AN MDO EXERCISE USING RESPONSE SURFACE METHODOLOGY: OPTIMAL SHAPE AND COMPOSITE STRUCTURE OF A WING FOR OPTIMAL RANGE

by

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Thesis Supervisor: Assoc. Prof. Melih Papila

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Abstract

Engineering problems of multidisciplinary nature are challenging where design optimization requires effective communication of the disciplines. This communication is typically referred as multidisciplinary design optimization (MDO) framework. One of the strategies in such a framework is to use of approximations within and among the disciplines to facilitate the navigation of information through a discipline A by an expert in discipline B. Response surface methodology (RSM) for instance is an effective way to bridge the information and expertise between the disciplines within the framework to complete an MDO problem.

This thesis makes a demonstration of RSM in an aircraft composite wing design example. Approximation by RSM aims to generate a prediction tool for optimal structural weight which is required to optimize wing exterior planform for maximum performance, here set as the range of the aircraft. Three planform/shape parameters are chosen: wing span, tip and chord length. For each planform there exists an optimal structure to be found by finite element based structural optimization. The structural optimization level for a given planform makes also use of a different kind of approximation associated with the laminated composite materials. Laminates are treated as homogenized through the thickness and equivalent laminate mechanical properties are implemented. In other words, homogenized laminates approach allows using single continuous thickness variables for each assigned laminate domain replacing the ply-by-ply description of the laminated structure within the structural analyses. Comparison of the homogenized laminate approach and ply-by-ply analyses for a reference wing design is also provided and concluded that former can be incorporated into the design optimization cycles.

The MDO framework for the present example is as follows: Wing planforms are described by full factorial DOE. For each configuration/planform: a) LAMDES was used to calculate aerodynamic forces., b) weight optimization of the wing structure subject to displacement and stress constraints was accomplished using MSC Nastran SOL 200 module. Statistical software JMP 7 was then used to construct an RS weight equation. Genetic Algorithm tool of MATLAB was applied for the range optimization.

TEPKİ YÜZEYİ YÖNTEMİ ESASLI BİR MULTİDİSİPLİNER TASARIM ÖRNEĞİ

OPTİMUM MENZİL İÇİN KANAT GEOMETRİSİ VE KOMPOZİT YAPISI

Burcu Atay

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Keywords: MDO, Design of Experiment, Response Surface Methodology, Homogenized Composite Laminates, Equivalent Material Properties

Özet

Multidisipliner yapılı mühendislik problemleri, disiplinlerin etkin iletişimini gerektiren tasarım optimizasyonlarında (MTO) zorlu görevlerdir. Bu iletişim genel anlamda multidisipliner tasarım optimizasyon çerçevesi olarak ifade edilebilir. Böyle bir çerçevede stratejilerden biri, örneğin B disiplininde uzman olan birisinin A disiplinine bilgi akışını kolaylaştırmak için disiplinler içinde ve arasında kullanılabilecek arayüzler kullanılmasıdır. Tepki Yüzeyi Metodolojisi (TYM)

örneğin, MTO problemlerini çözümlemek için çerçeve içindeki disiplinler arasında bilgi köprüsü oluşturmak ve uzman görüşü sunmak amacıyla kullanılan etkin bir yoldur.

Bu tez kompozit uçak kanadı tasarım örneğinde bir TYM uygulaması sunmaktadır. Bu çalışmada uçağın menzilli olarak seçilen; en yüksek performans için kanat dış geometrisini (planform) optimize etmek amacıyla en ideal yapsıal ağırlığı tahmin aracı olarak TYM yaklaşımının kullanılması hedeflenmiştir. Üç planform/şekil parametresi seçilmiştir, bunlar; kanat açıklığı, uç veteri ve kök veteridir. Her bir planform için sonlu elemanlar çözümlemelerine dayanan yapısal optimizasyon ile bulunacak optimum bir yapı mevcuttur. Verilmiş bir planform için yapısal optimizasyon seviyesinde, laminat kompozit malzemelerle ilişkilendirilmiş farklı yaklaşımlar kullanılmıştır. Katmanlı yapı kalınlık boyunca eşlenik homojen malzeme özellikleri ile tanımlanır. Diğer bir değişle, homojenize edilmiş katmanlı yapı yaklaşımı, yapısal analiz içinde laminat yapısının kat kat tanımlanması yerine, tahsis edilmiş alt-laminat yapı taşı için tek bir sürekli kalınlık değişkeninin kullanımına imkan sağlar. Referans kanat tasarımı için homojenize laminat yaklaşım ve kat kat analiz karşılaştırması da yapılmıştır, ayrıca homojenize yaklaşımdan tasarım optimizasyon çevrimi içinde faydalanılmıştır.

Multidisipliner Tasarım Optimizasyon için çerçeve şu şekilde sunulmuştur: Kanat planformu tasarım uzayı tam faktöriyel deney tasarımıyla tanımlanmıştır. Her bir konfigürasyon/planform için aerodinamik kuvvetlerin hesabında LAMDES kullanılmıştır. Ağırlık optimizasyonu MSC Nastrana bağlı SOL 200 modülü kullanılarak tamamlanmıştır. Jmp 7 istatistiksel yazılımı TYM esaslı ağırlık denklemini kurmak için kullanımıştır. MATLAB'in Genetik Algortima aracı menzil optimizasyonu için uygulanmıştur. Yönelim, kalınlık ve tabaka sayısına bağlı kat kat analiz MSC Nastran yapısal analiz modülü ile gerçekleştirilmiştir.

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CHAPTER 1

1. Introduction

Engineering design problems can be very challenging in correlation with their complexity, such as involved modeling and computational tools, in particular if multidisciplinary nature is dominant. Thanks to recognition of the importance of interdisciplinary communication and harmony in design, multidisciplinary optimization (MDO)strategies have matured over the years(as reviewed in Chapter 2).Combining the computational tasks has vital importance in sense of managing time and executing analysis as MDO problems typically engage large number of variables, parameters, and constraints. Aircraft design for instance, involves complex engineering systems entail analyses which consider interactions between a number of disciplines such as aerodynamics, structures, propulsion systems, performance, and so calls for an efficient MDO framework. Multidisciplinary design optimization strategies typically deal with both decomposition approaches which involve firmly coupled disciplines, and data organization methods according to the level of the problem. Organization of coupling simulations of different disciplines could be the key aspect of the MDO problem. A very good example is High-Fidelity Aerostructural Design Optimization, which requires interaction of the disciplines and a structure to make this interaction effective during the design cycles.

One of the indispensible approaches is to make use of surrogate models and approximations in order to ease of communication and data organization among and within the disciplines. Response surface (RS) approximations for instance may filter out numerical noise, and facilitate a convenient representation of the data extracted from discipline A to discipline B. Therefore, even an expert in discipline B who does not have a great level of expertise in the discipline can deal with an MDO problem with the help of RS which enables a useful interface with an optimizer due to their easiness of implementation.

Present study is aiming to demonstrate an application of response surface (RS) methodology as an influential tool in multidisciplinary design optimization (MDO) framework along with a homogenized laminate approach nested within analysis and optimization of composite aerostructures.

A transport wing design is chosen as the case study (defined in Chapter 3). The case study should typically have an optimal planform for aerodynamic characteristics such as minimum drag to facilitate maximum range of the aircraft, and minimal structural weight while being able to sustain the design load cases. In other words, the case study herein involves explicitly two disciplines namely structures and aerodynamics that are coupled for performance objectives. The design of a transport aircraft (Airbus A380, see Figure 10) is considered as a reference in order to ease choosing design conditions and parameters. The wing structure is made of composite materials for which homogenized laminate concept and associated equivalent properties are incorporated within the structural optimization level. This approach can be noted as another type of approximation that can provide easier structural problem formulation as opposed to ply-by-ply laminate description within the structural optimization.

Involved tools can be summarized as follows: Elliptic span load and optimal twist distributions, which are computed by Lamar's wing design program LAMDES are used as input in structural analysis software. MSC Nastran finite element software package, specifically its aeroelasticity and optimization modules are utilized for aerostructural stress analysis and determination of the optimal weight for the wing structure. Models are generated by a Fortran code used by Papila et al.[1] The relation between planform design variables and the optimum structural weight are set by a Response Surface (RS). Overall, the RS enables the integration of optimal structural weight information into planform design and moderate computational efforts in MDO. Range is maximized by MATLAB Optimization Toolbox using also additional RS based weight formula fitted to structural optimization results.

1.1 Motivation

Most of the work for designing an aircraft based on with improvements, and optimization generally takes more time than creative works at conceptual design phase.
How can the designers manage the process to reduce work load and make a framework systematically? Multidisciplinary design optimization offers a pleasant solution to improve this problem. In Background section, it is enlightened how this methodology presents an accomplished result and what different strategies are available.
Multidisciplinary Design Optimization is a process for the systems in which have strong interactions between different disciplines. The interdisciplinary coupling in MDO leads to challenging computational and organizational problems, so these challenges motivate designers to operate with variables from several disciplines in a systematic way. At that point, instead of dealing with the multi objectives simultaneously, response surface methodology may be utilized after a set of experiments. Consequently, it is aimed that this methodology enables the designer to acquire more freedom in design and proceed step by step and easier throughout the conceptual design phase.

