	e	18.3304	0.0360	0.0734	32564.5059	0.0000	3299240.000	47.1320
0.1434	1.0676	117.2388	m	20.6631	0.0360	0.0734	28888.3418	0.0000	2983286.250	45.1682
0.1139	1.0676	119.4408	ŝ	23.4073	0.0360	0.0734	25501.5098	0.0000	2682994.000	43.2043
0.0894	1.0676	121.6254	ŝ	26.6608	0.0360	0.0734	22389.4512	0.0000	2398660.500	41.2405
0.0693	1.0676	123.7931	ŝ	30.5504	0.0360	0.0734	19538.9277	0.0000	2130582.000	39.2767
0.0537	1.0676	124.2313	m	34.7735	0.0360	0.0734	17165.9688	0.0000	1878452.250	37.3128
0.0411	1.0676	124.2313	ŝ	39.7452	0.0360	0.0734	15018.6953	0.0000	1643478.750	35.3490
0.0309	1.0676	124.2313	m	45.8354	0.0360	0.0734	13023.1621	0.0000	1425109.750	33.3852
0.0228	1.0676	124.2313	ŝ	53.3949	0.0360	0.0734	11179.3613	0.0000	1223344.750	31.4213
0.0164	1.0676	124.2313	m	62.9179	0.0360	0.0734	9487.2998	0.0000	1038184.313	29.4575
0.0115	1.0676	124.2313	ŝ	75.1131	0.0360	0.0734	7946.9687	0.0000	869627.625	27.4937
0.0082	1.0676	124.2313	m	88.8384	0.0360	0.0734	6719.1787	0.0000	735272.000	25.5298
0.0064	1.0676	120.0712	ŝ	99.0743	0.0360	0.0734	6024.9893	0.0000	637229.750	23.5660
0.0047	1.0676	112.9981	m	111.7374	0.0360	0.0734	5342.1821	0.0000	531729.500	21.6022
0.0031	1.0676	105.7277	m	132.6108	0.0360	0.0734	4501.3037	0.0000	419206.312	19.6383
0.0017	1.0676	98.2337	m	172.1056	0.0360	0.0734	3468.3428	0.0000	300112.031	17.6745
0.0008	1.0676	90.4832	m	248.3074	0.0360	0.0734	2403.9612	0.0000	191600.266	15.7107
0.0004	1.0676	82.4333	m	342.0873	0.0360	0.0734	1744.9386	0.0000	126702.141	13.7468
0.0002	1.0676	74.0269	m	500.4382	0.0360	0.0734	1192.7972	0.0000	77778.055	11.7830
0.0001	1.0676	65.1837	m	720.6165	0.0360	0.0734	828.3481	0.0000	47561.238	9.8192
	0.0001 0.0008 0.0008 0.0008 0.0017 0.0008 0.0016 0.00115 0.00168 0.00115 0.00115 0.00115 0.00115 0.00115 0.00115 0.00113 0.0000000000	$\begin{array}{cccccccccccccccccccccccccccccccccccc$	65.1837 1.0676 0.0001 74.0269 1.0676 0.00031 90.4832 1.0676 0.00031 90.4832 1.0676 0.0017 98.2337 1.0676 0.00131 1105.7277 1.0676 0.00131 112.9981 1.0676 0.00131 112.9313 1.0676 0.00132 112.9313 1.0676 0.00143 124.2313 1.0676 0.00164 124.2313 1.0676 0.01159 124.2313 1.0676 0.01163 124.2313 1.0676 0.01139 124.2313 1.0676 0.01139 124.2313 1.0676 0.01139 124.2313 1.0676 0.01139 124.2313 1.0676 0.01139 124.2313 1.0676 0.01139 124.2313 1.0676 0.01139 124.2313 1.0676 0.01139 119.4408 1.0676 0.01139 117.2388 1.0676 0.01139 117.2388 1.0676 0.01139 117.2388 1.0676 0.01139 117.2388 1.0676 0.01139 117.2388 1.0676 0.01431 117.2388 1.0676 0.01139 1117.2389 1.0676 0.01139 1117.2389 1.0676 0.01139 1117.2389 1.0676 0.01139 1117.2389 1.0676 0.01139 1117.2389 1.0676 0.01399 1117.2389 <th>3$65.1837$$1.0676$$0.0004$3$92.4333$$1.0676$$0.0004$3$90.4832$$1.0676$$0.0004$3$98.2337$$1.0676$$0.0011$3$105.7277$$1.0676$$0.0047$3$112.0712$$1.0676$$0.0047$3$1224.2313$$1.0676$$0.00164$3$124.2313$$1.0676$$0.0115$3$124.2313$$1.0676$$0.01164$3$124.2313$$1.0676$$0.01164$3$124.2313$$1.0676$$0.0113$3$124.2313$$1.0676$$0.0113$3$124.2313$$1.0676$$0.0113$3$124.2313$$1.0676$$0.0141$3$124.2313$$1.0676$$0.0141$3$124.2313$$1.0676$$0.0143$3$124.2313$$1.0676$$0.0141$3$124.2313$$1.0676$$0.0141$3$124.2313$$1.0676$$0.0141$3$124.2313$$1.0676$$0.0141$3$124.2313$$1.0676$$0.0141$3$124.2313$$1.0676$$0.0141$3$112.7799$$1.0676$$0.0141$3$112.7799$$1.0676$$0.0141$3$112.738$$1.0676$$0.0141$4$108.432$$1.0676$$0.0141$5$102.238$$1.0676$$0.0141$6$100.2289$$1.0676$$0.1431$5$101.2$</th> <th>720. 616565.18371.06760.0001500. 4382365.18371.06760.0003248. 3074392. 43331.06760.0004248. 3074392. 43331.06760.0014248. 3074392. 23371.06760.0014172. 1056392. 23371.06760.0014183. 61083105. 72771.06760.001499. 74533122. 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7946.9891 0.0774 0.0360 94.7422 3 124.2131 11.0676 0.014 11179.3612 0.0774 0.0360 94.7422 3 124.2131 11.0676 0.014 11176.5993 0.0734 0.0360 94.7422 3 124.2131 11.0676 0.014 11176.5993</th><th>0.0000 832.341 0.0734 0.0306 720.615 3 5.2337 1.0676 0.0000 0.0000 174.3165 0.7314 0.3360 742.317 1.0676 0.0001 0.0000 174.3165 0.734 0.360 112.5156 3 98.2337 1.0676 0.001 0.0000 4501.307 0.7134 0.360 112.5156 3 98.2337 1.0676 0.001 0.0000 4501.307 0.7134 0.306 112.12431 1.12.9981 1.0676 0.001 0.0001 91477.3611 0.734 0.306 92.3134 3 1.24.2311 1.0676 0.001 0.0001 1125.5411 0.734 0.306 34.7352 1.0676 0.013 0.0001 1125.5411 0.734 0.306 34.735 1.0676 0.014 0.0001 1125.5411 0.734 0.306 34.735 1.0676 0.014 0.0001 1125.5411 1.75124 1.12.4231 1.0676</th><th>TYT78 0.000 182.3 441 0.011 12.3 14.0 10.77 0.000 TYT78 0.000 112.3 10.71 0.000 112.3 10.77 0.000 115670.141 0.0000 1441.107 0.773 0.0100 1441.107 0.773 0.0100 1441.107 0.773 0.0100 1441.107 0.0000 1441.107 0.0112 100.772 0.0001 1441.107 0.0001 1441.107 0.0001 1441.107 0.0001 1441.107 0.0001 1441.107 0.0001 1441.107 0.0001 1441.107 0.0001 1441.107 0.0001 1441.107 0.0001 1441.107 0.0001 1441.107 0.0001 1441.107 0.0001 1441.107 0.0011 1441.107 0.0011 1452.107 0.0000 1491.2401 0.0734 0.0300 57.111 1075 0.0101 1455.107 0.0000 11011.714 0.0101 0.0111 0.0101 0.0111 0.0111 0.0111 0.0101</th></t<></th>	3 65.1837 1.0676 0.0004 3 92.4333 1.0676 0.0004 3 90.4832 1.0676 0.0004 3 98.2337 1.0676 0.0011 3 105.7277 1.0676 0.0047 3 112.0712 1.0676 0.0047 3 1224.2313 1.0676 0.00164 3 124.2313 1.0676 0.0115 3 124.2313 1.0676 0.01164 3 124.2313 1.0676 0.01164 3 124.2313 1.0676 0.0113 3 124.2313 1.0676 0.0113 3 124.2313 1.0676 0.0113 3 124.2313 1.0676 0.0141 3 124.2313 1.0676 0.0141 3 124.2313 1.0676 0.0143 3 124.2313 1.0676 0.0141 3 124.2313 1.0676 0.0141 3 124.2313 1.0676 0.0141 3 124.2313 1.0676 0.0141 3 124.2313 1.0676 0.0141 3 124.2313 1.0676 0.0141 3 112.7799 1.0676 0.0141 3 112.7799 1.0676 0.0141 3 112.738 1.0676 0.0141 4 108.432 1.0676 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0.0734 0.0360 342.087 3 94.333 1.0676 0.0000 1144.92 7772 0.0360 342.087 3 94.333 1.0676 0.0000 3468.332 0.0734 0.0360 342.087 3 94.332 1.0676 0.0001 5461.307 0.0360 94.0361 3 10.571 1.0676 0.001 5461.307 0.0360 94.0361 3 112.200712 0.001 94.047 5461.3179 0.0360 94.0431 3 112.3167 0.001 7946.9891 0.0774 0.0360 94.7422 3 124.2131 11.0676 0.014 11179.3612 0.0774 0.0360 94.7422 3 124.2131 11.0676 0.014 11176.5993 0.0734 0.0360 94.7422 3 124.2131 11.0676 0.014 11176.5993</th><th>0.0000 832.341 0.0734 0.0306 720.615 3 5.2337 1.0676 0.0000 0.0000 174.3165 0.7314 0.3360 742.317 1.0676 0.0001 0.0000 174.3165 0.734 0.360 112.5156 3 98.2337 1.0676 0.001 0.0000 4501.307 0.7134 0.360 112.5156 3 98.2337 1.0676 0.001 0.0000 4501.307 0.7134 0.306 112.12431 1.12.9981 1.0676 0.001 0.0001 91477.3611 0.734 0.306 92.3134 3 1.24.2311 1.0676 0.001 0.0001 1125.5411 0.734 0.306 34.7352 1.0676 0.013 0.0001 1125.5411 0.734 0.306 34.735 1.0676 0.014 0.0001 1125.5411 0.734 0.306 34.735 1.0676 0.014 0.0001 1125.5411 1.75124 1.12.4231 1.0676</th><th>TYT78 0.000 182.3 441 0.011 12.3 14.0 10.77 0.000 TYT78 0.000 112.3 10.71 0.000 112.3 10.77 0.000 115670.141 0.0000 1441.107 0.773 0.0100 1441.107 0.773 0.0100 1441.107 0.773 0.0100 1441.107 0.0000 1441.107 0.0112 100.772 0.0001 1441.107 0.0001 1441.107 0.0001 1441.107 0.0001 1441.107 0.0001 1441.107 0.0001 1441.107 0.0001 1441.107 0.0001 1441.107 0.0001 1441.107 0.0001 1441.107 0.0001 1441.107 0.0001 1441.107 0.0001 1441.107 0.0011 1441.107 0.0011 1452.107 0.0000 1491.2401 0.0734 0.0300 57.111 1075 0.0101 1455.107 0.0000 11011.714 0.0101 0.0111 0.0101 0.0111 0.0111 0.0111 0.0101</th></t<>	$\begin{array}{cccccccccccccccccccccccccccccccccccc$	$\begin{array}{llllllllllllllllllllllllllllllllllll$	82.3481 0.0734 0.0360 32.0.482 3 6.1837 1.0676 0.0000 1192.7772 0.0734 0.0360 342.087 3 94.333 1.0676 0.0000 1144.92 7772 0.0360 342.087 3 94.333 1.0676 0.0000 3468.332 0.0734 0.0360 342.087 3 94.332 1.0676 0.0001 5461.307 0.0360 94.0361 3 10.571 1.0676 0.001 5461.307 0.0360 94.0361 3 112.200712 0.001 94.047 5461.3179 0.0360 94.0431 3 112.3167 0.001 7946.9891 0.0774 0.0360 94.7422 3 124.2131 11.0676 0.014 11179.3612 0.0774 0.0360 94.7422 3 124.2131 11.0676 0.014 11176.5993 0.0734 0.0360 94.7422 3 124.2131 11.0676 0.014 11176.5993	0.0000 832.341 0.0734 0.0306 720.615 3 5.2337 1.0676 0.0000 0.0000 174.3165 0.7314 0.3360 742.317 1.0676 0.0001 0.0000 174.3165 0.734 0.360 112.5156 3 98.2337 1.0676 0.001 0.0000 4501.307 0.7134 0.360 112.5156 3 98.2337 1.0676 0.001 0.0000 4501.307 0.7134 0.306 112.12431 1.12.9981 1.0676 0.001 0.0001 91477.3611 0.734 0.306 92.3134 3 1.24.2311 1.0676 0.001 0.0001 1125.5411 0.734 0.306 34.7352 1.0676 0.013 0.0001 1125.5411 0.734 0.306 34.735 1.0676 0.014 0.0001 1125.5411 0.734 0.306 34.735 1.0676 0.014 0.0001 1125.5411 1.75124 1.12.4231 1.0676	TYT78 0.000 182.3 441 0.011 12.3 14.0 10.77 0.000 TYT78 0.000 112.3 10.71 0.000 112.3 10.77 0.000 115670.141 0.0000 1441.107 0.773 0.0100 1441.107 0.773 0.0100 1441.107 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×	ж ,	Area	X-Xnose	ж о́с	Area	X	Ϋ́Υ	2.7
0.00	0.00	0.00	0.00	0.70	τ.1	15.91 20.01	13.42	-0.64
1.24	1.10	3.78	0.37	0.70	1. 53	18.9f	-13.42	-6.64
2.47	1.81	10.29	0.37	0.70	1.53			
3.71	2.40	18.17	1.76	0.70	1.53			
4.95	2.92	26.85						
6.19	3.38	35.97						
7.42	3.80	45.27						
8.66	4.17	54.55						
9.90	4.50	63.64						
1.14	4.80	72.40						
2.37	5.07	80.73						
3.61	5.31	88.52						
4.85	5.52	95.69						
6.08	5.70	102.17						
7.32	5.86	107.90						
8.56	5.99	112.83						
9.80	6.10	116.91						
1.03	6.18	120.12						
2.27	6.24	122.43						
3.51	6.28	123.83						
4.74	6.29	124.29						
6.16	6.29	124.29						
7.57	6.29	124.29						
8.98	6.29	124.29						
0.40	6.29	124.29						
1.81	6.29	124.29						
3.22	6.29	124.29						
4.64	6.29	124.29						
6.05	6.29	124.29						
7.46	6.29	124.29						
8.88	6.29	124.29						
2.83	6.28	123.83						
6.77	6.24	122.43						
0.72	6.18	120.12						
4.67	6.10	116.91						
8.62	5.99	112.83						
2.56	5.86	107.90						
6.51	5.70	102.17						
0.46	5.52	95.69						
4.41	5.31	88.52						

											Nacelles - 2	1.397	7.743 (each)
												:	
80.73	72.40	63.64	54.55	45.27	35.97	26.85	18.17	10.29	3.78	0.00	Fuselage	12.580	9.366 3522.634 9348.345
5.07	4.80	4.50	4.17	3.80	3.38	2.92	2.40	1.81	1.10	0.00		neter	Ratio
78.35	82.30	86.25	90.20	94.14	98.09	102.04	105.99	109.93	113.88	117.83		Max. Dian	Fineness Surface <i>l</i> Volume

Dimensions of Planar Surfaces (each)

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	Wing	H.Tail	V.Tail	Canard	Units
NUMBER OF SURFACES.	1.0	1.0	1.0	1.0	
PLAN AREA.	1450.0	277.5	203.5	0.0	(SQ.FT.)
SURFACE AREA	2923.7	403.1	408.6	0.0	(SQ.FT.)
VOLUME	2064.4	106.0	175.0	0.0	(CU.FT.)
SPAN	107.339	39.669	15.885	0.000	(FT.)
L.E. SWEEP	23.725	37.176	45.001	0.000	(DEG.)
C/4 SWEEP	20.000	33.400	39.400	0.000	(DEG.)
T.E. SWEEP	7.826	19.921	15.935	0.000	(DEG.)
ASPECT RATIO	7.946	5.670	1.240	0.000	
ROOT CHORD	21.614	10.923	18.485	0.000	(FT.)
ROOT THICKNESS	37.867	11.797	19.964	0.000	(IN.)
ROOT T/C	0.146	0.090	0.090	0.000	
TIP CHORD	5.403	3.069	7.135	0.000	(FT.)
TIP THICKNESS	7.133	3.315	7.706	0.000	(IN.)
TIP T/C	0.110	0.090	0.090	0.000	
TAPER RATIO	0.250	0.281	0.386	0.000	
MEAN AERO CHORD	15.130	7.731	13.648	0.000	(FT.)
LE ROOT AT	42.200	104.550	99.345	0.000	(FT.)
C/4 ROOT AT	47.603	107.281	103.966	0.000	(FT.)
TE ROOT AT	63.814	115.473	117.830	0.000	(FT.)
LE M.A.C. AT	51.635	110.664	106.114	0.000	(FT.)
C/4 M.A.C. AT	55.417	112.597	109.527	0.000	(FT.)
TE M.A.C. AT	66.764	118.395	119.763	0.000	(FT.)
Y M.A.C. AT	21.468	8.062	0.000	0.000	
LE TIP AT	65.787	119.592	115.230	0.000	(FT.)
C/4 TIP AT	67.137	120.359	117.014	0.000	(FT.)
TE TIP AT	71.190	122.661	122.365	0.000	(FT.)
ELEVATION	-6.290	5.032	6.290	0.000	(FT.)
GEOMETRIC TOTAL VOLU	ME COEFI	5 0.771	0.076	0.000	
REQUESTED TOTAL VOLU	ME COEFI	F 0.771	0.076	0.000	
ACTUAL TOTAL VOLUME	COEFF	0.771	0.076	0.000	
EXTENS	NOIS	10			
		Strake	e Rear	Extensic	u.
Centroid location at		0.0	0 00	.00	
Area		0.0	0 00	.00	
Sweep Angle		0.0	0 0(.00	
Wetted Area	•••••	0.0	0 0	.00	
Volume		0.0	0 00	.00	

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י - י ו Mission Fuel Required = 64309. lbs. Extra Fuel Carrying Capability = -9276. lbs. Available Fuel Volume in Wing = 1101. cu.ft.