1.2 Flow of the Work

Flowchart of the MDO framework implemented within this thesis is given in Figure 1. The details of the modules of this framework are presented in the following chapters.

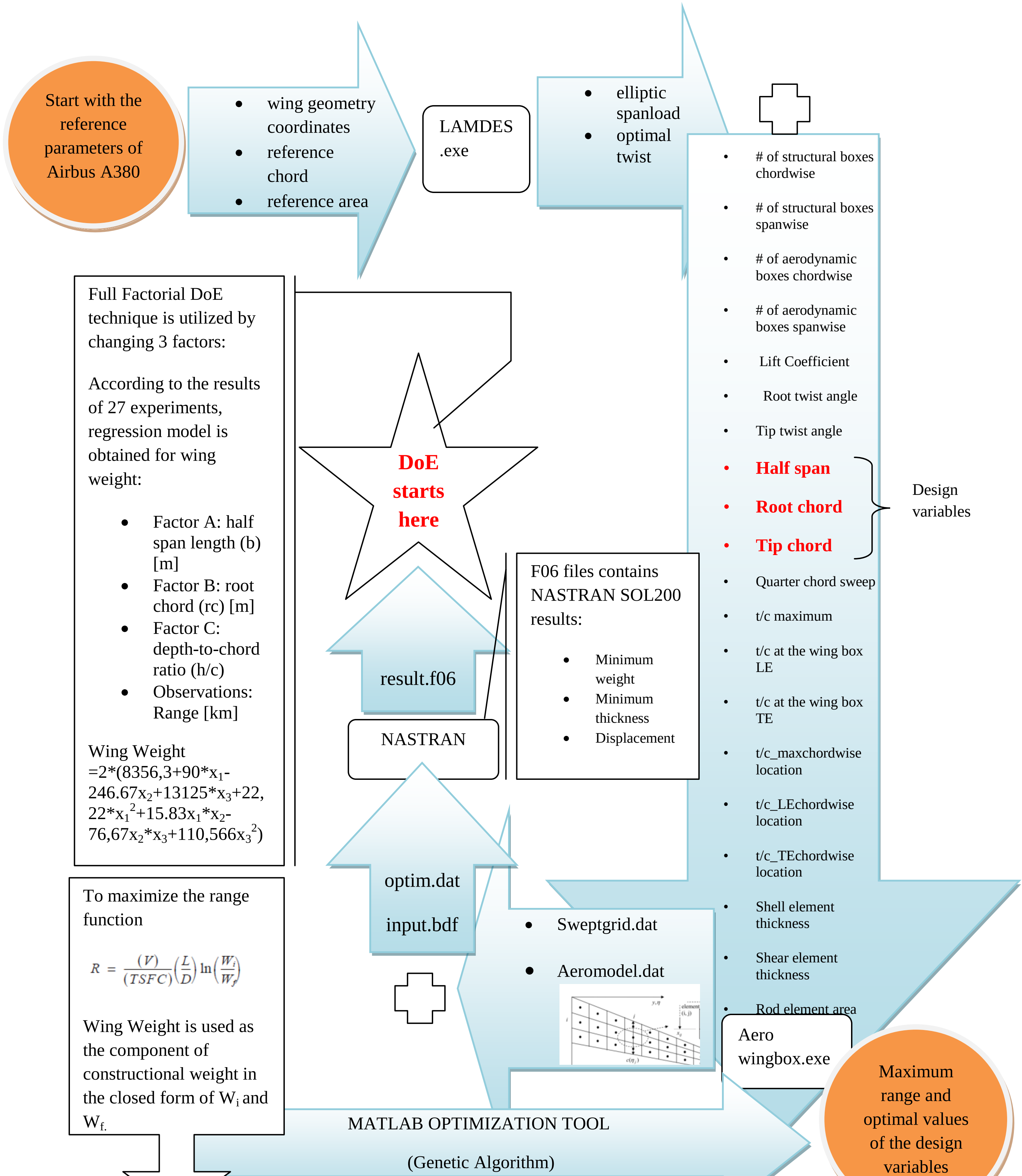


Figure 1. Flowchart of the thesis steps

CHAPTER 2

2. Background

2.1 Multidisciplinary Design Optimization

Multidisciplinary design optimization (MDO) is the application of optimization methods to solve design problems of engineering systems incorporating multiple disciplines. Aircraft design is one of the prime applications because aircraft as a system has many sub systems associated with different disciplines, missions and priorities (see Figure 2).

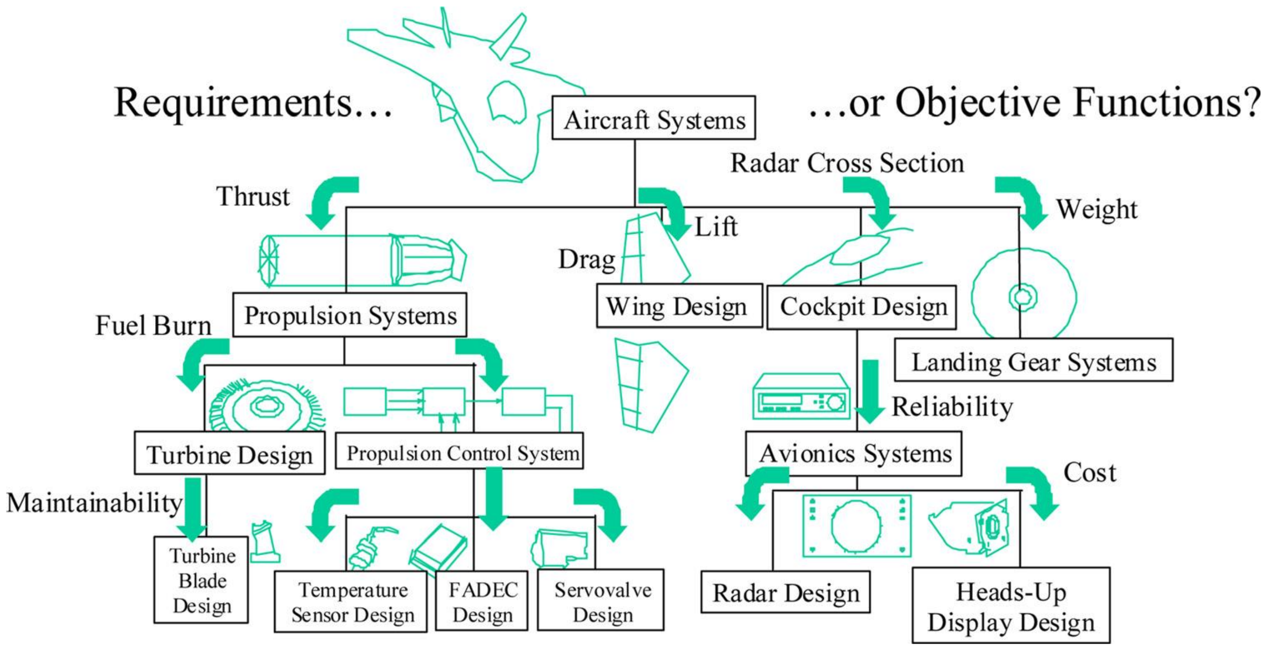


Figure 2. Example of requirements (left) versus objective function (right) flow-down[2]

Disciplines such as aerodynamics, structures, propulsion, controls and stability are tightly coupled and their objectives could conflict with each other. As an example one of the most common tradeoffs can be given to the relation between aerodynamics and structures. Aero-structural optimization entails coupled sensitivities.