Aircraft Weight = 152181.000 lbs. Aircraft Volume = 11693.691 cu.ft. Aircraft Density = 13.014 lbs./cu.ft. Actual - Required Fuel Volume = -185.529 cu.ft.

ICASE = 4 (Fineness Ratio Method)

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- ' \ Weight Statement - Transport TRANSPORT

Qmax: Design Load Factor: Ultimate Load Factor: Structure and Material: Wing Equation: Body Equation:	400. 2.50 3.75 Aluminum Skin, Stri Ardema/Chambers WMI Ardema/Chambers PDC	inger CNG Analys SYL Analys	iis Si s			
Component	Pounds K	(ilograms	Percent	Slope	Tech	Fixed
Airframe Structure	35228.	15979.	21.15			No
Wing	10315.	4679.	6.19	1.20	1.00	No
Fuselage	15652.	7100.	9.40	0.90	1.00	No
Horizontal Tail (Low) 1503.	682.	0.90	1.00	1.00	No
Vertical Tail	1480.	671.	0.89	1.00	1.00	No
Nacelles	4.	2.	0.00	1.00	1.00	No
Landing Gear	6275.	2846.	3.77	1.00	1.00	No
Propulsion	6546.	2969.	3.93			No
Engines (2)	6546.	2969.	3.93	0.85	1.00	Yes
Fuel System	0.	.0	00.00	1.00	1.00	No
Thrust Reverser	0.	0.	0.00	1.00		No
Fixed Equipment	24555.	11138.	14.74		1.00	No
Hyd & Pneumatic	661.	300.	0.40	1.00		No
Electrical	3891.	1765.	2.34	1.00		No
Avionics	2390.	1084.	1.43	1.00		No
Instrumentation	780.	354.	0.47	1.00		No
De-ice & Air Cond	1634.	741.	0.98	1.00		No
Aux Power System	928.	421.	0.56	1.00		No
Furnish & Egpt	12439.	5642.	7.47	1.00		No
Seats and Lavatories	6600.	2994.	3.96	1.00		No
Galley	1950.	885.	1.17	1.00		No
Misc Cockpit	234.	106.	0.14	1.00		No
Cabin Finishing	2900.	1315.	1.74	1.00		No
Cabin Emergency Equip	405.	184.	0.24	1.00		No
Cargo Handling	350.	159.	0.21	1.00		No
Flight Controls	1831.	831.	1.10	1.00		No
Empty Weight	66329.	30087.	39.82			

Operating Items		4707.	2135.	2.83	NO
Flight Crew	(2)	340.	154.	0.20	NO
Crew Baggage and	Provisions	175.	79.	0.11	No
Flight Attendents	: (4)	520.	236.	0.31	NO
Unusable Fuel and	l oil	542.	246.	0.33	NO
Passenger Service		3130.	1420.	1.88	No
Cargo Containers		0.	0.	0.00	NO
Operating Weight En	pty	71036.	32222.	42.65	
Fuel		64029.	29043.	38.44	
Payload		31500.	14288.	18.91	No
Passengers	(150)	27000.	12247.	16.21	No
Baggage		4500.	2041.	2.70	No
Cargo		0.	0.	0.00	No
Calculated Weight		166565.	75554.	100.00	No
Estimated Weight		152181.	69029.		
Percent Error				9.45	

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2 -Calling Module # Calling Module # TAKEOFF

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W = 0.1489386E+06	SMN2 = 0.2034092E+00	SFC0 = 0.100000E+01	
WFTO = 0.3242445E+04	V2 = 0.2259222E+03	TNO = 0.000000E+00	
WFTO2 = 0.000000E+00	VS = 0.1337697E+03	SFC2 = 0.100000E+01	FLTO =-0.1450002E+05
WFT01 = 0.3242445E+04	CLS = 0.1769708E+01	TN2 = 0.0000000E+00	SFCAVE = 0.1000000E+01
WGTO = 0.1521810E+06	HNTO = 0.1500000E+04	CL2 = 0.1758139E+01	TNAVE = 0.0000000E+00 :

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LANDING

 WGTO =
 0.1521810E+06
 WFUED =
 0.6402882E+05
 WFRES =
 0.1176211E+05
 WFTOT =
 0.6430918E+05
 WFUEL =
 0.6430918E+05

 WPL =
 0.3150000E+05
 W =
 0.8815213E+05
 WLAND =
 0.1246486E+06
 XGRLAN =
 0.1692168E+04
 FLLAND =
 0.5155803E+04

 WCCALC =
 0.1524791E+06
 WCCALC =
 0.1524791E+06
 Control =
 0.5155803E+04

	12.58000	9348.345	117.8300	0.2500000	7.946000	0.4040000	23.72453	0.9800000	1811.579	5780.107	24.74486	38.87792	0.1460000	0.1100000	2.000000	2.000000	0.2500000	0.0000000E+00	0.000000E+00	0.0000000E+00	0.000000E+00	2.50000	1.500000	4.3013584E-02	104.9524	7.110270	0.1389904
	H	11	0	łI	п	0	ŧ	0	0	IJ	п	п	H	П	IJ	II	u	IF	11	н	н	Ħ	II	П	II	0	0
Calling Module # 6	FROM geometry: body diameter	BODY VOLUME	BODY LENGTH	TAPER RATIO	ASPECT RATIO	RATIO 1/4 CHORD	WING SWEEP	HOR. TAIL / CL	NOSE VOLUME	TAIL VOLUME	CLIA	CLIB	T/C AT ROOT	T/C AT TIP	ENTEMP	ENWINGTEMP	CLRW1	CLRW2	CLRW3	CLRP1	CLRP2	FROM weights.acs SLFMTEMP	FACSTEMP	WFPTEMP	WINGLTEMP	UWWGTEMP	ARTTEMP

ASA ZIJO COMPOSITE

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8160000.	U.UUUUUUE+00	8160000.	0.0000000E+00		8160000.	0.0000000E+00	816000	0 000000E+00		5.600002E-02	0.0000000E+00		5.600002E-02	0.0000000E+00	5 6000028-02	0 0000000000000000000000000000000000000	0014000000.0	5.600002E-02	0.0000000E+00				11.25000	11.25000		11.25000	00062.11	3 2000002E-02	0.000000E+00		3.2000002E-02	0.0000000E+00									4	4
8160000.	0.0000000000000000000000000000000000000	8160000.	0.0000000E+00	0.0000000E+00	8160000.	0.0000000E+00	U.UUUUUUE+UU B160000	0 000000E+00	0.0000000E+00	5.600002E-02	0.0000000E+00	0.0000000E+00	5.600002E-02	0.0000000E+00	5.6000002E-02	0 0000000000000000000000000000000000000	0.0000000E+00	5.600002E-02	0.000000E+00	0.0000000E+00			11.25000	11.25000	11.25000	11.25000	11.25000	3.2000002E-02	0.000000E+00	0.0000000E+00	3.2000002E-02	0.0000000E+00	0.0000000E+00								4	4 4
8160000.	0.0000000E+00	8160000.	0.0000000E+00	0.0000000E+00	8160000.	0.0000000E+00	U.UUUUUUE+UU 8160000	0.000000000+00	0.0000000E+00	5.600002E-02	0.0000000E+00	0.0000000E+00	5.6000002E-02	0.0000000E+00	5.6000002E-02	0.00000008+00	0.0000000E+00	5.600002E-02	0.0000000E+00	0.0000000E+00	1.000000	6.2500003E-05	11.25000	11.25000	11.25000	11.25000	11.25000	3.20000028-02	0.000000E+00	0.0000000E+00	3.2000002E-02	0.0000000E+00	0.0000000E+00	1.0000000E-03	5.240000	000010.5	0.0000000E+00	0.000000E+00	2	0.0000000E+00	4	4
EST = =	0.0000000E+00	ESB =	0.0000000E+00	0.0000000E+00	EFT = =	0.0000000000000000000000000000000000000	U.UUUUUUE+UU EFB =	0.0000000E+00	0.000000E+00	DST =	0.0000000E+00	0.0000000E+00	DSB = =	0.0000000000000000000000000000000000000		0.0000000E+00	0.0000000E+00	DFB ≃	0.0000000E+00	0.0000000E+00	PS =	"	PGT =	11.25000	11.25000	PGB = 11 75000 =	11.25000	TMGT =	0.0000000E+00	0.0000000E+00	TMGB =	0.0000000E+00	0.0000000E+00	CLAQR =	CKF =	MCW III	WCA ==		IFUEL =	WEITR =	KCONT =	4
	0.0000000E+00		8160000.	0.0000000E+00		0 0000000 .00	0.420000000	8160000.	0.000000E+00		5.600002E-02	0.0000000E+00		20-320000000000000000000000000000000000	001400000000000000000000000000000000000	5.600002E-02	0.000000E+00		5.600002E-02	0.0000000E+00				11.25000	11.25000	11 76000	11.25000		3.2000002E-02	0.0000000E+00		3.2000002E-02	0.0000000E+00									4

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DENW LB/FT3 41.598 WFUEL LBS 42488.93 VWING FT3 1269.270 GAMS DEG. 20.000 GAMT DEG. 7.82576 TIPC TAPER TRATWR TRATWT GAML FT DEG. 5.4034 0.250 0.146 0.110 23.72453 0.2090000 0.000000E+00 PC TAPER TRATWR TRATWT TIPC BS ROOTC FT FT 50.764 19.8116 107.3392 SPAN FT