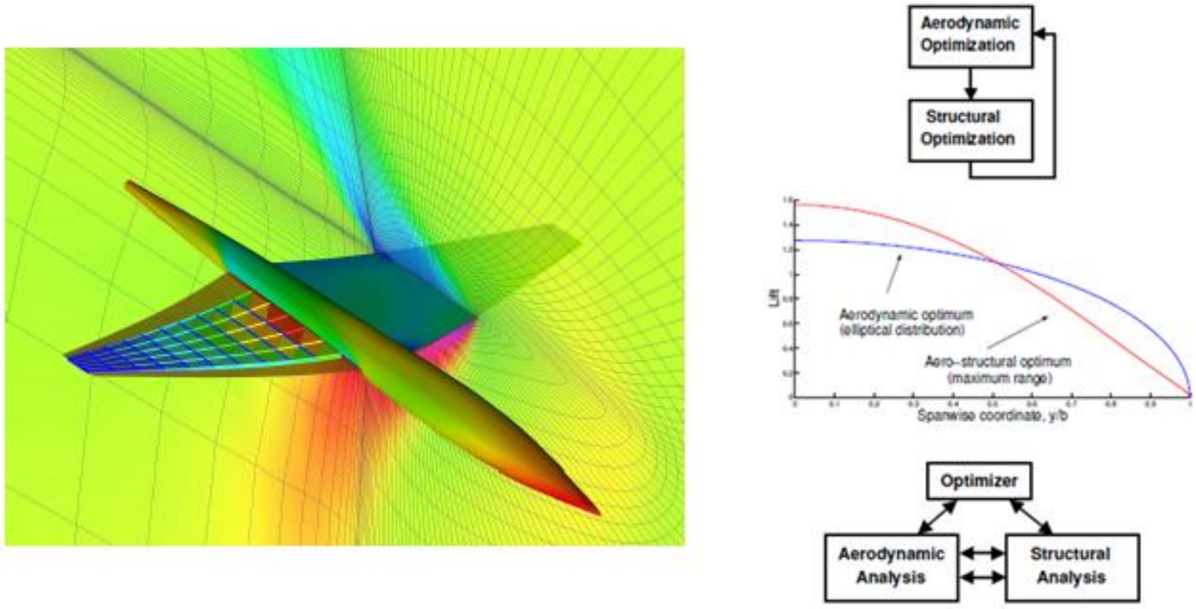


Figure 3. Trade - off between aerodynamics and structural modules.[3]

Herein (Figure 3), sequential optimization does not direct to the real optimum. As trying to solve a design problem in the high fidelity wing optimization, adding structural element sizes to the design variables, will allow larger alterations in the design.

Multidisiplinary Design Optimization has two types based on the statement that a nonlinear objective function does not have to have the same optimum point with the sub disciplinary systems. So, distinction between single level optimization and multilevel optimization is defined[4]. For multilevel optimization case, system design variables are decided by the system optimizer and disciplinary design variables are decided by disciplinary optimizers. For single level optimization case, both system and disciplinary design variables are managed by the system optimizer. Because of the only interaction is between the system and disciplinary it constitutes an important advantage. As well as system level optimization is generally ideal and, it is used to avoid overlooked side effects of a discipline. So, some side effects could be absorbed by another discipline, but may also damage whole system performance. For instance, high aspect ratio is intended for high lift to drag ratio but excessively high aspect ratio lead to flutter which is unfavorable. [5]

Decomposing and integrating multidisciplinary design models are leading key points since each sub module has its own requirements and constraints interacting with other modules. Then problems that affect the process are formulated mathematically and design space is explored. At that point choosing the appropriate strategy for MDO becomes critical. Most well-known strategies are given in Table 1.

Multidisciplinary design problems need some special methods at system levels for gradient based optimization techniques. The collaborative subspace optimization method of Sobieski [6, 7] and the collaborative optimization method developed by Kroo et al. [8]are major prior approaches in the aerospace field. Gillmore and Kelley [9] also improved implicit filtering technique. Another methodology is presented as trust-region methods of Dennis et al. [10] Some direct search methods such as simulated annealing, genetic algorithm and other heuristic optimization methods such as tabu search, particle swarm, ant colony also were revealed.

Sequencing task according to interdisciplinary input and output relation is the fundamental strategy for MDO. Sequential optimization may lead to sub-optimal solution and may not permit parallel execution of analyses. [7] While numerous such methods have been introduced, collaborative optimization (CO) is the most common method that makes parallel execution of decomposed analyses and optimization available (see Figure 4.). The system coordinator adjusts its design variables to both improve the objective function of the system and to ensure that local constraints are satisfied and also preserves disciplinary-level design freedom. Other widespread distributed design method is Concurrent Subspace Optimization (CSSO) which separates the design problem into several discipline subspaces, as each subspace contributes to the task for fulfilling constraint while attempt to decrease a global objective. [11]

The application of approximate models allows smooth design space and Response Surface Methodology in MDO can provide less time consuming operation for the entire multi-level optimization (see left side of Figure 5). RSM can also help to model results of the subspace design problems (as in this thesis work, see right side of Figure 5). In addition, it is appropriate effortlessly to put into the collaborative optimization framework, it limits the model to target variables and makes easier to avoid the 'curse of dimensionality'. [12]

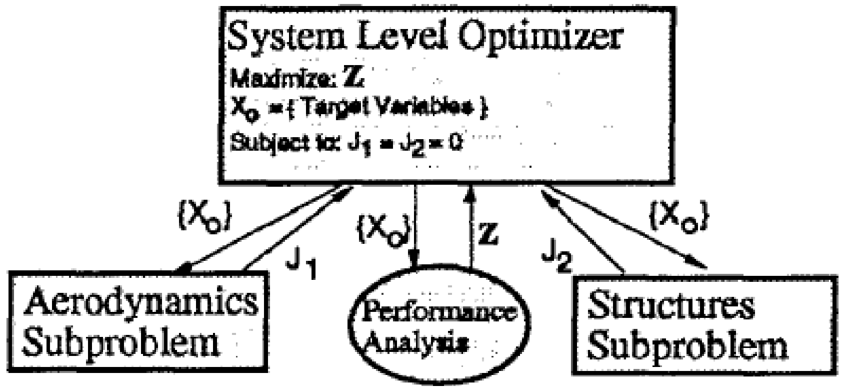


Figure 4.Colloborative Organization of Analyses[7]

Table 1.Overview of MDO Decomposition Frameworks[11]

Methods*	**BLISS**	**CO**	**ATC**	**CSSO**
System- level Analysis Required?	No	No	No	Yes
Subspace Sensitivity Analysis Required?	No	No	No	Yes
Number of Levels	Two	Two	Multiple	Two
Partitioned by:	Discipline Analysis	Discipline Analysis	Object/Compone nt	Discipline Analysis
Subspace optimization influenced by targets?	Yes, indirectly	Yes	Yes	No
Autonomous Subspace Optimizations?	Yes	Yes	Yes	Yes

*BLISS: Bi-Level Integrated System Synthesis, CO: Collaborative Optimization, ATC: Analytical Target Cascading, CSSO: Concurrent Subspace Optimization

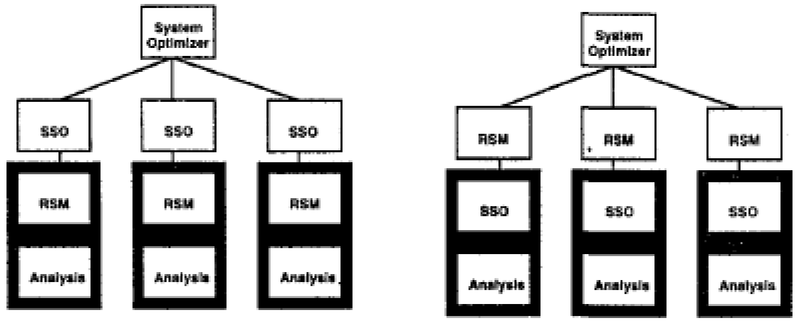


Figure 5. RSM's in Multilevel Optimization[12]

Note that structural optimization of airplane wings to avoid flutter is not necessarily an MDO although flutter is aero-structural phenomenon. Because not only the communication of structures and aerodynamics has to be analyzed, but also the aerodynamic model of the wing needs to be optimized to be categorized as MDO problem. [13]

The Nonlinear Programming (NLP) formalism is commonly recognized. Since the complexity of the MDO system, system analysis generally entails costly nonlinear methods, even if sub disciplines operate linear analysis. For instance, even pressure distributions on airplane wing are calculated by linear aerodynamics, and in that case linear structural analysis is utilized for expected displacements. Still, the relation between pressures and displacements may be nonlinear [11, 13] which can emphasize the need for an efficient MDO framework.

Subsequent to the accomplishment of applications of numerical techniques to structural analysis, Haftka introduced a paper entitled Optimization of flexible wing structures subject to strength and induced drag constraints at the end of 1970's [14]. The Multidisciplinary Analysis and Optimization (MA&O) conference in 1985 was the pioneer conference in about MDO.

After computational fluid dynamics (CFD) started to be applied more consistently, aerodynamic shape optimization came into sight. Airfoil shape analysis and algorithm development for shape optimization were also studied at the end of 1980's. [15, 16] The adjoint based design formulations were coupled with unconstrained optimization algorithms and used for aerodynamic designs of complex airfoils and wings successfully.[17] High-fidelity models of the Euler equations for the aerodynamics and finite element methods for the structural analysis provide opportunity working on both aerodynamics and structures.[18] Shape and structural topology optimizations which try to achieve maximum stiffness (minimum compliance) for a given condition were introduced with the presentation of adaptive mesh-refinement within five years.[19]

Finite element analysis is a common computational tool to execute engineering analysis on mathematical physics, solid mechanics for instance. It incorporates the use of mesh generation techniques by separating a complex problem into small elements with the use of Finite Element Method (FEM) algorithm. FEM is a numerical method which is used to obtain approximate solutions of field problems. The field refers the domain of interest and generally characterizes physical structures. [20]

There is several software which provides Finite Element Analysis practically. MSC Nastran is the prior software presented by NASA is a very effective tool for especially aviation industry with its aeroelastic module. MSC Nastran aerodynamic analysis, similar to structural analysis, is based upon a finite element method. [21]

As a matter of course, optimization has become integral part of the design cycle with the advancing technology, computational resources and increased expectations. Multidisciplinary design optimization with FEM based software is also enabled for aviation applications considering both structural and aerodynamic concerns. Worldwide known aircraft companies such as Fairchild Dornier GmbH and Lockheed Martin Aeronautics Company benefited from this software and published their works using MSC Nastran. [22, 23]

2.2 Wing Structural Design and Optimization Using Response Surface Methodology

Wing design has also an important place for aerodynamic industry because it directly affects performance of the airplane. Geometry influences the lift force on the wing, structural design have an effect on weight. Both geometry and structure has parameters interacting with several disciplines, thus even just wing problem involves MDO concerns. Response Surface Methodology are also beneficial for wing problems and placed studies used for wing design in literature [24-30].

Response Surface Methodology is about characterization of the relationship between a response and a set of quantitative driving factors of experimentation that may be physical or numerical. For this reason, researcher who carries out the experiment may build a model that describes the response over the valid ranges of the factors of interest.[31]As a crucial part of the methodology, a collection of statistical methods providing a systematic way to sample the design space is named Design of Experiments (DOE). Often, DOE is used in the framework of robust design and prior to establishing a formal optimization problem as detecting key drivers among potential design variable, suitable design variable ranges, and feasible objective function values. [32] The response surface fitted to the data then may be utilized to reveal critical characteristics such as optimum operating conditions (factor levels which yield the maximum or minimum expected response), or appropriate tradeoffs if multiple responses exist. [31]

For instance wing weight estimation is an important issue for aviation industry. Formulating wing weight equation, was started with fitting of historical data using tailored expressions with variables raised to various powers. Equations modified to fit historical data were developed based on stress analysis of simple beam models of the wing and fuselage [33-36].

In last three decades, weight equation took an important place with structural optimization within the context of aircraft system level optimization. For instance, Kroo et al.[37] researched aerodynamic-structural design studies of joined-wing aircraft by this attitude. They examined the outcomes of design parameter on drag and structural weight.

From the MDOmethodology point of view, in addition to structural design procedure, aeroelastic load distribution started to be taken into account to estimate the wing weight by aircraft design engineers[38, 39]. Two-level collaborative optimization, which allows the designer to incorporate other disciplines as well structures and aerodynamics, were studied [40, 41]. Then, deficiencies of multi-leveltreatments were detected. For instance, collaborative optimization may cause to ill-conditioning and computational difficulties [42].Furthermore, considerable effort is required for the integration of structural optimization software and local level optimization commonly is not smooth function of the global level design variables. Rohl et al. [43] presented a profitable integration by applying three-level decomposition approach for design of an HSCT wing. Concerns about the integration of global and local level optimization revived Response Surface Methodology. Venkataraman and Haftka [44] , Liu et al. [45] and Ragon et al. [46] also used RS for coordination of the local and global design processes.

Design of experiments theory and response surface modeling are profitable statistical methods and were used in numerous promising aerospace modeling studies [29, 47-53]. A limited number of computational analyses within the design prescribed as design space are performed using the design of experiment techniques. This phase could include finite element method based structural analyses, computational fluid dynamics based aerodynamic analyses or both of them within in an advanced software packages as in this work. Generally more black box part has lots of parameters of which flow of input to output are hard to control, thus arrangement of these flows also need some methodologies to conduct couplings of physical equations, separating modules, such as N square diagram, and sequential algorithms. After a set of experiments, a mathematical model is established which is generally named response surface model. This model can be used in the following calculations throughout the optimization procedure, or can be utilized for sensitivity analyses. Even though setting up a response surface model takes noteworthy time, this cost is sacrificed to avoid computational expense arising from numerical optimization.

2.3 Composites and Homogenized Laminate Approach

Wing structure has been also changed and developed substantially with regards to manufacturing technology and material know-how. Coming up of composite materials to the field of aerospace is likely the foremost advancement in conservative stereotyped wing structure. Even though the conventional wing structure, which is used in World War II, is still valid, some modifications made by choosing some elements of wing such as spars, ribs, stringers and skin through composite materials concerning about weight reduction. [54]

A composite material is the combination of two or more materials to form a better performance than each separate constituent can. Concept of composite materials had been in use for ages and advanced composites have become vital part in aviation. Some of the properties such as strength, stiffness, weight, fatigue life, corrosion and wear resistance incite the usage of composite materials. Particularly in aerospace industry, high strength/weight ratio makes composites more preferable. Currently, fiber reinforced composite materials are indispensible for aerospace companies. Since 1960, which year started many of the US Air Force programs to support aircraft structures made of composites, there had been several stages in progress. Starting with military aircrafts such as F-111 (as horizontal stabilizer, in 1960s), F-14 (as horizontal stabilizer, in 1970s, as in the Figure 6), F-15 (as stabilizer), F-16, commercial airplanes for instance Boeing 767, Antonov An-124, Airbus A310-300 follow the trend. The record-breaking Voyager (which is shown in Figure 7) was also an all-composite airplane which records the first nonstop flight around the world. In the last two decades, confidence in advanced structural composites has been further elevated as characterization and modeling of composite materials have been matured.

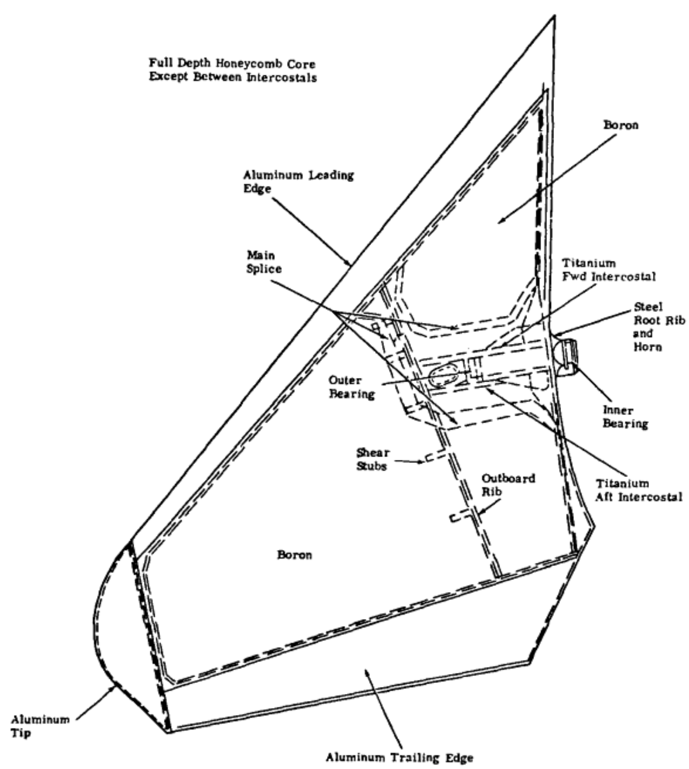


Figure 6. Details of F-14 boron stabilizer [55]

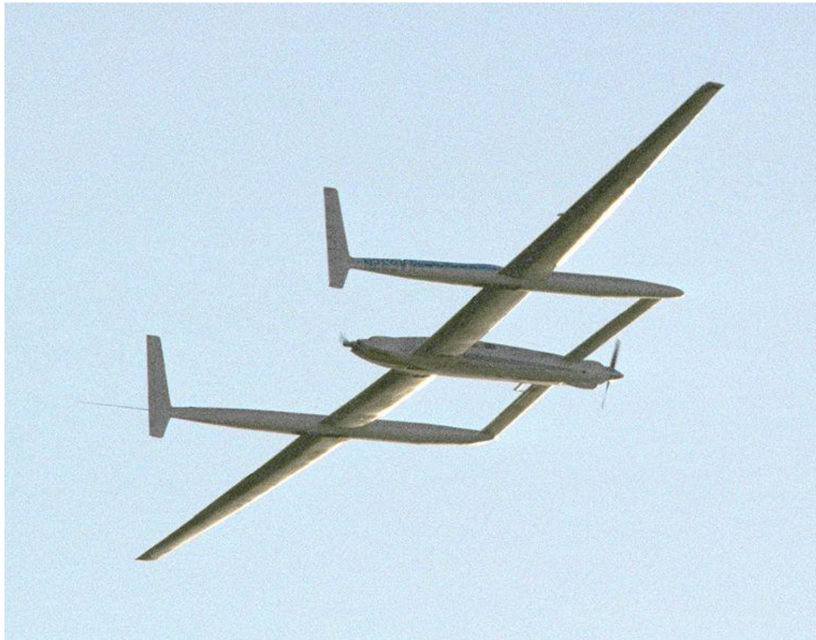


Figure 7. Voyager design by Burt Rutan and his coworkers [56]

Structural module in the framework of overall system can also behave like an MDO problem; especially working with composites is discussed. To ease the burden of optimization at system level, homogenized laminate approach can be a useful treatment for composite application.