WUN TWTINU	WEBS	2 LB/FT2	0.5760 3	0 5760 3		0.5760 3	0.5760 3	0.5760 3 0.5760 3 0.5760 3	0.5760 3 0.5760 3 0.5760 3 0.5760 3	0.5760 3 0.5760 3 0.5760 3 0.5760 3 0.5760 3	0.5760 0.5760 0.5760 3 0.5760 3 0.5760 3 0.5760 3 0.5760 3	0.5760 0.5760 0.5760 0.5760 3 0.5760 3 0.5760 3 0.5760 3 0.5760 3 0.5760 3	0.5760 0.5760 0.5760 0.5760 3 0.5760 3 0.5760 3 0.5760 3 0.5760 3 0.5760 3 0.5760 3 0.5760 3 0.5760 3 0.5760 3 0.5760 3 3 0.5760 3 3 0.57760 3 3 0 3 0.5760 3 3 0.57760 3 3 0 3 0.5760 3 3 0 3 0 3 0 0 3 0 3 0 3 0 0 3 0 3 0	0.5760 0.5760 0.5760 0.5760 3 0.5760 3 0.5760 3 0.5760 3 0.5760 3 0.5760 3 0.5760 3 0.5760 3 0.5760 3 0.5760 3 0.5760 3 3 0.5760 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.5760 3 3 0 3 3 0.57760 3 3 0 3 3 0.57760 3 3 0 3 3 0.57760 3 3 0 3 0 3 0 0 3 0 3 0 3 0 3 0 3 0 0 3 0 3 0 0 3 0 3 0 0 3 0 0 3 0 3 0 0 0 3 0 0 0 3 0 3 0 0 3 0 3 0 0 3 0 0 0 3 0 0 0 3 0 3 0 0 3 0 0 3 0 3 0 0 3 0 3 0 0 3 0 3 0 0 3 0 0 3 0 0 3 0 0 3 0 0 3 0 0 3 0 0 3 0 0 3 0 0 3 0 0 3 0 0 3 0 0 3 0 0 3 0 0 3 0 0 3 0 0 3 0 0 3 0 0 0 3 0 0 0 0 3 0 0 0 3 0 0 0 0 3 0	0.5760 0.5760 0.5760 0.5760 3 0.5760 3 0.5760 3 0.5760 3 0.5760 3 0.5760 3 0.5760 3 0.5760 3 0.5760 3 0.5760 3 0.5760 3 5 0.5760 3 5 0.5760 3 5 0.57760 3 0.57760 3 5 0.57760 3 5 0 0.57760 3 5 0.57760 3 5 00.57760 3 5 0.57760 3 5 00.57760 3 5 0.57760 3 5 0.57760 3 5 0.57760 3 5 0.57760 3 5 0.57760 3 5 0.57760 3 5 0.57760 3 5 0 5 0.57760 3 5 0.57760 3 5 0 5 0.57760 3 5 0 5 0 5 0 5 0 5 0 5 0 5 0 5 0 5 0	0.5760 0.5760 0.5760 0.5760 0.5760 3 0.5760 3 0.5760 3 0.5760 3 0.5760 3 0.5760 3 0.5760 3 0.5760 3 0.5760 3 0.5760 3 0.5760 3 3 0.5760 3 3 0.5760 3 3 0.5760 3 3 0.5760 3 3 0.5760 3 3 0.5760 3 3 0.5760 3 3 3 0.5760 3 3 0.5760 3 3 0.5760 3 3 0.5760 3 3 0.5760 3 3 0.5760 3 3 0.5760 3 3 3 0.5760 3 3 3 0.5760 3 3 3 0.5760 3 3 3 0.5760 3 3 3 0.5760 3 3 0.5760 3 3 3 0.5760 3 3 3 0.5760 3 3 3 0.5760 3 3 3 0.5760 3 3 3 0.57760 3 3 0.5760 3 3 3 0.5760 3 3 0.57760 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 0.57760 3 3 0.57760 3 3 0.57760 3 3 3 0.5760 3 3 0.57760 3 3 3 0.57760 3 3 0.57760 3 3 0.57760 3 3 0.57760 3 3 0.57760 3 3 0.57760 3 3 3 0.57760 3 3 0.57760 3 3 0.57760 3 3 0.57760 3 3 0.57600 3 3 0.57760 5 0 3 0 5 0 5 0 5 0 5 0 5 0 5 0 5 0 5	0.5760 0.5760 0.5760 0.5760 0.5760 3 0.5760 3 0.5760 3 0.5760 3 0.5760 3 0.5760 3 0.5760 3 0.5760 3 0.5760 3 0.5760 3 0.5760 3 3 0.5760 3 3 0.5760 3 3 0.5760 3 3 0.5760 3 3 3 0.5760 3 3 3 0.5760 3 3 3 3 0.5760 3 3 3 0.5760 3 3 3 0.5760 3 3 0.5760 3 3 0.5760 3 3 3 0.5760 3 3 3 3 0.5760 3 3 3 0.5760 3 3 3 3 0.5760 3 3 3 0.5760 3 3 3 0.5760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 3 0.57760 3 3 3 3 3 0.57760 3 3 3 3 3 3 0.57760 3 3 3 3 0.57760 3 3 3 3 0.57760 3 3 3 0 3 3 0.57760 3 3 3 3 3 0.57760 3 3 3 3 3 00 3 5 0 0 3 5 0 0 3 5 0 0 3 3 5 0 0 5 0 5	0.5760 0.5760 0.5760 0.5760 0.5760 3 0.5760 3 0.5760 3 0.5760 3 0.5760 3 0.5760 3 0.5760 3 0.5760 3 0.5760 3 0.5760 3 0.5760 3 3 0.5760 3 3 0.5760 3 3 0.5760 3 3 0.5760 3 3 0.5760 3 3 0.5760 3 3 0.5760 3 3 0.5760 3 3 0.5760 3 3 0.5760 3 3 0.5760 3 3 0.5760 3 3 0.5760 3 3 3 0.5760 3 3 3 0.5760 3 3 3 0.5760 3 3 3 0.5760 3 3 3 0.5760 3 3 0.5760 3 3 0.5760 3 3 0.5760 3 3 3 0.5760 3 3 3 0.5760 3 3 3 0.5760 3 3 3 0.5760 3 3 3 0.57760 3 3 3 0.5760 3 3 3 0.57760 3 3 3 0.5760 3 3 3 0.57760 3 3 3 0.5760 3 3 3 0.57760 3 3 3 0.5760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 3 0.57760 3 3 3 3 0.57760 3 3 3 3 0.57760 3 3 3 0.57760 3 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.5760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 3 3 3 0.57760 5 5 5 0 5 5 5 5 5 5 5 5 5 5 5 5 5 5 5	0.5760 0.57760 0.577600 0.577600 0.57760000000000000000000000000000000000	0.5760 0.57760 0.577600 0.577600 0.57760000000000000000000000000000000000	0.5760 0.57760 0.577600 0.577600 0.57760000000000000000000000000000000000	0.5760 0.57760 0.577600 0.577600 0.57760000000000000000000000000000000000	0.5760 0.57760 0.55760 0.557600 0.557600 0.55760000000000000000000000000000000000	0.5760 0.57760 0.55760 0.557600 0.557600 0.55760000000000000000000000000000000000	0.5760 0.57760 0.577600 0.577600 0.57760000000000000000000000000000000000
IMLIND	COVERS	LB/FT2	0.7904	0.7904		0.7904	0.7904 0.7904	0.7904 0.7904 0.7904	0.7904 0.7904 0.7904 0.7904	0.7904 0.7904 0.7904 0.7904 0.7904	0.7904 0.7904 0.7904 0.7904 0.7904 0.7904 0.7904	0.7904 0.7904 0.7904 0.7904 0.7904 0.7904 0.7904	0.7904 0.7904 0.7904 0.7904 0.7904 0.7904 0.7904 0.7904	0.7904 0.7904 0.7904 0.7904 0.7904 0.7904 0.7904 0.7904	0.7904 0.7904 0.7904 0.7904 0.7904 0.7904 0.7904 0.7904 0.7904	0.7904 0.7904 0.7904 0.7904 0.7904 0.7904 0.7904 0.7904 0.7904 0.7904	0.7904 0.7904 0.7904 0.7904 0.7904 0.7904 0.7904 0.7904 0.7904 0.7904 0.7904 0.7904	0.7904 0.7904 0.7904 0.7904 0.7904 0.7904 0.7904 0.7904 0.9225 1.0179 1.1120	0.7904 0.7904 0.7904 0.7904 0.7904 0.7904 0.7904 0.7904 0.7904 0.9222 1.0179 1.1120	0.7904 0.7904 0.7904 0.7904 0.7904 0.7904 0.7904 0.7904 0.7904 0.7904 1.0179 1.0179 1.2040 1.2040	0.7904 0.7904 0.7904 0.7904 0.7904 0.7904 0.7904 0.7904 0.7904 1.0179 1.0179 1.2122 1.2040 1.2936 1.3808	0.7904 0.7904 0.7904 0.7904 0.7904 0.7904 0.7904 0.7904 0.7904 1.1120 1.1120 1.2120 1.2120 1.2622 1.4652	0.7904 0.7904 0.7904 0.7904 0.7904 0.7904 0.7904 0.7904 0.7904 1.1120 1.2022 1.2120 1.2120 1.2652 1.5468 1.5468	0.7904 0.7904 0.7904 0.7904 0.7904 0.7904 0.7904 0.7904 0.7904 1.0179 1.1120 1.2017 1.20179 1.2120 1.6256 1.6556 1.6256	0.7904 0.7904 0.7904 0.7904 0.7904 0.7904 0.7904 0.7904 0.7904 1.1120 1.2017 1.2017 1.2022 1.2120 1.2120 1.2625 1.6256 1.7014
WGAGE	THICK	IN	0.02000	0.02000		0.02000	0.02000	0.02000 0.02000 0.02000	0.02000 0.02000 0.02000	0.02000 0.02000 0.02000 0.02000	0.02000 0.02000 0.02000 0.02000 0.02000	0.02000 0.02000 0.02000 0.02000 0.02000 0.02000	0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000	0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000	0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000	0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000	0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000	0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000	0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000	0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000	0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000	0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000	0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000	0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000	0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000
CGAGE	THICK	NI	0.0200	0.0200	00000	0040.0	0.0200	0.0200	0.0200 0.0200 0.0200	0.0200 0.0200 0.0200 0.0200	0.0200 0.0200 0.0200 0.0200	0.0200 0.0200 0.0200 0.0200 0.0200 0.0200	0.0200 0.0200 0.0200 0.0200 0.0200 0.0200 0.0200	0.0200 0.0200 0.0200 0.0200 0.0200 0.0200 0.0200	0.0200 0.0200 0.0200 0.0200 0.0200 0.0200 0.0200 0.0200 0.0200	0.0200 0.0200 0.0200 0.0200 0.0200 0.0200 0.0200 0.0200 0.0200 0.0200 0.0200	0.0200 0.0200 0.0200 0.0200 0.0200 0.0200 0.0200 0.0200 0.0203 0.0203 0.0233	0.0200 0.0200 0.0200 0.0200 0.0200 0.0200 0.0200 0.0200 0.0203 0.0203 0.0233	0.0200 0.0200 0.0200 0.0200 0.0200 0.0200 0.0200 0.0200 0.0203 0.0203 0.0203 0.0203 0.0203 0.0203 0.0203 0.0203 0.0203 0.0203 0.0203 0.0203 0.0203 0.0203 0.0203 0.0203 0.0203 0.0203 0.0200	0.0200 0.0200 0.0200 0.0200 0.0200 0.0200 0.0200 0.0209 0.0233 0.0233 0.0233	0.0200 0.0200 0.0200 0.0200 0.0200 0.0200 0.0209 0.0209 0.0233 0.0233 0.0233 0.0233	0.0200 0.0200 0.0200 0.0200 0.0200 0.0200 0.0209 0.0233 0.0233 0.0233 0.0233 0.0233 0.0233 0.0233	0.0200 0.0200 0.0200 0.0200 0.0200 0.0200 0.0209 0.02281 0.02281 0.02281 0.02333 0.02333 0.02319 0.03319 0.03319	0.0200 0.0200 0.0200 0.0200 0.0200 0.0200 0.0209 0.02281 0.02281 0.02281 0.02281 0.02281 0.02333 0.02349 0.03349 0.03341 0.03341	0.0200 0.0200 0.0200 0.0200 0.0200 0.0200 0.02281 0.02200 0.022010000000000
WEB	THICK	NI	0.03960	0.03960	0.03960	~~~~~	0.03960	0.03960	0.03960 0.03960 0.03960	0.03960 0.03960 0.03960 0.03960	0.03960 0.03960 0.03960 0.03960 0.03960	0.03960 0.03960 0.03960 0.03960 0.03960 0.03960	0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960	0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960	0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960	0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960	0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960	0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960	0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960	0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960	0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960	0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960	0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960 0.03960	0.03960 0.03960	0.03960 0.033960 0.03396000000000000000000000000000000000
COVER	THICK	IN	0.0543	0.0543	0.0543		0.0543	0.0543	0.0543 0.0543 0.0543	0.0543 0.0543 0.0543 0.0543	0.0543 0.0543 0.0543 0.0543 0.0543	0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543	0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543	0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543	0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543	0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0567	0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543	0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0567 0.0567 0.0706	0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0567 0.0567 0.0765 0.0765 0.0765	0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0567 0.0567 0.0765 0.0765 0.0765 0.0765 0.0765 0.0765 0.0765 0.0765 0.0765 0.0765 0.0765 0.0765 0.0765 0.0766 0.0768 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.05630 0.05630 0.05630 0.05630000000000000000000000000000000000	0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0567 0.0567 0.0567 0.0760 0.0760 0.0760 0.0760 0.0760 0.0760 0.0760 0.0760 0.0760 0.0760 0.0760 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.056430 0.056430 0.056430 0.056430 0.0564300000000000000000000000000000000000	0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0567 0.057000000000000000000000000000000000	0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0567 0.0543 0.05670000000000000000000000000000000000	0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0543 0.0567 0.0543 0.0567 0.0543 0.05670000000000000000000000000000000000	0.0543 0.05543 0.05643 0.05667 0.05643 0.05667 0.05670000000000000000000000000000000000
MEB	SPACE	NI	0.1830	0.3699	0.5195	CC33 C	2200.0	0.7751	0.8915	0.7751 0.8915 1.0032	0.7751 0.8915 1.0032 1.1114	0.7751 0.8915 1.0032 1.1114 1.2169	0.7751 0.8915 1.0032 1.114 1.2169 1.3202	0.7751 0.8915 1.0032 1.1114 1.1114 1.2169 1.3202 1.3202 1.4217	0.7751 0.7751 0.8915 1.08315 1.1114 1.1114 1.2169 1.3202 1.3202 1.5217	0.7751 0.7751 0.8915 1.08315 1.1114 1.1114 1.2169 1.3202 1.5217 1.5217 1.6204	0.7751 0.8915 0.8915 1.0032 1.1114 1.1114 1.3202 1.3202 1.4217 1.5217 1.5217	0.7551 0.7751 0.8915 1.08915 1.1114 1.1114 1.3205 1.3205 1.4217 1.5217 1.5217 1.6204 1.7179 1.8146	0.7551 0.7751 0.8915 1.08915 1.1114 1.1114 1.2202 1.3202 1.4217 1.5217 1.6204 1.7179 1.8146 1.9104	0.7551 0.8915 0.8915 1.08915 1.1114 1.21032 1.32059 1.4217 1.6204 1.7179 1.9104 2.0054 2.0054	0.7551 0.7751 0.8915 1.08915 1.1114 1.21032 1.32059 1.4217 1.6204 1.7179 1.9104 1.9104 2.00974	0.7551 0.7751 0.8915 1.08915 1.1114 1.1114 1.32059 1.32059 1.62217 1.62217 1.9104 1.9104 2.0997 2.0997 2.09974 2.1934	0.7051 0.7751 0.7751 1.1114 1.1114 1.2169 1.2170 1.2170 1.2170 1.2170 1.2170 1.2170 1.2170 1.2170 1.2170 2.0000 2.00005 2.28655 2.28655 2.286555 2.28655555555555555555555555555555555555	0.7955 0.7751 0.7751 1.100315 1.110315 1.11114 1.21114 1.21114 1.21114 1.21174	0.000 0.707512 0.707512 0.707512 1.1009355 1.7110325 1.7110325 1.7110325 1.75217 1.75217 1.75217 1.75217 2.20997 2.100997 2.20997 2.20997 2.20997 2.20997 2.20997 2.20997 2.20997 2.20997 2.20997 2.20997 2.20997 2.20997 2.20997 2.20997 2.20997 2.20997 2.209777 2.209777 2.209777 2.209777 2.209777 2.20977 2.209777 2.209777 2.209777 2.209777 2.209777 2.209777 2.209777 2.209777 2.209777 2.209777 2.209777 2.2097777 2.2097777 2.209777 2.209777 2.2097777 2.209777777 2.20977777777777777777777777777777777777
DENU	MOM	FT-LBS	130.	1713.	5798.	12923.		23489.	23489. 37831.	23489. 37831. 56236.	23489. 37831. 56236. 78955.	23489. 37831. 56236. 78955. 106209.	23489. 37831. 56236. 78955. 106209. 138198.	23489. 37831. 56236. 78955. 106209. 138198. 175102.	23489. 37831. 56236. 78955. 106209. 138198. 175102. 217079.	23489. 37831. 56236. 78955. 106209. 138198. 175102. 217079. 264273.	23489. 37831. 56236. 78955. 106209. 138198. 175102. 217079. 264273. 316811.	23489. 37831. 56236. 78955. 106209. 138198. 175102. 217079. 217079. 264273. 316811. 374806.	23489. 37831. 56236. 78955. 106209. 175102. 175102. 175102. 217079. 217079. 218713. 3168113. 3168113. 3168357.	23489. 37831. 56236. 78955. 106209. 1351028. 1751028. 1751028. 1751028. 217079. 217070. 216800. 217070. 216800. 217070. 216800. 217070. 216800. 217070. 216800. 217070. 216800. 217070. 216800. 217070. 216800. 216800. 217070. 216800. 217070. 216800. 217070. 217070. 217070. 217070. 217070. 218000. 218000. 217070. 21800. 218000.	23489. 37831. 56236. 78955. 106209. 138198. 1351020. 1751020. 1751020. 1751020. 1751020. 316811. 374806. 374806. 507548. 582452. 582452.	23489. 37831. 56236. 78955. 106209. 138198. 1751020. 175102. 175102. 217079. 217079. 217079. 217079. 316811. 374806. 374800. 3	23489. 37831. 56236. 78955. 106209. 138198. 175102. 175102. 264213. 374806. 374806. 374806. 374806. 5075457. 5075457. 582452. 582452. 749622.	23489. 37831. 56236. 78955. 106209. 138198. 175102. 175102. 264213. 374806. 841966. 841966.	23489. 37831. 56236. 78955. 106209. 138198. 175102. 175102. 175102. 264213. 374806. 841966. 663128. 663128. 749622. 841966. 940182.
UTONITI	PRIME	FΤ	2.4504	2.6341	2.8179	3.0017	3 1055	CC01.C	3.3692	3.5530 3.5530	3.1692 3.5530 3.7368	3.1000 3.3692 3.5530 3.7368 3.9206	3.1000 3.5530 3.5530 3.7368 3.9206 4.1043	4.1043 3.5530 3.7368 4.1043 4.2881	4.1043 3.3530 3.7368 3.7368 4.1043 4.2881 4.4719	4.6557	4.15530 3.35530 3.35530 3.75530 3.75530 3.75530 4.1043 4.1043 4.4719 4.6557 4.8395	5.0232 6.0230 5.1092 3.75530 3.75530 3.75530 4.1043 4.2881 4.4719 4.6557 4.6557 5.0232	4.1092 3.3530 3.3530 3.3530 3.75530 4.1043 4.4719 4.4719 4.6557 4.6557 4.6557 4.6557 5.0232 5.2070	5.2070 5.2070 5.2070 5.2070 5.2070 5.2070 5.2070 5.2070	5.2070 5.2746 5.2746 5.27530 5.2276 5.2270 5.2746 5.2746 5.2746	5.1693 3.3692 3.3530 3.75530 4.1043 4.4719 4.6557 4.6557 4.6557 4.6557 5.0232 5.2070 5.2768 5.7583	5.2070 5.7583 5.7583 5.7583 5.7584 5.7583 5.2070 5.7583 5.7583 5.7583 5.7583	5.1259 3.2530 3.25530 3.75530 4.1043 4.4719 4.6557 4.6557 4.8395 5.2070 5.2070 5.27583 5.7583 6.1259	5.1093 3.1693 3.75530 3.75530 4.1043 4.4719 4.6557 4.6557 5.2070 5.2070 5.2070 5.7583 6.1259 6.3097
HISNET		ΕT	2.4504	2.6341	2.8179	3.0017	3.1855		3.3692	3.3692 3.5530	3.3692 3.5530 3.7368	3.3692 3.5530 3.7368 3.9206	3.3692 3.5530 3.7368 3.9206 4.1043	3.3692 3.5530 3.7368 3.9206 4.1043	3.3692 3.5530 3.7558 3.7568 3.9206 4.2881 4.2881 4.2881	3.3692 3.5530 3.7368 3.9206 4.1043 4.2881 4.2881 4.2881 4.6557	3.3692 3.5530 3.7368 3.9206 4.1043 4.2881 4.2881 4.2881 4.8719 4.6557	3.3692 3.5530 3.7368 3.9206 4.1043 4.2881 4.2881 4.4719 4.6557 4.8395 5.0232	3.3692 3.5530 3.5530 4.1043 4.2881 4.4719 4.6557 4.8395 5.0232 5.2070	3.3692 3.5530 3.5530 4.1043 4.1043 4.2881 4.6557 4.8395 5.0232 5.2070 5.3908	3.3692 3.5530 3.5530 4.1043 4.1043 4.2881 4.6557 4.8395 5.0232 5.2070 5.3908 5.3908	3.3692 3.5530 3.5530 4.1043 4.1043 4.2194 4.6557 5.2070 5.232 5.2746 5.7583 5.5783	3.3692 3.5530 3.5530 4.1043 4.1043 4.2066 4.21043 5.20705 5.20702 5.5746 5.5746 5.5746 5.5746	3.3692 3.5530 3.5530 4.1043 4.1043 4.206 4.6557 5.2070 5.232070 5.5746 5.7583 5.7583 6.1259	3.3692 3.5530 3.5530 4.1043 4.1043 4.206 4.4781 4.6557 5.2070 5.2070 5.5746 5.7583 5.7583 6.1259 6.1259
CHUKU		L.	5.4034	5.7636	6.1238	6.4840	6.8442		7.2044	7.2044 7.5646	7.2044 7.5646 7.9249	7.2044 7.5646 7.9249 8.2851	7.2044 7.5646 7.9249 8.2851 8.6453	7.2044 7.5646 7.9249 8.2851 8.2851 8.6453 9.0055	7.2044 7.5646 7.9249 8.2851 8.6453 9.0055 9.3657	7.2044 7.5646 7.9249 8.2851 8.6453 9.0055 9.3657 9.7259	7.2044 7.5646 8.2851 8.6453 9.0055 9.3657 9.7259 10.0861	7.2044 7.5646 7.9249 8.2851 8.6453 9.0055 9.3657 9.7259 10.0861 10.4463	7.2044 7.5646 7.9249 8.2851 8.6453 9.0055 9.3657 9.7259 10.0861 10.4463	7.2044 7.5646 7.9249 8.2851 8.2851 8.6453 9.055 9.3657 9.3657 9.7259 10.4463 10.4463 10.8065 11.1667	7.2044 7.5646 7.9249 8.2851 8.6453 9.655 9.3657 9.7259 10.4465 11.1667 11.5269	7.2044 7.5646 7.9249 8.2851 8.6453 9.0555 9.0055 9.3055 9.3055 9.7259 10.4861 10.4863 11.1667 11.5269 11.8871	7.2044 7.5546 7.9249 8.2851 8.6453 9.0055 9.0055 9.7259 10.0861 11.1556 11.1667 11.5269 11.5269 11.8871 12.2473	7.2044 7.5546 7.9249 8.2851 8.6453 9.0055 9.0055 9.7259 10.0861 11.1506 11.1566 11.15269 11.15269 11.8871 12.2473 12.2473	7.2044 7.5546 7.9249 8.2851 8.6453 9.0055 9.0055 9.7259 10.4463 11.1556 11.1667 11.5269 11.8871 11.8871 11.8871 12.2473 12.2473
DNIM	STATION	FT	50.764	49.495	48.226	46.957	45.688		44.419	44.419 4 3.150	44.419 43.150 41.881	44.419 43.150 41.881 40.611	44.419 43.150 41.881 40.611 39.342	44.419 43.150 41.881 40.611 39.342 38.073	44.419 43.150 41.881 40.611 39.342 38.073 36.804	44.419 43.150 41.881 40.611 39.342 38.073 36.804 35.535	44.419 43.150 41.881 40.611 39.342 38.073 36.804 35.535 34.266	44.419 43.150 40.611 39.342 36.804 35.535 34.266 32.997	44.419 43.150 40.611 39.342 36.804 35.535 34.266 34.266 31.728 31.728	44.419 43.150 40.611 39.342 36.804 35.535 34.266 31.728 31.728 30.459	44.419 43.150 40.611 39.342 36.804 35.535 34.266 31.728 31.728 31.728 30.459 29.190	44.419 43.150 40.611 39.342 36.804 35.535 34.266 34.266 34.266 37.535 37.927 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.728 31.729 31.729 31.729 31.729 31.729 31.729 31.729 31.729 31.729 31.729 31.729 31.729 31.729 31.729 31.729 31.729 31.729 31.729 31.729 31.729 31.729 31.729 31.729 31.729 31.729 31.729 31.729 31.729 31.729 31.729 31.729 31.729 31.729 31.729 31.729 31.729 31.729 31.729 31.729 31.729 31.729 31.729 31.729 31.729 31.729 3	44.419 43.150 40.611 39.342 35.535 34.266 34.535 32.997 31.728 31.728 32.997 31.728 32.997 31.728 32.997 31.728 32.997 31.728 35.535 32.997 32.9190 26.651	44.419 41.881 41.881 40.611 39.342 35.535 34.266 34.535 32.997 31.728 32.979 31.728 32.979 25.382 25.382 25.382	44.419 41.881 41.881 40.611 39.342 35.535 34.266 34.728 32.997 31.728 32.997 31.728 32.997 31.728 32.997 32.997 32.997 32.997 33.920 24.113

0.0484 0.02000 1.8444 0.5760 0.0484 0.02000 1.9112 0.5760	0.0500 0.02000 1.9751 0.5760 0.0515 0.02000 2.0251 0.5760	09/510 10200 2.0300 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.02000 0.0	0.0544 0.02000 2.0342 0.5760 0.0544 0.02000 2.1493 0.5760			09/5.0 DT57.7 D0000 0 D750.0	0.0502 0.02000 2.2977 0.5760	0.0592 0.02000 2.3415 0.5760	0.0603 0.02000 2.3827 0.5760	0.0612 0.02000 2.4177 0.5760	0.0618 0.02000 2.4405 0.5760).0623 0.02000 2.4619 0.5760	0.0628 0.02000 2.4814 0.5760	0.0632 0.02000 2.4980 0.5760			TTO TECOV	IN IN	0.0134 0.0438	INGT WPOD DELTIP BS LBS FT 15.06 3272.93 3.849				CLF1 CLF2 0.00 0.00		B STAMA CGM 02 4.72 53.3		BMT BMG MAX MOMENT 0. 0.3974E+03
0.03960	0.03960	0.03960	0.03960	0.03960				0.03960	0.03960	0.03960	0.03960	0.03960	0.03960	09650.0	0.03960		TURN TTO	FT-LBS	586064.7	WWBOX WV LBS I 1268.58 103			RG 0.000	0D ASUR 9 3596.8		CLAQW CLAQ 16.45 0.		W BMP 0. 0.
44 0.1314 46 0.1314	40 U.1358 43 D.1400	37 0.1440	26 0.1478	11 0.1514	97 0 1548	69 0 1580	41 0 1610	0101.0 1400	70 0.1538	59 U. 1662	66 0.1678	71 0.1693	90/T'0 C/	6/ 0.1723	21 0.1733	N.TW WERCE	acada mun	- 1	5 0.3353	WTOBOX LBS 59.63			L CG 83. 53.255	WIDTH ABC 12.580 1144		TW LIFTT 7414925.	LIFTT - 37312.	BMBL BM 2.
270092. 2.74 270092. 2.74 291769 2.8	519224. 2.92	652423. 3.01	791295. 3.10	935760. 3.19	085725 3 27	241086. 3.36	401775 3 45			/34430. 3.02 004407 3.72	594460. J./U	81.6 .2026CU 30 c crorce		190900. J.94	743631. 4.10	твох	Eth.	7	2.892	X WBDBOX LBS .93 1177.02	SPLAN FT	1450.	WPROP WTAI) 6546. 298	RAT LENGTH 3665 117.830		LIFTB LIF 132. 1669	JETB LIFTW 330. 417435.	BMBW - - 399.
6.8610 1 7 0448 1	7.2286 1	7.4123 1	7.5961 1	1.7799 1	7.9637 2	8.1474 2	8 3312 2	8 5150 5'		0.0700 707 707 707	C700.0	10 7000.6			2.2720 3	OX WROX		1 <i>F</i> I	9058 11.9334	G WSHBC LBS 46.48 31.	E AREA	62	WWING 52804.	CL1 FIN 58.915 9.3		HA DEFLEC 00 -11.50	DEFLEC I 0 -28.75	WSAV(I) 219.
483 6.8610 085 7.0448	687 7.2286	289 7.4123	891 7.5961	493 7.7799	095 7.9637	697 8.1474	299 8.3312	901 8 5150		105 0.0700 105 0 0075	C700'0 COT	309 9.0003	011 0 110	513 0 6177	115 9.8014	CLINTP LB		1 1 1 1 1 1 1 1 1 1 1 1 1 1 1 1 1 1 1	.9.861 9.	END WWIN BS LBS 532.25 90	STRUCTUR FT2	719.	TO WBOD 2181. 8357 4	UME DENSITY 348. 16.2789	AIL CLT 32. 115.47	WEIGHT ALP 152181. 7.	SLFM ALPHA 2.50 17.5	BEND MOMENT -0.3974E+03
20.306 14.0 19.037 14.4	17.768 14.7	16.498 15.1	15.229 15.4	13.960 15.8	12.691 16.2	11.422 16.5	10.153 16.9	R 884 17 2	3.11 E00.0			3 807 18 7	2 538 10 0	1 769 19 19 A	0.000 19.8	CLBOX1 CLINT	바고		24.603 44.603	WSHEAR WB LBS L 71.50 2	CONTROL AREA FT2.	514.99	VEIGHTS W	30DY/PROP VOLI PARAMETERS 9.	EAIL AT PARAMETERS 2(CRUISE PARAMETERS	AANEUVER PARAMETERS	X Y 1.96 2.60

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		0. 0.3060E+05	0. 0.5006E+05	05915. 0.8179E+05	015157. 0.1240E+06	024399. 0.1739E+06	033642. 0.2323E+06	042884. 0.2998E+06	052126. 0.3771E+06	061368. 0.4646E+06	070610. 0.5629E+06	079852. 0.6718E+06	089095. 0.7915E+06	098337. 0.9218E+06	0107579. 0.1063E+07	0116821. 0.1215E+07	0126063. 0.1374E+07	0135306. 0.1547E+07	0144548. 0.1731E+07	0153790. 0.1925E+07	0163032. 0.3277E+29	0172274. 0.2343E+07	0203004. 0.2611E+07	0333668. 0.3071E+07	0522706. 0.3595E+07	0713745. 0.4021E+07	0850414. 0.4188E+07	0896168. 0.4006E+07	0926973. 0.3778E+07	0957778. 0.3557E+07	0988583. 0.3345E+07	01019388. 0.3140E+07	01050193. 0.2943E+07	01080997. 0.2754E+07	01111802. 0.1274E+30	01142607. 0.2397E+07	01173412. 0.2229E+07	01204217. 0.2067E+07	01235022. 0.1912E+07	01265827. 0.1763E+07	01296631. 0.1620E+07	01327436. 0.1482E+07	01358241. 0.1350E+07	01389046. 0.1223E+07	01419851. 0.1101E+07	01450656. 0.9830E+06
 0	. 0	0.	0.	.0	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	.0	0.	-1154.	-8748.	-23094.	-43795.	-70454.	-101961.	-134099.	-166236.	-198374.	-230511.	-262649.	-294786.	-326923.	-359061.	-391198.	-423336.	-455473.	-487611.	-519748.	-551886.	-584023.	-616160.	-648298.	-680435.
.0	.0	0.	.0	.0	.0	.0	.0	0.	0.	0.	.0	0.	0.	.0	0.	.0	0	.0	0.	0.	.0	0.	-20922.	-119059.	-206138.	-177886.	69967.	589028.	1149554.	1710081.	2270608.	2831135.	3391663.	3952190.	4512717.	5073245.	5633772.	6194300.	6754827.	7315354.	7875882.	8436409.	8996936.	9557463.	10117991.	10678518.
24.	47.	80.	123.	176.	241.	318.	407.	509.	624.	753.	896.	1053.	1224.	1409.	1608.	1821.	4895.	5038.	5176.	5307.	5432.	5551.	5662.	5767.	5866.	5957.	6043.	6122.	6195.	6262.	6325.	6382.	6435.	6484.	6529.	6571.	6610.	6647.	6683.	6718.	6752.	6787.	6822.	6860.	6900.	6944.
-7734.	-16806.	-30683.	-50178.	-76054.	-109036.	-149820.	-199073.	-257442.	-325555.	-404015.	-493168.	-593034.	-703614.	-824907.	-956914.	-1099635.	-1253067.	-1417161.	-1591746.	-1776630.	-1971622.	-2176535.	-2391175.	-2615345.	-2848849.	-3091491.	-3343067.	-3603380.	-3872219.	-4149384.	-4434665.	-4727849.	-5028724.	-5337075.	-5652680.	-5975320.	-6304769.	-6640800.	-6983183.	-7331679.	-7686053.	-8046055.	-8411446.	-8781966.	-9157360.	-9537363.
1417.	2309.	3372.	4595.	5970.	7489.	9147.	10939.	12860.	14907.	17068.	19250.	21432.	23614.	25797.	27979.	30161.	32342.	34498.	36615.	38694.	40734.	42734.	0.	0.	0.			32660.	30944.	29269.	27638.	26050.	24505.	23005.	21549.	20139.	18775.	17458.	16189.	14967.	13795.	12673.	11602.	10583.	9617.	8/00.
-0.7710E+04	-0.1676E+05	-0.3060E+05	-0.5006E+05	-0.8179E+05	-0.1240E+06	-0.1739E+06	-0.2323E+06	-0.2998E+06	-0.3771E+06	-0.4646E+06	-0.5629E+06	-0.6718E+06	-0.7915E+06	-0.9218E+06	-0.1063E+07	-0.1215E+07	-0.1374E+07	-0.1547E+07	-0.1731E+07	-0.1925E+07	-0.2129E+07	-0.2343E+07	-0.2611E+07	-0.3071E+07	-0.3595E+07	-0.4021E+07	-0.4188E+07	-0.4006E+07	-0.3778E+07	-0.3557E+07	-0.3345E+07	-0.3140E+07	-0.2943E+07	-0.2754E+07	-0.2572E+07	-0.2397E+07	-0.2229E+07	-0.2067E+07	-0.1912E+07	-0.1763E+07	-0.1620E+07	-0.1482E+07	-0.1350E+07	-0.1223E+07	-0.1101E+07	-0.98306+06
3.81	4.21	4.56	4.86	5.12	5.37	5.59	5.80	6.00	6.18	6.29	6.29	6.29	6.29	6.29	6.29	6.29	6.28	6.22	6.17	6.11	6.05	5.99	5.93	5.87	5.81	5.74	5.68	5.61	5.54	5.47	5.40	5.33 1.1	7.25 7.25	5.18	5.10	5.01	4.93	4.84	4.75	4.66	4.56	4.46	4.35	4.24	4.13	4.00
5.89	7.86	9.82	11.78	13.75	15.71	17.67	19.64	21.60	23.57	25.53	27.49	29.46	31.42	33.39	35.35	37.31	39.28	41.24	43.20	45.17	47.13	49.10	51.06	53.02	54.99	56.95	58.91	60.88	62.84	64.81	66.77	68.73 70 70	0/.0/	72.66	74.63	76.59	78.55	80.52	82.48	84.44	86.41	88.37	90.34	92.30	94.26	96.23