2.4 Numerical Simulation Based MDO

Throughout different design phases, MDO approaches could be varied. In Figure 8, design phases and their steps took place.

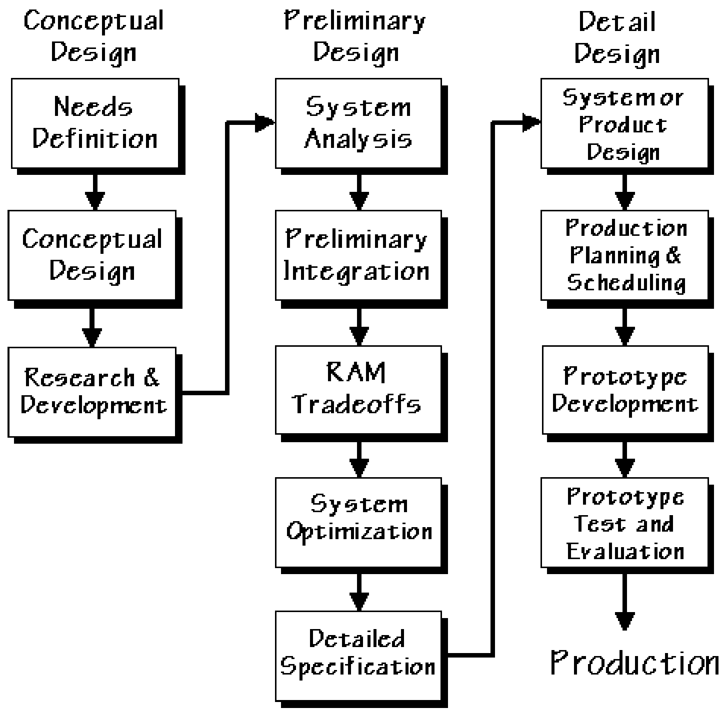


Figure 8. Design Phases [57]

Roth and Crossley [58] presented as for that MDO approaches applications during conceptual design (i.e. conceptual design phase interactions are given in Figure 9) of morphing aircraft and they also revealed inadequacies for sizing needs. One of the approaches which they pointed out is building a database of finite element models using several types and magnitudes of wing shape versions. In order to build the basis for new empirical equations, FEM wing designs could be carried out to deal with strength and displacement concerns. They also presented an alternative instead of empirical approach, and signified that by taking into consideration on bending strength, flexural stiffness and torsional stiffness, a more theoretical formulation could be improved. Obviously, they emphasized that further effort is required by declaring that a combination of these two approaches may serve the purpose.

Ricci and Terraneo [59] reported a morphing aircraft study using MDO techniques for preliminary design phase and identified deficiencies of MDO approaches. They indicate that actually, the majority of the available MDO approaches enable the optimization by keeping the structure configuration as fixed. Despite of the small number of applications which considers different configurations of the internal structural are available, these topological optimization studies keep the exterior geometry fixed. Quite the opposite, aircraft morphing requires not only modeling and optimizing aircraft configurations, but also having the ability taking into account the large change of main geometric properties such as wingspan, wing aspect ratio, wing thickness, swept angle, etc. Advantages of morphing are predominantly examined throughout the conceptual design phase, and occasionally during the preliminary design phase. The notable amount of literature which uses conventional task profile, applies statistical-based or semi-empirical formula models to forecast aircraft performances. But empirical database need to be developed for morphing wings. Additionally, these approaches occasionally do not consider the aeroelastic outcomes which manipulate the structural weight notably. Also, the adoption of more detailed structural models is an achievable option for optimization. For example, through preliminary design phase, finite elements models may be utilized but due to the requirement of checking many different configurations during conceptual design phase, global computational efficiency must be kept high.

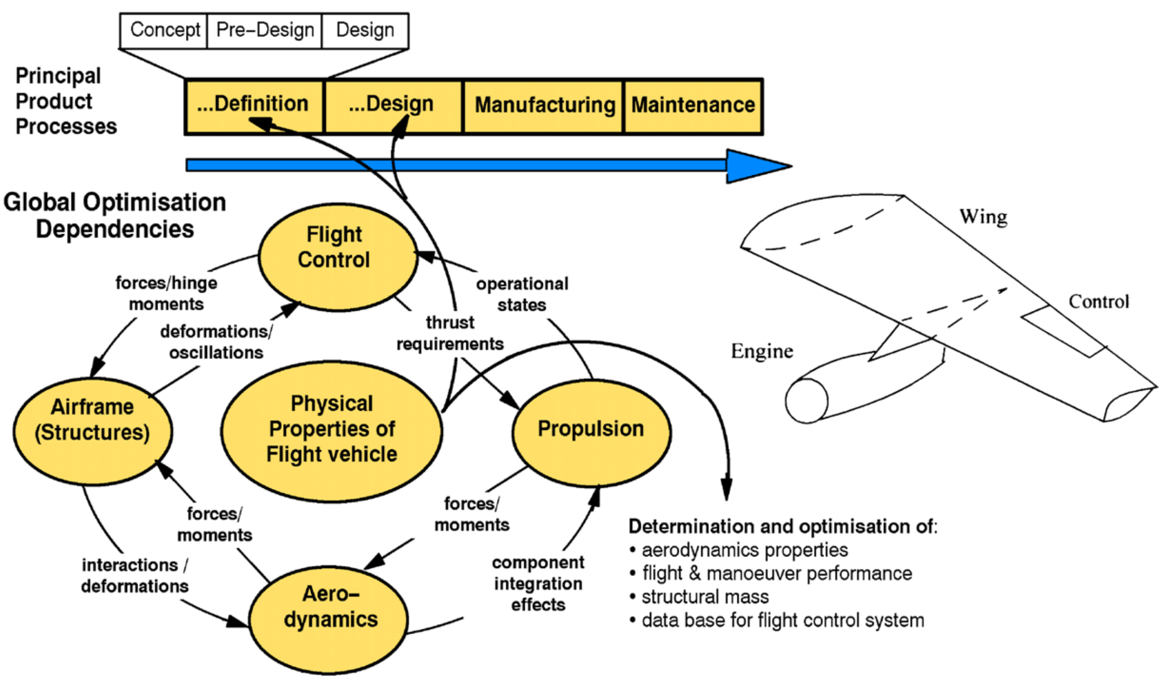


Figure 9. Conceptual design phase interactions [60]

Xia and Friswell [61] also highlighted that morphing aircraft wings has been remarked in current years and foremost concern is the design of the skins in many cases. They explored that equivalent material models which decrease the size of the finite element models; hence the skin may be integrated in the system level model. It is considered that geometric parameters of the skin model may be utilized for optimization at the conceptual design phase when corrugated laminates used for morphing skins. They examined both elastic linear deformation and the nonlinear behavior. A homogenization approach is applied using analytical models and they started out a simplified geometry for a unit-cell which is appropriate for any corrugated shape. They acquired stiffness properties of the original sheet that is well-situated for the optimal design of morphing skin. They validated their approach as demonstrating by the comparison of detailed finite element analysis.

CHAPTER 3

3. Problem Definition

3.1 Design Conditions, Parameters

The reference wing shape is the wing of Airbus A380 which is a double-deck, wide-body, four-engine jet, the world's largest passenger airliner. It is chosen for this study due to availability of dimensions and geometry. It should be noted however, neither the reference wing analyses nor the results from the optimization herein are to compare with the actual A380 design. The models, analyses and design considerations herein are self consistent and representative to allow this exercise, but much simpler than what may have been used in the actual design.

Figure 10. Airbus A 380 commercial airplane [62]

3.2 Wing Planform Definition

The geometric data describing the wing planform is given in

Table 2 along with the sketch in Figure 11.

Table 2. Airbus A 380- like swept-tapered wings for cruise speed of Mach 0.85

Wing span , b (m)	79.8
Root chord, rc (m)	16.3
Tip chord, tc (m)	4.9
Quarter swept angle, $\Lambda_{1/4}$ (º)	34.7
Aspect ratio, AR	7.5
Taper ratio, λ	0.3
Root geometric twist, α (º)	0

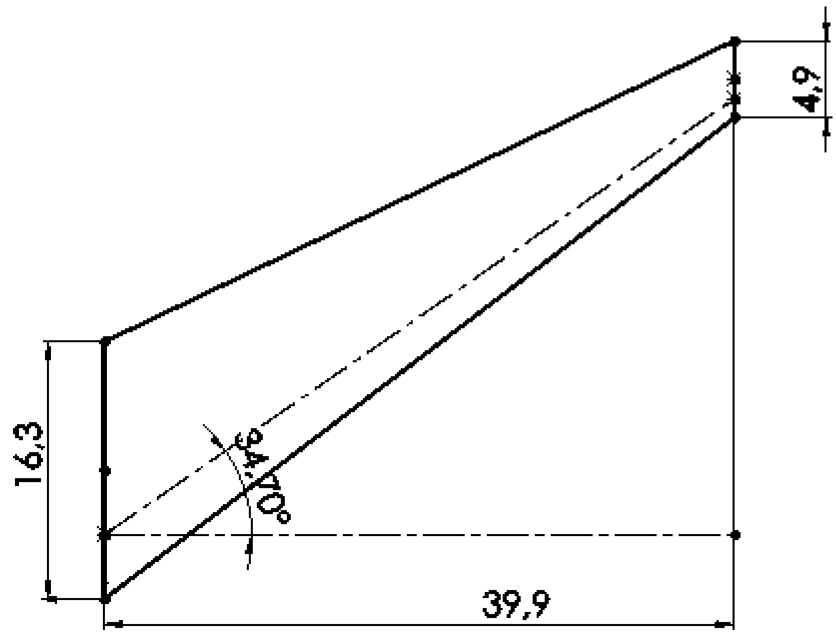


Figure 11.Wing planform geometry

In the present study, optimal planform geometry was aimed. Design objective is maximizing the range by changing wing planform geometric parameters, while minimizing the weight of wing for the given geometry and requirements at the given cruise condition. Planform optimization variables are geometric namely the tip chord ratio, root chord and half span length. To achieve a successful optimization process, planform design space was chosen narrow based on the reference aircraft. Each planform through the design cycles is associated with its optimal structural design for which variables are the structural panel thicknesses.

Wing structural design problem, includes aero-structural model. Structural analysis uses Finite Element Method discretizing the system as nodes and elements. As indicated literature, aerodynamic and structural modules are tightly coupled, so an aerodynamic model was built up to take into account the flexibility of the wing and aerodynamic load distributions. These outputs were used as inputs in structural module.

Within the context of RSM based MDO framework, optimal weight data for the wing models of 27 configurations, i.e. planforms (see section 4.4.1 Design of Experiments) for the three level full factorial design of experiments) were first generated by structural optimizations using MSC Nastran. This allowed generating an optimal wing weight equation as a function of the three planform design variables.

Range was then maximized within a prescribed design space by MATLAB Optimization Toolbox.

From the materials point of view, composite material was used in the skin of the wing box to lighten weight of the wing.

CHAPTER 4

4. MDO Framework

The MDO framework in this study engages three disciplines/branches: aerodynamics, performance and structures as summarized by Figure 12.

Elliptic spanload and optimal twist distributions for a given wing planform were obtained by LAMDES in the aerodynamic optimization phase, Huang et al. [63]. After that, wing box model to be incorporated into aerostructural analyses using MSC Nastran was constructed by a FORTRAN code due to Papila et al. [1]

To optimize the structural weight, the aeroelasticity and optimization modules of MSC Nastran software are used. Static loading was applied and half-span of the wing is fixed at its root section, subsequently stress and tip deflection constraints were defined in the MSC Nastran input file. This file can be viewed in also MSC Patran, visual model of the wing is given in Chapter 5. After MSC Nastran solve the model, optimum structural weight was found for each experiment that is for each wing planform considered in the design of experiment representing the design space.

Three level full factorial experimental design of the three planform variables requires 27 aerostructural optimization runs. The results are optimum structural weights to construct a RS based wing weight equation. The three factors or variables (tip chord, root chord, half span length) and 3 levels for each factor are chosen according to reference wing summarized in

Table 2.

Regression model for the RS was obtained by the software of JMP 7 using Least Square Method. Detailed information about Response Surface Methodology is given in section 4.4.2 Multivariable Regression. Finally, MATLAB Optimtool is used to find maximum range by gradient based optimization technique and genetic algorithm.

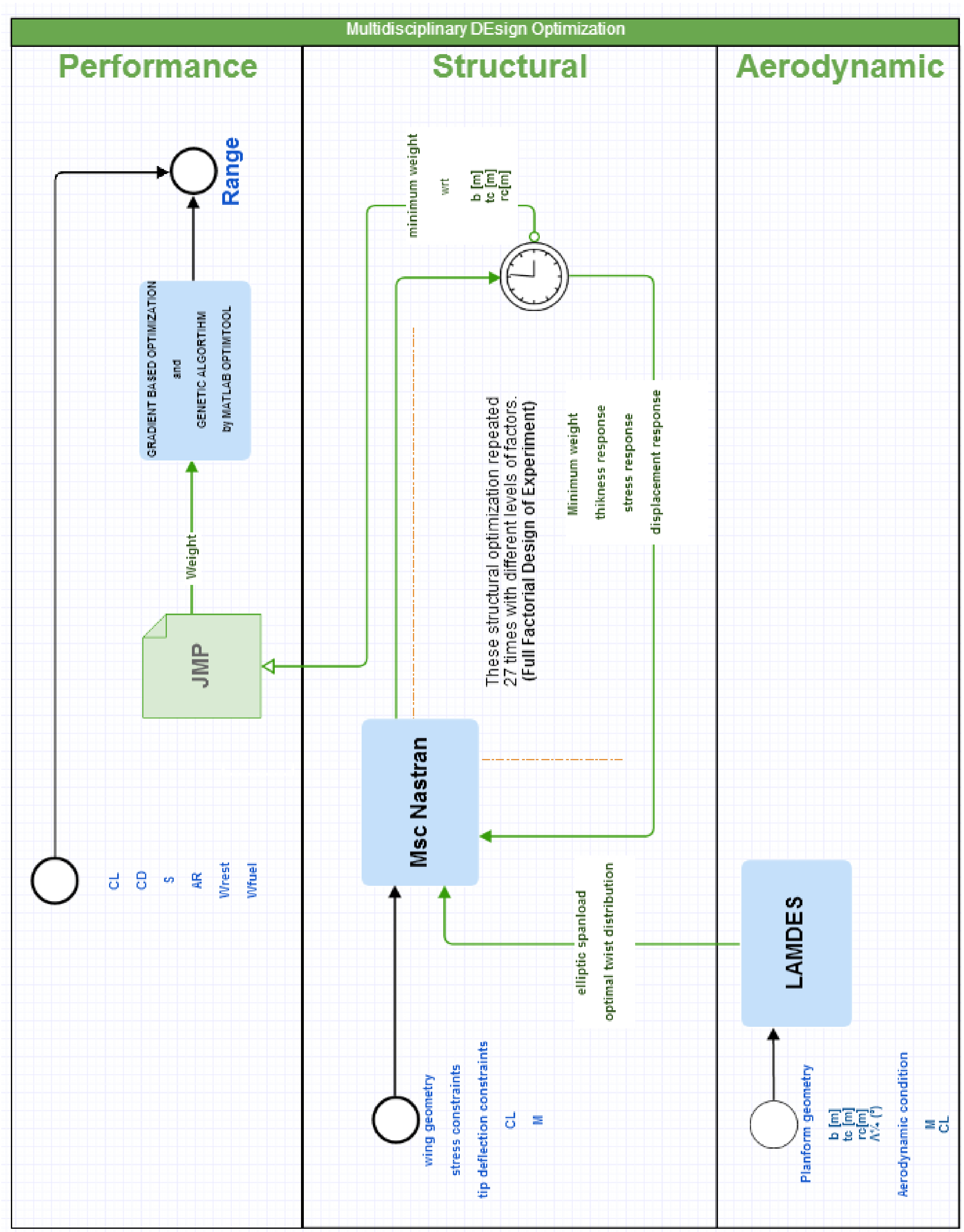


Figure 12. Design structure matrix of MDO system with aerodynamics, performance and weight

4.1 Aerodynamic Module

Aerodynamic forces, which are shown in Figure 13, determine important constants stated in performance section according to the design of the aircraft.

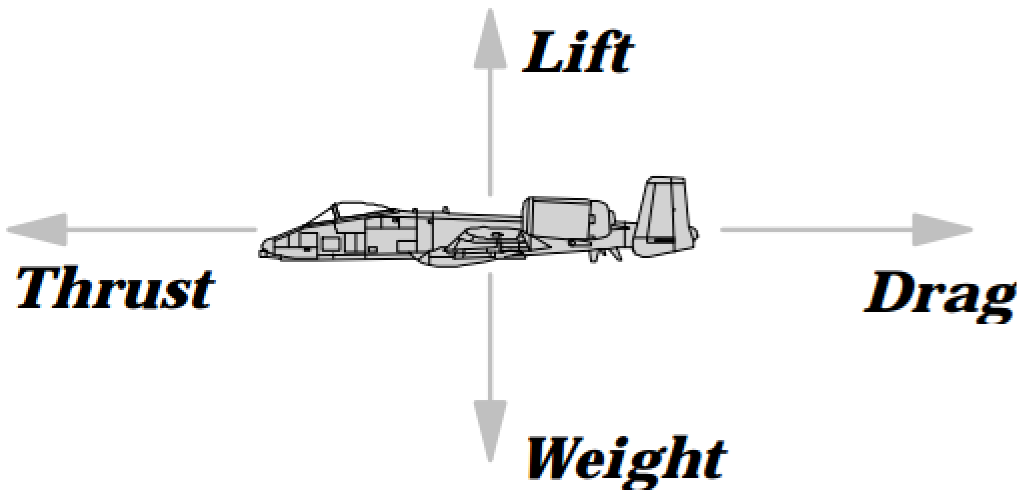


Figure 13. Forces acting on an airplane[64]

Lift is a mechanical aerodynamic force against the weight of the airplane produced by the motion through the air. Lift is generated by all part of the airplane, but most of the lift on an aircraft is generated by the wings.