0.8697E+06 0.7604E+06 0.5530E+06 0.5530E+06 0.4541E+06 0.4541E+06 0.3580E+06 0.3580E+06 0.3580E+06 0.3582E+06 0.1731E+06 0.8337E+05 0.1254E+05		MAX MOMENT	0.4127E+03	0.2677E+04	0.7993E+04	0.1737E+05	0 5185F+05	0.8447E+05	0.1277E+06	0.1791E+06	0.2391E+06	0.3086E+06	0.3882E+06	0.4785E+06	0.5798E+06	0.6921E+06	0.8156E+06	0.9501E+06	0.1096E+07	0.1252E+07	0.1420E+07	0.1599E+07	0.1789E+07	0.1989E+07	0.2199E+07	0.2420E+07	0.2611E+07	0.3071E+07	0.3595E+07	0.4556E+07	0.6198E+07
-1481460. -1512265. -1543070. -1573875. -1604680. -1635485. -165290. -1697094. -1727899.	CGM 51.1	BMG	.0	0.	. 0			-5878.	-15063.	-24248.	-33433.	-42618.	-51803.	-60988.	-70173.	-79358.	-88543.	-97728.	-106913.	-116098.	-125283.	-134468.	-143653.	-152838.	-162023.	-171208.	158344.	2063313.	4888512.	7745263.	9744895.
0. 0. 0. 0. 0. 0. 17587.	STAMA 6.55	BMT	0.	0.				0.	0.	0.	0.	0	0.	0.	0.	.0	.0	.0	.0	.0	.0	0.	0.	.0	0.	.0	.0	0.	0.	0.	0.
712573. 744710. 776848. 808985. 841123. 873260. 905398. 937535. 969672.	CLAQB 0.02	BMP	0.	0.		c		0.	0.	0.	0.	0.	0.	.0	.0	.0	.0	0.	0.	.0	.0	.0	0.	.0	0.	0	-1192.	-9037.	-23857.	-45242.	-72782.
39045. 99573. 60100. 20627. 81155. 141682. 02209. 23264. 23264. 1	T CLAQW 2. 16.45 FGEAR 184619.	BMW	0.	.0	0 0			0.	0.	.0	0.	.0	.0	.0	0.	0.	.0	.0	.0	.0		.0	0.	0.	0.	0.	5934.	33770.	8469.	0455.	.9845.
993. 112 047. 117 109. 123 178. 129 178. 129 349. 140 454. 146 575. 151 716. 157 716. 157 716. 157	<pre>d LIFT 9. 153033 LIFTT 153032.</pre>	ABL.	0.	-1.		-10.	-16.	-23.	-31.	-41.	-53.	- 66.	-81.	-98.	116.		[59.	.83.	.09.			55.	572.	.06	.06.	21.	.36.	49. 3	.62. 5	74. 5	.851
	B LIFTW -54239 LIFTW -54239.	B								1		,				-						9 1	9	-0		L-	L-	L-	L-	L-	L-
-9921700 -10310093 -10702260 -11097895 -11496694 -11496694 -11898332 -11898332 -12302474 -12302474 -12302474 -13116777 -13526116	: LIFT) -43 LIFTB -43.) BMBW	-412	-2676	173671-	-31697	-51836	-78567	-112639	-154770	-205650	-265948	-336311	-417364	-509462.	-972779-	-726861	-852162	.020888-	./96CETT-	- T234409.	-1463984.	-1644337.	-1835330.	-2036764.	-2248447.	-2470179.	-2701755.	-2942975.	-3193633.	-3453522.
7851. 7053. 6316. 5640. 5028. 4483. 4483. 3613. 3299. 3299. 98.	A DEFLEC 0 -149.10 DEFLEC ******	WSAV (I	219.	712.	2309	3372.	4595.	5970.	7489.	9147.	10939.	12860.	14907.	17068.	19250.	21432.	23614.	.19162	- 61 61 2	. 10105	. 24020	34498. 	36615.	38694.	40734.	42734.	0.	0.	0.	0.	.0
978+06 048+06 508+06 308+06 418+06 418+06 318+06 318+06 318+06 378+05 548+05	T ALPHI . 7.00 ALPHA 7.00	MOMENT	27E+03	77E+04	93E+04 37E+05	71E+05	85E+05	47E+05	77E+06	91E+ 06	91E+06	86E+06	82E+06	85E+06	98E+06	21E+U5	00E+00	01E+06	90E+U/	101970	101902	99E+U/	89E+07	89E+07	99E+07	20E+07	08E+07	45E+06	79E+07	56E+07	98E+07
87 -0.86 73 -0.76 58 -0.65 24 -0.45 24 -0.45 81 -0.35 81 -0.35 21 -0.37 91 -0.26 91 -0.26	WEIGH 109722 SLFM 2.58	BEND		31 -0.26	21 -0.17 21 -0.17	56 -0.31	36 -0.51	12 -0.84	37 -0.12	59 -0.17	30 -0.23	0.0-0.30		29 -0.47	19.0- 62	20 - 07 20 0 - 07		106.0- 63 107.0				CT	-0.I7	I -0.19	15 -0.21	9 -0.24	3 -0.23	37 -0.61	10.19	4 0.45	8 0.61
98.19 3. 100.16 3. 102.12 3. 104.08 3. 106.05 3. 108.01 3. 111.94 2. 111.94 2. 115.87 1.	LANDING PARAMETERS LANDING PARAMETERS	X Y	1.96 2.5		7.86 4	9.82 4.1	11.78 4.1	13.75 5.	15.71 5.	17.67 5.	19.64 5.	21.60 6.1 22 22 22	23.57 6.1	.0 .0.02 .0 .0.02	1.0 49.12							41.24 0.1 42 00 04	43.20 6.1	45.17	47.13 6.(49.10 5.5	51.06 5.5	53.02 5.6	54.99 5.8	56.95 5.7	58.91 5.6

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62.84 5.54	0.6176E+07	30944	-4000157.	-805.	-326058.	-138529.	. 0	10642047.	0.6176E+07
64.81 5.47	0.6029E+07	29265	4286480.	-814.	-485045.	-171729.	.0	10972785.	0.6029E+07
66.77 5.40	0.5873E+07	27635	4581186.	-822.	-644032.	-204928.	.0	11303525.	0.5873E+07
68.73 5.33	0.5708E+07	26050	4884057.	-829.	-803020.	-238127.	.0	11634264.	0.5708E+07
70.70 5.25	0.5536E+07	24505	5194873.	-836.	-962007.	-271326.	.0	11965003.	0.5536E+07
72.66 5.18	0.5356E+07	23005	5513412.	-842.	-1120994.	-304526.	0.	12295742.	0.5356E+07
74.63 5.10	0.5168E+07	21545	5839444.	-848.	-1279982.	-337725.	0.	12626481.	0.5168E+07
76.59 5.01	0.4974E+07	20135	6172745.	-854.	-1438969.	-370924.	.0	12957220.	0.4974E+07
78.55 4.93	0.4772E+07	18775	6513078.	-859.	-1597956.	-404124.	.0	13287959.	0.4772E+07
80.52 4.84	0.4563E+07	17458	16860212.	-864.	-1756944.	-437323.	.0	13618698.	0.4563E+07
82.48 4.75	0.4348E+07	16185	7213907.	-868.	-1915931.	-470522.	0.	13949437.	0.4348E+07
84.44 4.66	0.4127E+07	14967	7573917.	-873.	-2074918.	-503721.	0.	14280176.	0.4127E+07
86.41 4.56	0.3899E+07	13795	7939999.	-877.	-2233906.	-536921.	0.	14610915.	0.3899E+07
88.37 4.46	0.3666E+07	12673	i8311897.	-882.	-2392893.	-570120.	0.	14941654.	0.3666E+07
90.34 4.35	0.3427E+07	11602	8689360.	-886.	-2551880.	-603319.	0.	15272394.	0.3427E+07
92.30 4.24	0.3183E+07	1058	i9072122.	-891.	-2710868.	-636518.	0.	15603133.	0.3183E+07
94.26 4.13	0.2933E+07	9617	9459920.	-897.	-2869855.	-669718.	.0	15933872.	0.2933E+07
96.23 4.00	0.2679E+07	8706	59852 4 77.	-902.	-3028842.	-702917.	0.	16264611.	0.2679E+07
98.19 3.87	0.2421E+07	7851	10249512.	-606-	-3187830.	-736116.	0.	16595349.	0.2421E+07
100.16 3.73	0.2158E+07	705	10650738.	-916.	-3346817.	-769316.	.0	16926088.	0.2158E+07
102.12 3.56	1 0.1892E+07	631(511055862.	-924.	-3505804.	-802515.	0.	17256826.	0.1892E+07
104.08 3.42	0.1622E+07	564()11464569.	-933.	-3664791.	-835714.	0.	17587568.	0.1622E+07
106.05 3.24	0.1348E+07	5026	311876544.	-943.	-3823779.	-868913.	0	17918306.	0.1348E+07
108.01 3.04	0.1072E+07	448	 -12291453. 	-955.	-3982766.	-902113.	0	18249046.	0.1072E+07
109.97 2.81	0.7928E+06	401()12708947.	-968.	-4141753.	-935312.	0.	18579784.	0.7928E+06
111.94 2.54	0.5116E+06	3613	 -13128646. 	-984.	-4300741.	-968511.	0.	18910524.	0.5116E+06
113.90 2.21	0.2287E+06	3295	13550155.	-1003.	-4459728.	-1001710.	0.	19241262.	0.2287E+06
115.87 1.73	0.1426E+04	36	313973018.	-1024.	-4618715.	-1034910.	57091.	19572002.	0.1254E+05
BUMP	WEIGHT AI	LPHA DEI	LEC LIFTE	LIFTW	LIFTT CL	AQW CLAQB	STAMA	CGM	
PARAMETERS	152181.	7.00 -57	,45 -126.	-159141.	159419. 1	6.45 0.02	4.72	53.3	
BUMP	SLFM ALP	IA DEFLI	SC LIFTB	DEFLTW L1	FTT FGE	AR			
PARAMETERS	1.20 7.	.00 -57.4	15 –126.	-159141. 15	9419. 1824	.65.			
Х Х	BEND MOMENT	ISM	V(I) BMBM	BMBL	BMW	BMP	BMT	BMG	MAX MOMENT
1.96 2.60) -0.1923E+03	215	192.	-1.	0.	0.	0.	.0	0.4127E+03
3.93 3.31	1 -0.1247E+04	712	21243.	-4.	0	0.	0.	0.	0.2677E+04
5.89 3.81	l -0.3722E+04	141	13712.	. 6-	.0	.0	0.	.0	0.7993E+04
7.86 4.21	l -0.8085E+04	2305	 -8067. 	-18.	0	0.	0.	.0	0.1737E+05
9.82 4.56	5 -0.1476E+05	3372	214728.	-30.	0.	0.	,	.0	0.3171E+05
11.78 4.86	5 -0.2413E+05	459	i24086.	-47.	0.	0.	0.	.0	0.5185E+05
13.75 5.12	2 0.2939E+05	597(). –36506.	-67.	0.	0.	.0	65966.	0.8447E+05
15.71 5.35	7 0.1166E+06	748	 -52337. 	-92.	0.	0.	0.	169037.	0.1277E+06
17.67 5.55	9 0.2001E+06	914	771913.	-121.	. 0			272109.	0.2001E+06
19.64 5.8() 0.2795E+06	1093	995555.	-155.	0.	.0	.0	375181.	0.2795E+06
21.60 6.00	0.3545E+U6	1286)123572.	-194.	.0	0.	.0	478253.	0.3545E+06

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581324 0 42488+06	00-70575:0 1000700 VBV30V U VOULATOR	787468. 0.5798E+06	800530 0 60318TU	993611 0.8156E+06	1096683 0.9501F+06	1199754 0.1096F+07	1302826 0.1252E+07	1405898 0 1420E+07		1612001 0 12000101	1715113 0 1000E.07	1010100 0 0100000000000000000000000000	1011256 0 2420E+07	1012025200 .021201 2022061 0 26118.02	2203901. 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0.4248E+06	0.4902E+06	0.5504E+06	0.6055E+06	0.6554E+06	0.7002E+06	0.7398E+06	0.7743E+06	0.8026E+06	0.8268E+06	0.8460E+06	0 86038+06	0 8697E+06	0.8744E+06	0 1130E+07	0.2552E+07	0 46098407	0 65915407	0.7789E+07	0 7754E+07		0.7516E+07	0.7516E+07 0.7274E+07	0.7516E+07 0.7274E+07 0.7029E+07	0.7516E+07 0.7274E+07 0.7029E+07 0.6779E+07	0.7516E+07 0.7274E+07 0.7029E+07 0.6779E+07 0.6526E+07	0.7516E+07 0.7274E+07 0.7029E+07 0.6779E+07 0.6526E+07 0.6526E+07	0.7516E+07 0.7274E+07 0.7029E+07 0.6779E+07 0.6526E+07 0.6526E+07 0.6269E+07 0.6009E+07	0.7516E+07 0.7274E+07 0.7029E+07 0.6779E+07 0.6526E+07 0.6526E+07 0.6209E+07 0.5745E+07	0.7516E+07 0.7274E+07 0.7029E+07 0.6779E+07 0.6526E+07 0.6526E+07 0.6509E+07 0.5745E+07 0.5745E+07 0.5478E+07	0.7516E+07 0.7274E+07 0.729E+07 0.6779E+07 0.6526E+07 0.6526E+07 0.6509E+07 0.5745E+07 0.5478E+07 0.5478E+07 0.5208E+07	0.7516E+07 0.7274E+07 0.729E+07 0.6779E+07 0.6526E+07 0.6526E+07 0.6509E+07 0.5745E+07 0.5745E+07 0.5745E+07 0.5208E+07 0.4934E+07	0.75168+07 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0T.0	6.29	6.29	6.29	6.29	6.29	6.29	6.29	6.28	6.22	6.17	e 11 9	4.05 6.05	5,99	5 93	10. J		TD.C	29.5	5.61		5.54	5.54 5.47	5.54 5.47 5.40	5.54 5.47 5.40 5.33	5.54 5.47 5.40 5.33 5.25	5.54 5.47 5.40 5.33 5.18	5.54 5.47 5.47 5.40 5.33 5.18 5.18	5.54 5.47 5.47 5.40 5.33 5.18 5.10 5.01	.01 .01 .01 .01 .01 .01 .01 .01 .01 .01	4.9 4.0 4.0 4.0 4.0 4.0 4.0 4.0 4.0					らいううちょう ううう ううう ううう ううう ううう ううう ううう うう うう うう	らいらうらうでいうか。 いちちつでいうでいた。 ちちちつでした。 ちちちつでいた。 ちちちちつでいた。 ちちちちちちちちちちちちちちちちちちちちちちちちちちちちちちちちちちちち		らいらうらうです。 ちょうでのです。 ちょうで、 ちっし、 ちっし、 ちょうなる。 ちょうないで、 ちょうなる。 ちょうなる。 ちょうなる。 ちょうなる。 ちょうなる。 ちょうなる。 ちょうなる。 ちょうなる。 ちょうなる。 ちょうなる。 ちょうなる。 ちょうなる。 ちょうなる。 ちょうなる。 ちょうなる。 ちょうなる。 ちょうで、 ちょう ちょうで、 ちょう ちょう ちょう ちょう ちょう ちょう ちょう ちょう	らいらららい。 「 い い ら ら ら ら ら ら ら ら ら ら ら ら ら	らららららららららった。 いちちついろののすねねねねねねねねる ちっちっしし。 ちっちっちっちっちゃっちょう。 ちっちっちっちゃっちゃんちょう。 ちっちゃっちゃんちゃんちゃんちゃんちゃんちゃん	らいらいらいらい。 「 「 「 「 「 「 」 」 「 」 」 「 」 」 「 」 」 「 」 「 」 「 」 「 」 「 」 」 「 」 」 」 「 」 」 」 一 一 し 一 一 一 一 一 一 一 一 一 一 一 一 一	らいううういううすすすすすすすすすすすすううでき いすすうこうしつののていいうすうこうののでいす。 すてつうちのつこうすうならららすうつつできっ	らいううううううすすすすすすすすすすすすううでき いすすう。 オーローのので、「」、、、、、、、、、、、、、、、、、、、、、、、、、、、、、、、、、、	らいらうらうらうか。 「ちっちって」 「ちっち」 「 いち」 「 い し い し い 「 い し い し い 「 い し い し い し い し い し い し い し い し い し い	らいらうらうらう 100000000000000000000000000000000	ちちちちちちち ちちょう ゆうゆう ゆう うろう うろう うろう うろう うろう うろう ひろう うろう うろう うろ	らいらららい。 ちょういうでですすすすすすすすすする。 ちょういい。 ちょうででで、 ちょうででで、 ちょうででで、 ちょうででで、 ちょうででで、 ちょうででで、 ちょうででで、 ちょうでで、 ちょうでで、 ちょうでで、 ちょうでで、 ちょうでで、 ちょう ちょうで、 ちょうで、 ちょうで、 ちょうで、 ちょう ちょうで、 ちょうで、 ちょうで、 ちょうで、 ちょうで、 ちょう ちょう ちょうで、 ちょう ちょう ちょう ちょう ちょう ちょう ちょう ちょう
23.57	25.53	27.49	29.46	31.42	33.39	35.35	37.31	39.28	41.24	43.20	45 17	47 13	49.10	51 06	20.12	00 75	26 95	58.91	10005	00.00	62.84	62.84 62.84 64.81	00.00 62.84 64.81 66.77	62.84 62.84 64.81 66.77 68.73	60.00 62.84 64.81 66.77 68.73 70.70	60.00 62.84 64.81 66.77 68.73 70.70 72.66	60.00 62.84 64.81 66.77 68.73 68.73 70.70 72.66 72.66	00.00 62.84 64.81 66.77 68.73 70.70 72.66 74.63 76.59	00.20 62.84 66.77 66.77 68.73 70.70 70.70 72.66 74.63 76.59 78.55	60.50 62.84 66.77 66.77 66.77 70.70 70.70 74.63 74.63 78.55 80.55	6000 6684 6677 6677 6677 7066 7463 7463 7659 8052 8052 82.48	6000 662.84 664.81 664.77 76.73 74.65 74.63 74.63 882.48 882.48 84.44	80.52 66.84 66.77 66.77 66.77 76.73 76.59 76.59 88.48 882.48 882.48 882.48	80.55 64.81 66.77 66.77 66.77 76.59 77.66 77.66 77.65 78.55 88.48 88.37 88.37	6000 662.84 664.81 664.77 76.59 77.66 77.65 74.65 88.73 88.55 88.44 88.37 90.34	90.36 66.88 66.38 66.37 66.37 70.55 71.66 71.55 71.55 88.55 88.55 88.44 88.37 90.34 92.30	90,20 66,20 66,71 66,71 70,700 70,7000 70,7000 70,7000 70,7000 70,7000 70,7000 70,7000 70,7000 70,7000 70,7000 70,7000 70,7000 70,7000 70,7000 70,7000 70,7000 70,7000 70,7000 70,7000 70,700000000	90,20 90,20 90,20 90,20 90,20 90,20 90,30 90,30 90,23 90	90,20 66,20 66,71 66,71 70,700 70,7000 70,7000 70,7000 70,700000000	00.10 00.00 00	60. 60. 60. 60. 60. 60. 60. 60. 60. 60.	60. 60. 60. 60. 60. 60. 60. 60. 60. 60.	60.00 66.18 66.28 66.77 66.77 77.66.77 76.59 88.41 88.41 88.41 88.41 88.37 99.34 99.34 99.34 99.34 99.34 99.37 99.26 100.16 100.05 100.05 100.05	60.00 66.184 66.77 66.77 66.77 66.77 76.59 76.59 88.44 88.44 88.37 98.37 98.37 98.37 98.37 98.37 98.37 98.26 98.37 98.26 98.37 98.26 98.37 98.26 98.37 98.26 98.37 98.08 1002.16 1002.08	600.00 662.84 662.84 665.77 770.70.70 666.77 776.59 882.44 882.44 882.44 992.30 992.30 992.30 992.26 992.30 992.26 1002.12 1002.12 1002.05 1002.05 1008.00 1008.00	600.00 662.84 662.84 665.77 770.70.70 665.77 776.59 882.44 882.45 992.30 992.30 992.30 992.25 992.30 992.12 1002.12 1002.05 1002.05 1008.00 1108.00 1208.000 1208.00000000000000000000	60. 60. 60. 60. 60. 60. 60. 60. 60. 70. 70. 70. 70. 70. 70. 70. 70. 70. 7