The drag force is opposed to the aircraft's motion. Aerospace engineers always aim to minimize the drag force by using direct and indirect techniques.

Lift and drag be stated in the simple parabolic form,

$$L = \frac{1}{2}\,\rho C_L V^2 S \tag{1}$$

$$D = \frac{1}{2}\,\rho C_D V^2 S \tag{2}$$

V is the airspeed, S is the wing area (reference area), ρ is the density of the air and C_L and C_D are the lift and drag coefficients (nondimensional).

The functional form of the lift and drag coefficients can be defined as following:

$$C_D = C_{D0} + C_{Di} = C_{D0} + \frac{C_L^2}{\pi ARe} \tag{3}$$

$$C_L = \frac{2mg}{\rho S V^2} \tag{4}$$

The drag and lift drives the aerodynamic design. They consequently determine the amount of power that is needed from propulsion to provide flight at desired speed (as equation indicates that the drag force increases quadratically with the velocity). The amount of power needed, the fuel consumed and overall aircraft weight are related to drag.

The drag coefficient term, C_D is an important parameter which is also used in range calculations.

MSC Nastran aeroelastic module is utilized and the wing separated into 8x50 aerodynamic panels for solution. Optimal twist data and elliptic span load are inputs which are obtained from LAMDES. Using Doublet Lattice subsonic lifting surface MSC Nastran calculates the aerodynamic loads and distributes them for structural analysis.

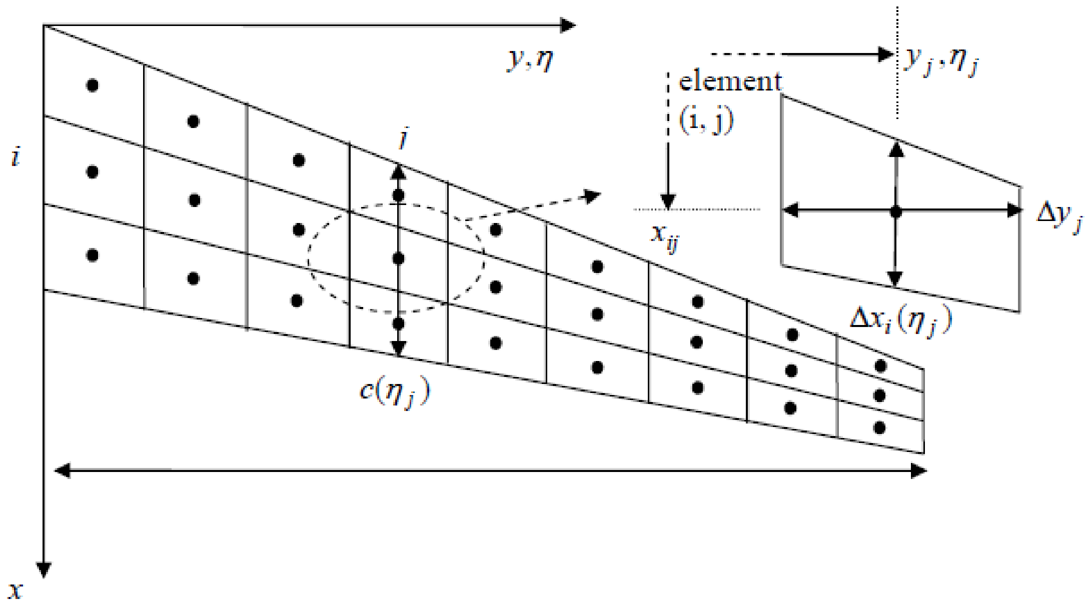


Figure 14. Aerodynamic MSC Nastran model description for the swept-tapered wing

Minimum induced drag

Both structural and aerodynamic models are needed for wing structural design with aeroelastic consideration. Three common methods are generally utilized for wing design and analysis. These are Lifting Line Theory, on the Vortex Lattice Method, and 3D Panel Method. Prandtl's classical lifting line theory (LLT) is generally used for unswept wings.

Craig and Mclean [65] have built up a computer program that optimizes spanloads concerning structural weight. The program works as minimizing a combination of wing drag and weight, so seeks out the optimum twist distribution. They handled a simple beam model to calculate the weight based on bending strength design for a critical condition spanload, and derived wing drag from Trefftz plane induced drag analysis[66] and an empirical profile drag approximation.

Iglesias and Mason[67] developed a method that can help to find out which spanloads obtain the maximum benefit to a specific aircraft design, hence an optimum lift distribution can be found.

They computed lift distributions for minimum induced drag subjected to root bending moment constraint, and presented relation the spanloads changes between wing weight, fuel weight and gross weight for transport aircraft configurations.

Gern et. al[57]described a structural and aeroelastic model for wing sizing and weight calculation of a strut-braced wing taking into account wing flexibility and spanwise redistribution of the aerodynamic loads for the duration of in- flight maneuvers. The aerodynamic loads are computed based on conventional vortex lattice concept (VLM). The wing was pretwisted and jig twisted to achieve an elliptical lift distribution. The pretwist of the wing planform is calculated using Lamar's design program LAMDES[68]. Gern et al.[66] built up wing structures and expanded study of the structural behavior and static aeroelastic response of wing by use of equivalent plate modeling. This study also took account of transverse shear effects based on the first-order shear deformation theory by regarding the wing as a plate. The structural model has been validated for a set of models by MSC Nastran aeroelastic module.

Papila et.al [1] presented a study tailoring wing structures for reduced drag penalty concerning also off-design flight conditions. It is revealed how alters in the flight condition and static aeroelastic response according to both near elliptic spanload and straight-line wrapped surfaces. Structural model is constructed by MSC Nastran and aeroelastic module of the software is used for aerodynamic model. The present study utilized complete structural and aerodynamic model from Papila et. al's paper.

Finding lift distribution over an isolated wing with the minimum induced drag is classical problem of aerodynamics. The Lamar design program is used to obtain the spanload to minimize the sum of the induced and pressure drag as fulfilling a pitching moment constraint. Lamar/Mason optimization code prompts users for the input file of forward swept wing as in Appendix A.

The wing aerodynamics are calculated using vortex lattice method (VLM), that is executed with LAMDES and it is available on Prof. W. H. Mason's homepage "Software for Aerodynamics and Aircraft Design" within Virgina Tech.[69]

4.2 Structural Module

Structural analysis deal with the outcomes of loads on physical structures and comprises the disciplines of applied mechanics, materials science and applied mathematics to work out deformations, internal forces, stresses, support reactions, accelerations, and stability. Verification of analysis with the test results is vital owing to physical test of large-size complex engineering systems costs. Therefore, key part of the engineering design is structural analysis and finite element analysis (FEA).

Finite Element Method design steps are given in Figure 15 and FEM based software are user friendly with their interfaces at the present time. But in this work, generally a parametric study was handled. Nonetheless, visual deformation results are given in Chapter5.

Structural analysis based on FEM typically comprises three fundamental steps:

1. Preprocessing: After a computer aided drawing (CAD) model is constructed, the complete body is divided into small elements, and these elements are connected at discrete points called as nodes. These elements and nodes are generally called as mesh in software applications. Because of this step may be exceedingly time consuming, there are some user- friendly graphical preprocessors for complex structures. Boundary conditions (loads, fixed displacements, etc.) are defined to be ready for processing.