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FS.	K 50 4	EFF 0.76000	CK 2.0390	PG 11.25	0 0.62	CF 50E-04	CTHIC 0.000	SAFEFAC 1.500	DEFL 0.000		
ТН	ы	SHELL	ليا -	NIUQ:	GAGE	FRAME	ĹΝ	SECTION	SHELL	FRAME	MAX
i	:	STRESS	÷	HICK	THICK	SPACE		AREA	TWITINU	INTINU	BENDING
⊐ č (2 0	rs1	, , , , , , , , , , , , , , , , , , ,	NT	TIN C	NT NT	ſ		2.1.3/81 0.1007		
		2010 2012	0 TAT 0	0.400	10157	9900.2410	n r	1402.12 34 4979	0.3220 776	0,000	1.AN
		2. CF2	716 0	0400	0 0157	440 7740	۲ ۳	12657 20 1957 20	0.3220	0 0000	1.AN
0.0	000	972.8	910 0	0400	0.0157	247.9446	i m	55.7851	0.3226	0.0001	LAN
0.0	000	1520.10	018 0	0400	0.0157	158.6888	m	65.1837	0.3226	0.0004	LAN
0.0	000	2188.9	026 0	0.0400	0.0157	110.2027	m	74.0269	0.3226	0.0010	LAN
0.0	000	3202.1.	372 0	0400	0.0157	75.3319	m	82.4333	0.3226	0.0023	LAN
0.0	000	4411.5	063 0	0400.0400	0.0157	54.6804	۳	90.4832	0.3226	0.0048	LAN
0.0	000	6364.7	568 0	0400	0.0157	37.8998	en.	98.2337	0.3226	0.0108	BUM
0.0	000	8260.3	418 0	.0400	0.0157	29.2025	£	105.7277	0.3226	0.0196	BUM
0.0	000	9803.4	395 0	.0400	0.0157	24.6060	ς Γ	112.9981	0.3226	0.0295	BUM
0.0	000	11056.4	580 0	0400	0.0157	21.8174	m (120.0712	0.3226	0.0398	BUM
000		12330.3	0 7293 0 834 0	0400	0.0157	14.5633 16 5408	יי רי יי	E162.421 F1FC AC1	0.3226	2160.0	HUM I.AN
0.0	000	17410.1	465 0	0.0400	0.0157	13.8553	n m	124.2313	0.3226	0.1021	LAN
0	0000	20515.2	480 0	0400	0.0157	11.7582	ŝ	124.2313	0.3226	0.1418	LAN
0.0	000	23898.8	1066 C	0.0400	0.0157	10.0935	m	124.2313	0.3226	0.1925	LAN
0.0	000	27560.8	1105 C	0400	0.0157	8.7524	m	124.2313	0.3226	0.2560	LAN
0.0	000	31501.2	734 0	0400	0.0157	7.6576	m I	124.2313	0.3226	0.3344	LAN
0.0	000	17927.9	1453 C	0.0800	0.0392	26.9103	ы С	123.7931	0.6451	0.0540	LAN
	000	20243.4	1434 U	. 0800	0.0392 0.0200	23.4842	Ωu	90000 017T	U. 5451	0.0090	TAN
		2.00000		00800	2650.0	18.2010	י רי	117.2388	0.6451	0.1117	LAN
0.0	000	29879.5	664 0	0.0800	0.0392	16.1464	ۍ ۱	115.0187	0.6451	0.1393	ILAN
0.0	0000	22355.2	173 0	0.1200	0.0589	32.3713	ß	112.7799	0.9677	0.0510	LAN
0.0	000	24604.8	1379 C	0.1200	0.0589	29.4117	ъ	110.5216	0.9677	0.0605	MAN
0.0	0000	29553.9	1219 C	0.1200	0.0589	24.4864	S	108.2432	0.9677	0.0855	MAN
0.0	0000	27191.8	1789 (0.2000	0.0981	44.3557	Ś	105.9439	1.6128	0.0425	BUM
0.	0000	28397.0	1078 C	0.2800	0.1373	59.4627	ഗ	103.6227	2.2579	0.0324	BUM
0.	0000	26702.7	754 0	0.3600	0.1766	81.3027	ы С	101.2787	2.9030	0.0218	BUM
0	0000	27220.9	1453 C	0.3600	0.1766	79.7550	ιΩ I	98.9111	2.9030	0.0221	BUM
0.0	0000	27039.7	020 0	0.3600	0.1766	80.2894	ι Γι	96.5186	2.9030	0.0213	MUR
 0 (0000	26842.1		0.3600	0.1766	80.8807	л I	1001.44	2.9030	0.0204	MUH
	0000	26627.1	102 0	0.3600	0.1766	81.5336	Ωı	91.6544	2.9030	0.0196	MUA
	0000	26394.1	270 (0.3600	0.1766	82.2534	in i	89.1801	2.9030	0.0187	BUM
0.0	0000	26142.0		0.3600	0.1766	83.0464	ກເ	7679.98	2.9030	6/10.0	MUA
		0.0/002			0 1 5 6 0	0616.00 67 063	n v	04C1.40 01 5600	2.9030 7 5005	0/10.0	
		2.51/02 2.84185	13 1 1 1 1 1 1 1 1 1 1 1 1 1 1 1 1 1 1	0075.0	0 1569	67 9058	. .	78.9646	2 5805	0.0216	BUN
0.0	000	28036.1	0 669	0.3200	0.1569	68.8320	, n	76.3216	2.5805	0.0203	BUM
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.0191 BUM	.0178 BUM	.0165 BUM	MUBI 2CIU.	0189 BUM	0170 BUM	.0151 BUM	.0133 BUM	.0183 BUM	.0156 BUM	.0129 BUM	MUE 6110	.0137 BUM	.0192 BUM	0122 BUM	.0148 BUM	0133 BUM	.0004 MAN	0004 NONE														LAME MAX	IITWT BENDING		0000 ILAN	0000 LAN	
2.5805 0.	2.5805 0.	2.5805 0.	0 2082.2	2.2579 0.	2.2579 0.	2.2579 0.	2.2579 0.	1.9354 0.	1.9354 0.	1.9354 0.	1.6128 0.	1.6128 0.	1.2902 0.	1.2902 0.	0.9677 0.	0.6451 0.	0.3226 0.	.0 0775.0														SHELL FF	UNITWT UN	LB/FT2	0.3226 0.	0.3226 0.	
73.6382	70.9119	68.1393	0/15.00 8040 63	59.5061	56.5072	53.4377	50.2898	47.0541	43.7189	40.2695	36.6869	32.9456	29.0100	24.8271	20.3117	15.3064	9.4367	1007.0														SECTION	AREA	SQ FT	21.2041	34.3929	
ŝ	ۍ ۱	ς Γ	റഗ	n vo	ъ	Ŋ	S	9	ъ	ъ	9	ß	9	ъ	S	ŝ	m r	'n														ΓN			m	m	
69.8572	70.9954	72.2635	75 2799	59.0214	60.6034	62.4280	64.5583	49.2857	51.5275	54.3052	40.1830	43.4994	30.8894	35.7777	25.5108	19.0274	58.0676	0/00.00														FRAME	SPACE	NI	3966.2410	991.6824	
0.1569	0.1569	0.1569	0.1569	0.1373	0.1373	0.1373	0.1373	0.1177	0.1177	0.1177	0.0981	0.0981	0.0785	0.0785	0.0589	0.0392	0.0157	1610.0	11T	IGHT	'FT*FT)		5737	2000		860	0000	2860				GAGE	THICK	NI	0.0157	0.0157	
0.3200	0.3200	0.3200	0.3200	0.2800	0.2800	0.2800	0.2800	0.2400	0.2400	0.2400	0.2000	0.2000	0.1600	0.1600	0.1200	0.0800	0.0400	0.400	5	MEI	(LBS/					. 4	0.0	4.2				EQUIV	THICK	IN	0.0400	0.0400	
27624.7090	27181.8398	26704.8340	25634.7813	28609.2949	27862.4805	27048.1484	26155.6289	29366.3047	28088.6504	26651.9238	30015.5234	27727.1777	31237.0234	26969.1016	28367.1660	25355.3828	4154.1777	WEIGHT SUMMARY	WEIGHT	FRACTION			0.0324	0.0675		0.0000	0.0000	0.1010	3587.15	1.00000000	96922266. 4 /861	SHELL	STRESS	ISA	60.8191	243.2463	
0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000.0	0.0000	0.0000	0.0000	0.0000	UCTURAL 1	IGHT	3S)		i	1.71 11	11.05	1/.00	00.00 14 53	0.00	14.53				THIC		NI	0.0000	0.0000	
7811461.500	7401658.500	6987453.000	6146523.000	5720152.500	5290087.500	4856512.500	4419615.000	3979597.500	3536653.500	3090988.500	2642824.500	2192373.000	1739874.000	1285563.000	829707.000	372576.000	18816.750	STF	WEI	(LE			492	3T 3COF	3701	1537	- 	т 1537	e Area. SOF	Ratio	EIGHT	BENDING	MOMENT	FT LBS	619.015	4015.657	
80.5172	82.4810	84.4449 ec 4007	88.3725	90.3364	92.3002	94.2640	96.2279	98.1917	100.1555	102.1194	104.0832	106.0471	108.0109	109.9747	111.9386	113.9024	115.8662	1					SHELL	FRAMES	JONON CER	TOTAT.	VOLPEN	GRANTO	Surface	Volume	BODY W.	FUSE	STAT	FТ	1.9638	3.9277	

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I'AN	LAN	ILAN	BUM	BUM	BUM	BUM	BUM	LAN	LAN	LAN	LAN	LAN	ILAN	LAN	LAN	LAN	LAN	ILAN	LAN	MAN	MAN	BUM	BUM	BUM	BUM	BUM	BUM	BUM	BUM	BUM	BUM	BUM	BUM	BUM	BUM	BUM	BUM	BUM	BUM	BUM	BUM	BUM	BUM	BUM	BUM	BUM
0.0010	0.0023	0.0048	0.0108	0.0196	0.0295	0.0398	0.0512	0.0717	0.1021	0.1418	0.1925	0.2560	0.3344	0.0540	0.0696	0.0887	0.1117	0.1393	0.0510	0.0605	0.0855	0.0425	0.0324	0.0218	0.0221	0.0213	0.0204	0.0196	0.0187	0.0179	0.0170	0.0229	0.0216	0.0203	0.0191	0.0178	0.0165	0.0152	0.0139	0.0189	0.0170	0.0151	0.0133	0.0183	0.0156	0.0129
0.3226	0.3226	0.3226	0.3226	0.3226	0.3226	0.3226	0.3226	0.3226	0.3226	0.3226	0.3226	0.3226	0.3226	0.6451	0.6451	0.6451	0.6451	0.6451	0.9677	0.9677	0.9677	1.6128	2.2579	2.9030	2.9030	2.9030	2.9030	2.9030	2.9030	2.9030	2.9030	2.5805	2.5805	2.5805	2.5805	2.5805	2.5805	2.5805	2.5805	2.2579	2.2579	2.2579	2.2579	1.9354	1.9354	1.9354
1010.11	82.4333	90.4832	98.2337	105.7277	112.9981	120.0712	124.2313	124.2313	124.2313	124.2313	124.2313	124.2313	124.2313	123.7931	121.6254	119.4408	117.2388	115.0187	112.7799	110.5216	108.2432	105.9439	103.6227	101.2787	98.9111	96.5186	94.1001	91.6544	89.1801	86.6757	84.1396	81.5699	78.9646	76.3216	73.6382	70.9119	68.1393	65.3170	62.4408	59.5061	56.5072	53.4377	50.2898	47.0541	43.7189	40.2695
2	m	e	e	m	m	m	m	٣	m	m	m	m	m	ъ	ъ	ъ	ഹ	ŝ	ۍ	ŝ	ъ	ъ	ŝ	S	ŝ	ŝ	ъ	ŝ	ഹ	S	ъ	9	ۍ ۱	'n	ъ	ഹ	ц	S	S	9	Ŋ	S	5	9	S	S
	75.3319	54.6804	37.8998	29.2025	24.6060	21.8174	19.5633	16.5408	13.8553	11.7582	10.0935	8.7524	7.6576	26.9103	23.4842	20.6183	18.2010	16.1464	32.3713	29.4117	24.4864	44.3557	59.4627	81.3027	79.7550	80.2894	80.8807	81.5336	82.2534	83.0464	83.9198	67.0673	67.9058	68.8320	69.8572	70.9954	72.2635	73.6828	75.2799	59.0214	60.6034	62.4280	64.5583	49.2857	51.5275	54.3052
	0.0157	0.0157	0.0157	0.0157	0.0157	0.0157	0.0157	0.0157	0.0157	0.0157	0.0157	0.0157	0.0157	0.0392	0.0392	0.0392	0.0392	0.0392	0.0589	0.0589	0.0589	0.0981	0.1373	0.1766	0.1766	0.1766	0.1766	0.1766	0.1766	0.1766	0.1766	0.1569	0.1569	0.1569	0.1569	0.1569	0.1569	0.1569	0.1569	0.1373	0.1373	0.1373	0.1373	0.1177	0.1177	0.1177
	0.0400	0.0400	0.0400	0.0400	0.0400	0.0400	0.0400	0.0400	0.0400	0.0400	0.0400	0.0400	0.0400	0.0800	0.0800	0.0800	0.0800	0.0800	0.1200	0.1200	0.1200	0.2000	0.2800	0.3600	0.3600	0.3600	0.3600	0.3600	0.3600	0.3600	0.3600	0.3200	0.3200	0.3200	0.3200	0.3200	0.3200	0.3200	0.3200	0.2800	0.2800	0.2800	0.2800	0.2400	0.2400	0.2400
0007 0007	3202.1372	4411.5093	6364.7568	8260.3418	9803.4395	11056.4580	12330.3652	14583.4834	17410.1465	20515.2480	23898.8066	27560.8105	31501.2734	17927.9453	20543.4434	23398.9141	26506.5000	29879.5664	22355.2773	24604.8379	29553.9219	27191.8789	28397.0078	26702.7754	27220.9453	27039.7930	26842.1113	26627.1602	26394.1270	26142.0918	25870.0254	28773.8477	28418.5586	28036.1699	27624.7090	27181.8398	26704.8340	26190.4316	25634.7813	28609.2949	27862.4805	27048.1484	26155.6289	29366.3047	28088.6504	26651.9238
	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000
	126702.141	191600.266	300112.031	419206.312	531729.500	637229.750	735272.000	869627.625	1038184.313	1223344.750	1425109.750	1643478.750	1878452.250	2130582.000	2398660.500	2682994.000	2983286.250	3299240.000	3630565.000	3915888.750	4606577.500	6913951.500	9887048.000	11683106.000	11631395.000	11274519.000	10911652.000	10542948.000	10168552.000	9788623.000	9403313.000	9012783.000	8617191.000	8216694.000	7811461.500	7401658.500	6987453.000	6569014.500	6146523.000	5720152.500	5290087.500	4856512.500	4419615.000	3979597.500	3536653.500	3090988.500
>>> · · · · · ·	13.7468	15.7107	17.6745	19.6383	21.6022	23.5660	25.5298	27.4937	29.4575	31.4213	33.3852	35.3490	37.3128	39.2767	41.2405	43.2043	45.1682	47.1320	49.0958	51.0597	53.0235	54.9873	56.9512	58.9150	60.8788	62.8427	64.8065	66.7703	68.7342	70.6980	72.6618	74.6257	76.5895	18.255	80.5172	82.4810	84.4449	86.4087	88.3725	90.3364	92.3002	94.2640	96.2279	98.1917	100.1555	102.1194

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BUM	BUM	BUM	BUM	BUM	BUM	MAN	NONE
0.0179	0.0137	0.0192	0.0122	0.0148	0.0133	0.0004	0.0004
1.6128	1.6128	1.2902	1.2902	0.9677	0.6451	0.3226	0.3226
36.6869	32.9456	29.0100	24.8271	20.3117	15.3064	9.4367	9.4367
9	S	9	5	5	S	٣	°.
40.1830	43.4994	30.8894	35.7777	25.5108	19.0274	58.0676	58.0676

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NONE

0.3226 0.0004

9.4367 -

1.0832 5.0471	264282 219237	4 .500 3.000	0.0000	30015.5234 27727.1777	0.2000	0.0981
147	1/398/	4 .000 3.000	0.0000	31237.023 4 26969.1016	0.1600	0.0785
186 174	82970	7.000	0.0000	28367.1660 75355 3070	0.1200	0.0589
162 162	1881	6.750	0.0000	4154.1777	0.0400	0.0157
562		0.000	0.0000	0.0000	0.0400	0.0157
		STI	RUCTURAL	WEIGHT SUMMARY		
		WED	IGHT	WEIGHT		LT
		[T]	3S)	FRACTION	MEIC	THE
					(LBS/I	гт*FТ)
ILL		492	27.71	0.0324	1.37	137
NAMES		16	30.11	0.0012	0.0	502
NOP		1026	56.71	0.0675	2.86	521
ប្អ			0.00	0.0000	0.0	000
DTAL		1537	74.53	0.1010	4.28	360
DLPEN			00.00	0.0000	0.0	000
NTO	FI	1537	14.53	0.1010	4.28	360
Jume DY WI	Ratio EIGHT	STF	NUCTURAL	1.00000000 15374.53222656 WEIGHT SUMMARY		
		MEI	CHT	WEIGHT	IND	E
		(LLE	3S) 	FRACTION	WEIG (LBS/F	ЗНТ *Т * FT)
ELL		492	17.71	0.0324	1.37	37
UAMES		18	80.11	90.0563	0.05	02
HOP		1026	6.71	0.0675	2.86	121
ប្ត			0.00	0.0000	0.00	00
TAL		1537	4.53	0.1010	4.28	160
OLPEN.			0.00	0.0000	0.00	00
UNTO	E.,	1537	4.53	0.1010	4.28	160
rface	e Area,	SQF		3587.15		
	Ratio			1.00000000 15374 53232666		
IUI ME	TrenT.			9097776C. #/ FCT		

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Fuselage	Definit	ion	Nacelle D	efinitic	ц	Nacelle	Locatior	
×	R	Area	X-Xnose	ж	Area	×	¥	2
0.00	0.00	0.00	0.00	0.70	1.53	39.31	13.42	-6.64
1.24	1.10	3.78	0.37	0.70	1.53	39.31	-13.42	-6.64
2.47	1.81	10.29	0.37	0.70	1.53			
3.71	2.40	18.17	1.76	0.70	1.53			
4.95	2.92	26.85						
6.19	3.38	35.97						
7.42	3.80	45.27						
8.66	4.17	54.55						
06.6	4.50	63.64						
11.14	4.80	72.40						
12.37	5.07	80.73						
13.61	5.31	88.52						
14.85	5.52	95.69						
16.08	5.70	102.17						
17.32	5.86	107.90						
18.56	5.99	112.83						
19.80	6.10	116.91						
21.03	6.18	120.12						
22.27	6.24	122.43						
23.51	6.28	123.83						
24.74	6.29	124.29						
26.16	6.29	124.29						
27.57	6.29	124.29						
28.98	6.29	124.29						
30.40	6.29	124.29						
31.81	6.29	124.29						
33.22	6.29	124.29						
34.64	6.29	124.29						
36.05	6.29	124.29						
37.46	6.29	124.29						
38.88	6.29	124.29						
42.83	6.28	123.83						
46.77	6.24	122.43						
50.72	6.18	120.12						
54.67	6.10	116.91						
58.62	5.99	112.83						
62.56	5.86	107.90						
66.51	5.70	102.17						
70.46	5.52	95.69						
74.41	5.31	88,52						

78.35	5.07	80.73			
82.30	4.80	72.40			
86.25	4.50	63.64			
90.20	4.17	54.55			
94.14	3.80	45.27			
98.09	3.38	35.97			
102.04	2.92	26.85			
105.99	2.40	18.17			
109.93	1.81	10.29			
113.88	1.10	3.78			
117.83	0.00	0.00			
		Fuselage		Nacelles	- 2
Max. Dia	meter	. 12.580	•••••	1.397	
Fineness	Ratio	. 9.366			
Surface	Area	. 3522.634		7.743	(each)
Uclume		9748 345			

Dimensions of Planar Surfaces (each)

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	Wing	H.Tail	V.Tail	Canard	Units
NUMBER OF SURFACES.	1.0	1.0	1.0	1.0	
PLAN AREA	1450.0	277.5	203.5	0.0	(SQ.FT.)
SURFACE AREA	2923.7	403.1	408.6	0.0	(SQ.FT.)
VOLUME	2064.4	106.0	175.0	0.0	(CU.FT.)
SPAN.	107.339	39.669	15.885	0.000	(FT.)
L.E. SWEEP	23.725	37.176	45.001	0.000	(DEG.)
C/4 SWEEP	20.000	33.400	39.400	0.000	(DEG.)
T.E. SWEEP	7.826	19.921	15.935	0.000	(DEG.)
ASPECT RATIO	7.946	5.670	1.240	0.000	
ROOT CHORD	21.614	10.923	18.485	0.000	(FT.)
ROOT THICKNESS	37.867	11.797	19.964	0.000	(IN.)
ROOT T/C	0.146	0.090	0.090	0.000	
TIP CHORD	5.403	3.069	7.135	0.000	(FT.)
TIP THICKNESS	7.133	3.315	7.706	0.000	(IN.)
TIP T/C	0.110	0.090	060.0	0.000	
TAPER RATIO	0.250	0.281	0.386	0.000	
MEAN AERO CHORD	15.130	7.731	13.648	0.000	(FT.)
					ļ
LE ROOT AT	44.400	NCC . #U1	C#C.YY	0.000	(·
C/4 ROOT AT	47.603	107.281	103.966	0.000	(FT.)
TE ROOT AT	63.814	115.473	117.830	0.000	(FT.)
LE M.A.C. AT	51.635	110.664	106.114	0.000	(FT.)
C/4 M.A.C. AT	55.417	112.597	109.527	0.000	(FT.)
TE M.A.C. AT	66.764	118.395	119.763	0.000	(FT.)
Y M.A.C. AT	21.468	8.062	0.000	0.000	
LE TIP AT	65.787	119.592	115.230	0.000	(FT.)
C/4 TIP AT	67.137	120.359	117.014	0.000	(FT.)
TE TIP AT	71.190	122.661	122.365	0.000	(FT.)
ELEVATION	-6.290	5.032	6.290	0.000	(FT.)
GROMETRIC TOTAL VOLU	IME COEFF	122 0 .	0 076	000 0	
REQUESTED TOTAL VOLU	ME COEFF	. 0.771	0.076	0.000.0	
ACTUAL TOTAL VOLUME	COEFF	0.771	0.076	0.000	
EXTENS	NOIS				
		Strake	e Rear	Extensio	u
Centroid location at		0.0	0 00	.00	1
		c			

	Strake	Rear Exten:
Centroid location at	00.00	0.00
Area	00.00	0.00
Sweep Angle	0.00	0.00
Wetted Area	0.00	0.00
Volume	0.00	0.00

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Mission Fuel Required = 64309. lbs. Extra Fuel Carrying Capability = -9276. lbs. Available Fuel Volume in Wing = 1101. cu.ft.

Aircraft Weight = 152181.000 lbs. Aircraft Volume = 11693.691 cu.ft. Aircraft Density = 13.014 lbs./cu.ft. Actual - Required Fuel Volume = -185.529 cu.ft.

ICASE = 4 (Fineness Ratio Method)

Weight Statement - Transport TRANSPORT

Qmax: Design Load Factor: Ultimate Load Factor: Structure and Material: Wing Equation: Body Equation:	400. 2.50 3.75 Aluminum Skin, Stri Ardema/Chambers WWI Ardema/Chambers PDC	nger NG Analys YL Analys	sis sis			
Component	Pounds K	tilograms	Percent	Slope	Tech	Fixed
Airframe Structure	34950.	15853.	21.02			No
Wing	10315.	4679.	6.20	1.20	1.00	No
Fuselage	15375.	6974.	9.25	06.0	1.00	No
Horizontal Tail (Low	r) 1503.	682.	06.0	1.00	1.00	NO
Vertical Tail	1480.	671.	0.89	1.00	1.00	No
Nacelles	4.	2.	0.00	1.00	1.00	No
Landing Gear	6275.	2846.	3.77	1.00	1.00	No
Propulsion	6546.	2969.	3.94			No
Engines (2)	6546.	2969.	3.94	0.85	1.00	Yes
Fuel System	0.	0.	00.00	1.00	1.00	No
Thrust Reverser	0.	0.	0.00	1.00		No
Fixed Equipment	24555.	11138.	14.77		1.00	No
Hyd & Pneumatic	661.	300.	0.40	1.00		No
Electrical	3891.	1765.	2.34	1.00		No
Avionics	2390.	1084.	1.44	1.00		No
Instrumentation	780.	354.	0.47	1.00		No
De-ice & Air Cond	1634.	741.	0.98	1.00		No
Aux Power System	928.	421.	0.56	1.00		No
Furnish & Eqpt	12439.	5642.	7.48	1.00		No
Seats and Lavatories	6600.	2994.	3.97	1.00		No
Galley	1950.	885.	1.17	1.00		No
Misc Cockpit	234.	106.	0.14	1.00		No
Cabin Finishing	2900.	1315.	1.74	1.00		No
Cabin Emergency Equip	405.	184.	0.24	1.00		No
Cargo Handling	350.	159.	0.21	1.00		No
Flight Controls	1831.	831.	1.10	1.00		No
Empty Weight	66052.	29961.	39.72			

Operating Items	4707.	2135.	2.83	No
Flight Crew (2)	340.	154.	0.20	No
Crew Baggage and Provisions	175.	79.	0.11	No
Flight Attendents (4)	520.	236.	0.31	No
Unusable Fuel and Oil	542.	246.	0.33	No
Passenger Service	3130.	1420.	1.88	NO
Cargo Containers	0.	0.	0.00	No
Operating Weight Empty	70759.	32096.	42.55	
Fuel	64029.	29043.	38.50	
Payload	31500.	14288.	18.94	No
Passengers (150)	27000.	12247.	16.24	No
Baggage	4500.	2041.	2.71	No
Cargo	0.	0.	0.00	NO
Calculated Weight	166287.	75428.	100.00	No
Estimated Weight	152181.	69029.		
Percent Error			9.27	

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Appendix D

Approximate Methods of Trajectory Optimization

by

Mark D. Ardema

Approximate Methods of Trajectory Optimization

Mark D. Ardema

Introduction

Application of optimal control theory in the form of the maximum principle to aircraft boundary- Value Problem trajectory optimization problems generally results in a two-point (2PBVP). The order of this problem is double the number of state variables and the equations are always "half unstable." Many schemes have been developed to numerically solve this difficult class of problem, but all are unsuitable in a vehicle synthesis code. Not only are they computationally expensive, but they are non-robust and not user-friendly.

What is needed in a vehicle synthesis code is a method that optimizes the trajectory in one pass, that is as an integral part of the trajectory integration. The method must also be robust and it should be easy to use and to interpret physically. The key to achieving this is to use judicious approximations to reduce the <u>functional</u> optimization problem to a <u>function</u> one.

In this report, two approximation techniques are reviewed and developed. The first is the use of the Energy State Approximation (ESA). This well-known technique substitutes the total mechanical energy for the speed as a state variable, and then neglects the altitude and flight path dynamics relative to the energy dynamics. The second technique is the use of Singular Perturbation Theory (SPT) to time-scale decouple equations of motion. These two techniques are related, and in fact, the ESA may be viewed as an example of SPT methods.

Trajectory Optimization: the Maximum Principle

The equations of motion of aircraft flight, no matter what the assumptions (see Appendix A), are of state equation form:

$$\underline{\dot{X}} = \underline{f}(\underline{X}, \underline{U})$$

where $\underline{X} \in \Re^{m}$ is the state and $\underline{U} \in \bigcup C \Re^{m}$ is the control vector. Suitable boundary conditions on the state vector components are prescribed. It is desired to find the components of \underline{U} along the trajectory so that

$$J = \int_{o}^{t_{f}} fo(\underline{X}, \ \underline{U}) dt$$

is minimized. It is assumed that time is free. The necessary conditions for optimal control are provided by the Maximum Principle (MP).

Theorem (the Maximum Principle): Introduce the variational Hamiltonian function

$$\mathbf{H} = -f\mathbf{o} + \sum_{i=1}^{a} \lambda_{i} f_{i}$$

where the components of the adjoint vector, λ_i , satisfy the differential equations

$$\dot{\lambda}_i = -\frac{\partial H}{\partial X_i}; \ i = 1, .., n$$

Then, if \underline{U} is an optimal control,

(a) $\underline{U} = \arg \max H$ $\underline{U} \in \bigcup$

(b) H = O

(c) Transversality conditions ("natural" boundary conditions on the λ_i) hold.

Thus, we must solve a 2n dimension 2PBVP in the states and adjoints; exactly n

boundary conditions are provided at t = O and the other half at $t = t_f$ (due to the travsversality conditions). The equations are unstable in the sense that if they are linearized about a nominal trajectory, one-half of the eigenvalues will have positive real parts and the other half negative (unless they are zero).

Approximation Techniques

Methods of reducing the 2PBVP to a simpler problem will now be developed. These methods focus on order reduction and are motivated by two simple observations.

First, we note that if all componets of \underline{f} , except possibly f_{i} and the function f_{o} are independent of a specific state variable, say X_{i} and the final value of X_{i} is not specified, then the corresponding adjoint is always identically zero and the state equation $\dot{X}_{i} = f_{i}$ drops out of the problem (decouples from the other states). To see this, consider the i^{\pm} adjoint equation and its travsversality condition:

$$\dot{\lambda}_{i} = -\frac{\partial f_{i}}{\partial X_{i}}\lambda_{i} \qquad \lambda_{i}(t_{f}) = 0$$

The only solution to this problem for any finite value of $\partial f_i / \partial X_i$, is $\lambda_i \equiv 0$.

Second, we note that if there is only one state equation, then the necessary conditions can be used to eliminate the adjoint variable and thus the problem reduces from a functional optimization problem to a function one. To see this, consider

$$\dot{X} = f(X,U)$$

$$J = \int_{o}^{t_{f}} fo(X, U) dt$$

We have

$$\mathbf{H} = -fo + \lambda f$$

$$\dot{\lambda} = \frac{\partial fo}{\partial X} - \lambda \frac{\partial f}{\partial X}$$

Applying the MP (assuming for the moment unbounded optimal control exists)

$$\mathbf{H}=-f\mathbf{o}+\lambda \ f=0$$

$$\frac{\partial H}{\partial U} = -\frac{\partial fo}{\partial U} + \lambda \frac{\partial f}{\partial U} = 0$$

Eliminating λ -gives

$$-\frac{\partial fo}{\partial U}f + \frac{\partial f}{\partial U}fo = 0$$

for the optimal control. Alternatively, a direct approach may be used:

$$J = \int f_o dt = \int \frac{fo}{f} dX$$

Thus (fo/f) is to be minimized with respect to U at constant X; this leads directly to the equation for optimal control derived just above from the MP.

SPT provides an organized, mathematical way to view order reduction of differential equations. Consider the initial value system

$$\dot{X} = f(X, Y) \qquad \qquad X(0) = X_{0}$$

$$\varepsilon \dot{Y} = g(X, Y)$$
 (0) = Y₀

where ε is a "small" parameter.

Since ε is small, an approximate system may be expected to be

$$\dot{X}_r = f(X_r, Y_r)$$
$$0 = g(X_r, Y_r)$$

It can be proved that under certain conditions, the solution of this problem is a good approximation to the solution of the original problem, except near t = 0 because the boundary condition $Y(0) = Y_0$ will be generally violated these.

The problem is that Y undergoes a rapid transition from its boundary condition to the approximate solution at t = 0. To analyze this motion, the time scale is stretched by $T = t / \varepsilon$. The resulting equations are called the boundary layer equations

$$\frac{dX}{dT} = \varepsilon f(X,Y)$$
$$\frac{dY}{dT} = g(X,Y)$$

Setting $\varepsilon = 0$ to approximate these equations

$$\frac{dX}{dT} = 0 \Longrightarrow X = const = X_0$$

$$\frac{dY_b}{dT} = g(X_0, Y_b)$$

The solution to this equation approximates the desired solution near t = 0. There are matching techniques to combine these two solutions to give an over-all approximation, if desired. The key observation is that a second order system has been replaced by two first order systems, and each of them reduces to a function optimization problem.

The SPT provides a convenient way to look at the energy state approximation. Define the aircraft energy per unit weight by

$$E = h + \frac{1}{2g}V^2$$

Differentiate and use the state equations in the Appendix

$$\dot{E} = \dot{h} + \frac{V}{g}\dot{V} = \frac{V(T_v - D)}{Mg} = P$$

Where T_{V} is the component of thrust along <u>V</u> and P is the specific excess power. Note that this equation is valid for all three sets of equations given in the Appendix.

Now replace V by E as state variable and use the observation that h and γ are capable of rapid change relative to E. This motivates writing

 $\dot{E} = P$

ε ἡ =····

$$\varepsilon \dot{\gamma} = \cdots$$

Setting $\varepsilon = 0$ then gives an order reduction of the equations of motion by two. We will use this approximation, the ESA, throughout.

This approximation has a long history of successful application in a wide variety of flight trajectory problems. The main drawback is that the variables h and γ may now jump instantaneously at points along the trajectory, as well as at the boundaries. These jumps could be accounted for by boundary layer analysis, but this is not done in this report.

ESA equations of motion for the three cases of interest are given in the Appendix. these equations will now be used as the basis for discussing specific trajectory optimization problems.

Minimum Time/Fuel to Climb

Starting from equations (1)'

 $\dot{m} = -CT \qquad m(0) = m_0$ $\dot{X} = V$ $\dot{E} = P \qquad E(0) = E_0, E(t_f) = E_f$ L = mg

 $J = \int (K_1 + K_2 CT) dt$

Here, the system functions and boundary conditions do not depend on X and thus the equation $\dot{X} = V$ drops out of the problem:

 $\dot{m} = -CT \qquad m(0) = m_0$ $\dot{E} = \mathbf{P} \qquad E(0) = E_0, E(t_f) = E_f$ $J = \int (K_1 + K_2 CT) dt$

with P evaluated at L = mg. The system functions and boundary conditions now depend on both E and M and hence neither state equation uncouples. Thus the MP must be applied and a 2PBVP solved. To reduce the problem to one of function optimization, SPT is used to further reduce the system.

$$\dot{m} = -CT$$

$$\varepsilon \dot{E} = \mathbf{P}$$

Setting $\varepsilon = 0$ gives a single state equation

$$\dot{m} = -CT$$
 $m(0) = m_0$
 $T = D, L = W$

The optimization problem is now (h and E are controls)

$$J = \left(K_1 + K_2 CT\right) t_f$$

With the obvious trivial solution $t_f = 0$. Also, because the system functions do not depend on m and $m(t_f)$ is free, $\lambda_M \equiv 0$.

The boundary layer system for this problem with $\varepsilon = 0$ is simply

$$\dot{E} = P$$

with m = const, so that

$$J = \int (K_1 + K_2 CT) \frac{dE}{P}$$

and the solution reduces to

$$\max_{\mathbf{h}} \left(\frac{\mathbf{P}}{K_1 + K_2 CT} \right) \Big|_{B = const}$$

assuming that $\frac{P}{(K_1 + K_2CT)}$ is positive and *E* is monotonic. This is the well-known energy climb path.

From now on we will assume "slowly varying" mass, that is that m is on a slower time scale than E and thus its state equation may be ignored. It is also assumed that the throttle is fixed.

Fixed Range

This problem is the same except that range is fixed

$$\dot{X} = V \qquad \qquad X(0) = X_0, X(t_f) = X_f$$

$$\dot{E} = \mathbf{P}$$
 $E(0) = E_0, E(t_f) = E_f$

$$J = \int (K_1 + K_2 CT) dt$$

Here, $\lambda_x = const \neq 0$ so that the $\dot{X} = V$ state equation does not uncouple, and we have a 2PBVP. To effect system order reduction, SPT is used.

$$\dot{X} = V$$

$$\varepsilon \dot{E} = P$$

$$H = -K_1 - K_2 CT + \lambda_X V + \lambda_E P$$

$$\dot{\lambda}_X = -\frac{\partial H}{\partial X} = 0 \implies \lambda_X = const$$

$$\varepsilon \dot{\lambda}_{E} = -\frac{\partial H}{\partial E} = K_{2} \frac{\partial (CT)}{\partial E} - \lambda_{x} \frac{\partial V}{\partial E} - \lambda_{z} \frac{\partial P}{\partial E}$$

Setting $\varepsilon = 0$ a problem with a single state is obtained

$$\dot{X} = V$$
 $T=D$
 $H = -K_1 - K_2CT + \lambda_X V$ $L=W$

Applying the MP:

$$\underset{h,E}{\operatorname{Min}} \left(\frac{K_1 + K_2 CT}{V} \right)_{\text{L=W}} = \frac{K_1 + K_2 C_C T_C}{V_C}$$

$$\lambda_x = \frac{K_1 + K_2 C_C T_C}{V_C}$$

This defines a cruise point, characterized by C_c , T_c , and V_c , in the flight envelope. By proper selection of K_1 and K_2 , this point can be made to closely approximate minimum direct-operating-cost cruise.

For minimum time $(K_1 = 1, K_2 = 0)$, the optimum cruise point is given by

$$\frac{Max}{h,E}(V)$$

For minimum fuel consumption $(K_1 = 0, K_2 = 1)$ it is

$$\frac{\operatorname{Max}}{h, E}\left(\frac{V}{CT}\right) = \frac{\operatorname{Max}}{h, E}\left(\frac{V(t_D)}{C}\right)$$

which is the classic Brequet cruise point.

The boundary layer with $\varepsilon = 0$ is

 $\dot{E} = P$

 $\mathbf{H} = -K_1 - K_2 CT + \lambda_X V + \lambda_E \mathbf{P}$

so that the optimal climb flight path is given by

$$\frac{\operatorname{Max}}{h} \left(\frac{P}{K_1 + K_2 CT - \lambda_X V} \right) \Big|_{B = const}$$

with λ_x as given above.

Maximum Turning With No Thrust

We start with (2)' with T=0.

$$\dot{X} = V \cos \chi$$
$$\dot{Y} = V \sin \chi$$
$$\dot{E} = P = -\frac{VD}{mg}$$
$$\dot{\chi} = \frac{L \sin \phi}{mV}$$

$$L\cos\phi = mg$$

.

with

$$J = -\int \dot{\chi} dt = -\int \frac{L\sin\phi}{mV} dt$$

In this case the \dot{X} , \dot{Y} and $\dot{\chi}$ equations all uncouple. Changing to more convenient variables:

$$\dot{E} = -V(B + C\omega^2)$$

$$J = -\int \omega \, f dt$$

where

$$\omega = \frac{\tan \phi}{\tan \phi} , \quad -1 \le \omega \le +1$$

$$\phi_M = \sec^{-1} \left[\min(C_L \text{ limit, load factor limit}) \right]$$

$$f = \frac{g \tan \phi_M}{V}$$

Thus

$$J = \int \frac{\omega f}{V(B + C\omega^2)} dE$$

and the optimal controls are given by

$$\max_{h,\omega}\left(\frac{\omega f}{V(B+CW^2)}\right)\Big|_{E=\cos \omega}$$

which leads to

$$\frac{\partial}{\partial \omega} (\) = 0 \Rightarrow \omega = \pm \sqrt{\frac{B}{C}}$$

$$\omega = \min\left(\sqrt{\frac{B}{C}}, 1\right) \qquad \text{(assuming right hand turn)}$$

$$\frac{\partial}{\partial h} (\) = 0 \Rightarrow$$

$$\left(V\frac{\partial}{\partial h}f + \frac{fg}{V}\right) \left(B + C\omega^2\right) - fV\left(\frac{\partial}{\partial h}B + \frac{\partial}{\partial h}C + \frac{\partial}{\partial h}\omega^2\right) = 0$$

where

$$B = \frac{D_o + D_{LO}}{Mg} , \ C = \frac{D_{LO}V^2 f^2}{Mg^3}$$

so that

$$\sqrt{\frac{B}{C}} = \frac{g}{Vf}\sqrt{1 + \frac{D_o}{D_{LO}}}$$

The search for the optimum h probably should be done numerically.

Next, consider the same problem but using (3)'. Now, the \dot{X} equation uncouples but the \dot{Y} and $\dot{\chi}$ equations do not. The coupling is of two types. First, through the Coriolis terms, which are relatively small and can be ignored. Second, through the centripetal terms, which are large at the start of descent trajectories from orbit. There are two ways to deal with this problem. First, the Coriolis and centripetal terms are ignored. This is justified because what is really sought is turning ability due to banking and these terms mask this. Second, the \dot{E} and $\dot{\chi}$ terms may be decoupled using SPT.

Maximum Cross-Range

Next consider, using (2)'

$$J = -\int \dot{Y} dt = -\int V \sin \chi \, dt$$

$$X = V \cos \chi$$

 $\dot{Y} = V \sin \chi$

 $\dot{E} = P$

$$\dot{\chi} = \frac{L\sin\phi}{mV}$$

$$L\cos\phi = mg$$

As before, the \dot{X} and \dot{Y} equations uncouple but now the $\dot{\chi}$ equation does not. To reduce this to a function optimization problem, further time-scale separation is required. Putting $\dot{\chi}$ on a slower time scale than \dot{E} gives the solution V = 0 and $\lambda_x = 0$.

Using equations (3)' results in the same problems as for maximum turning.

Appendix

Equations of Motion

The following are the aircraft point-mass equations of motion under various approximations.

(1) Flight in a vertical plane over a flat, non-rotating earth; no winds aloft and thrust aligned with velocity.

$$\dot{m} = -CT$$
$$\dot{X} = V \cos\gamma$$
$$\dot{h} = V \sin\gamma$$
$$\dot{V} = \frac{T - D - mg \sin\gamma}{m}$$

$$\dot{\gamma} = \frac{L - mg \cos \gamma}{mV}$$

(2) 3-D flight, otherwise the same as (1).

$$\dot{m} = -CT$$
$$\dot{X} = V \cos \gamma \sin \chi$$

$$Y = V \cos \gamma \cos \chi$$

$$\dot{h} = \sin \gamma$$
$$\dot{V} = \frac{T - D - mg \sin \gamma}{m}$$
$$\dot{\chi} = \frac{L \sin \phi}{mV \cos \gamma}$$
$$\dot{\gamma} = \frac{L \cos \phi - mg \cos \gamma}{mV}$$

(3) 3-D flight over a spherical, rotating earth; no winds aloft, thrust not aligned with velocity, terms in the square of the earth rotation ignored.

$$\dot{m} = -\frac{T}{g_s I_{sP}}$$

$$\dot{V} = \frac{T\cos\beta\,\cos\,(\alpha+\zeta)-D}{m} - g\,\sin\gamma$$

$$\dot{\gamma} = \left(\frac{T\cos\beta\,\sin(\alpha+\zeta)+L}{mV}\right)\cos\phi - \frac{g}{V}\cos\gamma + \frac{V}{r}\cos\gamma + 2\omega\cos\chi\cos\frac{Y}{R}$$

$$\dot{\chi} = \left(\frac{T\cos\beta\sin(\alpha+\zeta)+L}{mV\cos\gamma}\right)\sin\phi - \frac{V}{r}\cos\gamma\cos\chi\tan\frac{Y}{R}$$

$$+2\omega\left(\tan\gamma\sin\chi\cos\frac{Y}{R}-\sin\frac{Y}{R}\right)$$

$$\dot{X} = \frac{VR\cos\gamma\cos\chi}{r\cos\frac{Y}{R}}$$
$$\dot{Y} = \frac{VR\cos\gamma\sin\chi}{r}$$

 $\dot{h} = V \sin \gamma$

.

-

The following are the energy-state approximations of these equations.

(1)'

$$\dot{m} = -CT$$

$$\dot{X} = V$$

$$\dot{E} = \frac{V(T-D)}{mg} = P$$

$$L = mg$$

(2)'

•

$$m = -CI$$
$$\dot{X} = V \cos \chi$$
$$\dot{Y} = V \sin \chi$$
$$\dot{E} = P$$
$$\dot{\chi} = \frac{L \sin \phi}{mV}$$
$$L \cos \phi = mg$$

.

(3)' (with
$$T = 0$$
 and $m = const$)

$$\dot{E} = -\frac{VD}{mg_{s}}$$

$$O = \frac{L}{mV}\cos\phi - \frac{g}{V} + \frac{V}{r} + 2\omega\cos\chi\cos\frac{Y}{R}$$

$$\dot{\chi} = \frac{L}{mV}\sin\phi - \frac{V}{r}\cos\chi\tan\frac{Y}{R} - 2\omega\sin\frac{Y}{R}$$

$$\dot{X} = \frac{VR\cos\chi}{r\cos\frac{\gamma}{R}}$$

$$\dot{Y} = \frac{VRsin \chi}{r}$$