2. Analysis: The model arranged by the preprocessor is transferred to the finite element program, and then system of linear or nonlinear algebraic equations is built and solved. Computers conduct numerical analysis.

$$K_{ij} u_{ij} = f_i \tag{5}$$

where u and f are the displacements and externally applied forces at the nodes. The formation of the K matrix depends on sort of problem being tackled, and this component delineates the method for truss and linear elastic stress analyses

3. Postprocessing: Modern postprocessors presents stress levels on the models with colored scale without the need to make user comb out in the list of displacements and stresses at discrete positions within the model. So, user can see results.[70]

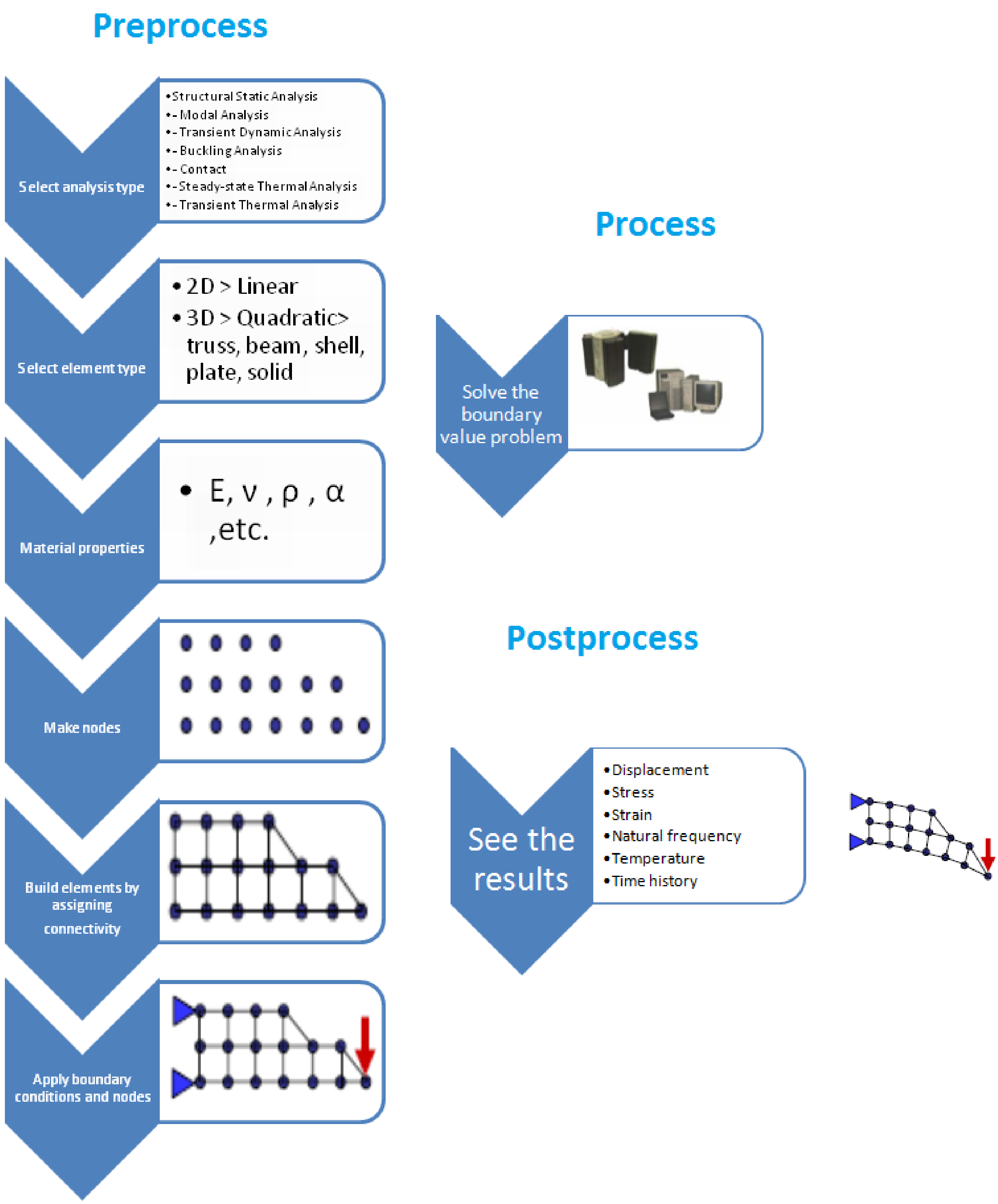


Figure 15. Fundamental steps of FEM [71]

For the structural design in this work, a hexagonal wing box is constructed to be used as an input in MSC Nastran software packages. MSC Nastran input file includes 90 nodes, 270 elements, 2 materials (Aluminum and Carbon/Epoxy NCF).

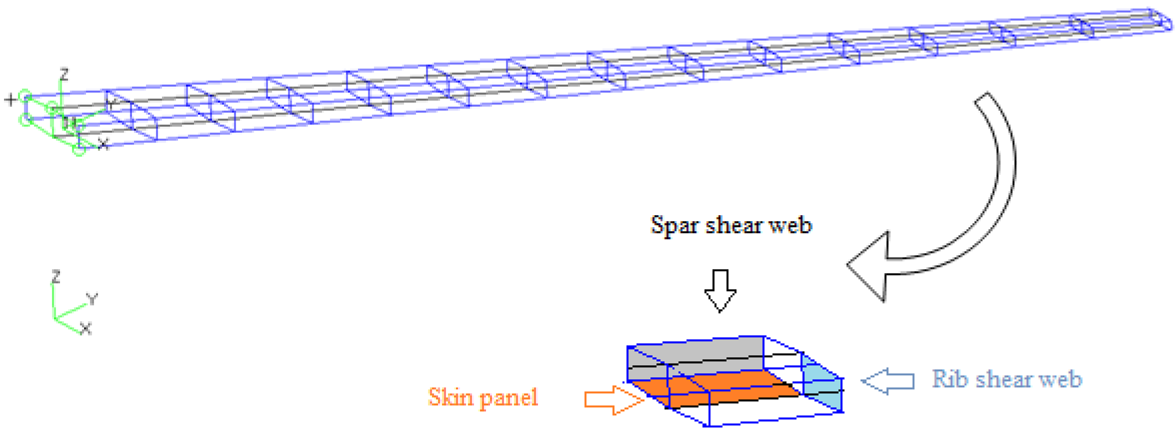


Figure 16. Structural MSC Nastran model for hexagonal wing box of the swept-tapered wing

Wing box includes 14 bays and thickness variables are assigned according to these bays. The upper and lower skins are identified as shell elements. 56 quadrilateral shell elements are used to characterize skins. The spar webs and cap regions are modeled by 14 shear elements and 28 rod elements (14 for upper, 14 for lower). There are totally 30 shear elements to separate ribs. The wing is assumed as fixed at the root as boundary condition.

4.2.1 Finite Element Model (FEM) Based Structural Optimization

Following generic mathematical formula represents the optimization problem:

$$\min_{x} \quad f(x)$$

$$\text{subject to} \quad g_i(x) \le 0 \;\; i = 1, \dots, m$$

$$h_j(x) = 0 \;\; j = 1, \dots, n$$

Objective function is minimization of structural weight, so f(x) is weight of the structural model of the wing. Functions g and h represents the inequality and equality constraints, respectively.

Solution 200 (DESOPT) is the design optimization solution in MSC Nastran structural analysis software. The optimization solution operates the outcomes from various MSC Nastran analysis solutions which are specified in discrete sub cases.[72]

Table 3. List of design variables for wing geometry and structure

Design variables	PSHELL[m] x 14
Initial values	0.05
Lower bound	0.005
Upper bound	0.05

Upper skin and lower skin was defined with SHELL element in MSC Nastran. Thickness of these shell elements were chosen as design variables for each 14 bay. Hence, there are 14 design variables for structural optimization level. Note that these are the structural variables of the sub-level optimization for any planform of interest (by the three geometric wing parameters) which is the goal to be optimized in the MDO framework.

Table 4. List of design constraints for structural optimization: tip displacement and stress allowables (strength parameters) for sublaminate[0/45/-45/0]

Design constraints	Lower bound	Upper bound
y_{max}[m]	-0.15 * b	+0.15 * b
X [Pa]	0	775E6
X’ [Pa]	0	750E6
Y [Pa]	0	378E6
Y’ [Pa]	0	180E6
S [Pa]	0	298E6

$y_{max}$is displacement constraint which is defined at the tip of the wing, and chosen as 15% of half span length.

Stress constraints come from composite material criteria. The modes of failure of composite materials are more complicated than isotropic materials. Other than the different tensile and compressive strengths, the strengths along the fibers vary for transverse to tensile and compression. Hence there are four uniaxial strengths; i.e., X, X', Y, and Y'. Because of shear strength is also independent, we obtain total of five strengths for structural constraints. The objective of a failure criterion is typically to choose an envelope which identifies the strength of an orthotropic ply under combined stresses. This is significant because whole layers in a laminate are under combined stresses[73]. The allowables herein are strength parameters for the homogenized laminate rather than ply data. This is addressed in the next section. MSC Nastran can incorporate several traditional failure criteria. Tsai–Wu failure criteria was selected here, which is explained in Appendix C in detail, thus needs these design constraints to define an envelope.

4.2.2 Homogenized Laminate Approach

While working with composite laminates, stacking sequence, by repeated set of layers, often called sub-laminates were treated as a homogenous single layer material. Equivalent properties were obtained by Composite Lamination Theory (CLT) given in Appendix D.

Homogenized laminate approach provides ease of designing laminated composite parts as if there are homogenous materials. Equivalent properties, namely stiffness and strength parameters are determined. It defines a building-block material that can be associated with a single thickness variable as opposed to ply-by-ply description. This means a continuous thickness variable can replace the discrete variables (number of layers) provided that the laminate can be set as a repeat of building block or sub-laminate of distinct fiber orientations. This schematically described in Figure 19. Non crimp fabric (NCF) composites deliver the homogenized laminates as they are indeed a pack of layers with distinct fiber orientations. That is NCF itself is a sublaminate or building block for composite design. In other words, homogenized laminate approach is also practical while working with NCF easily. In this study T700 NCF/Epoxy material, which had fiber volume fraction of 64 percent, is used.

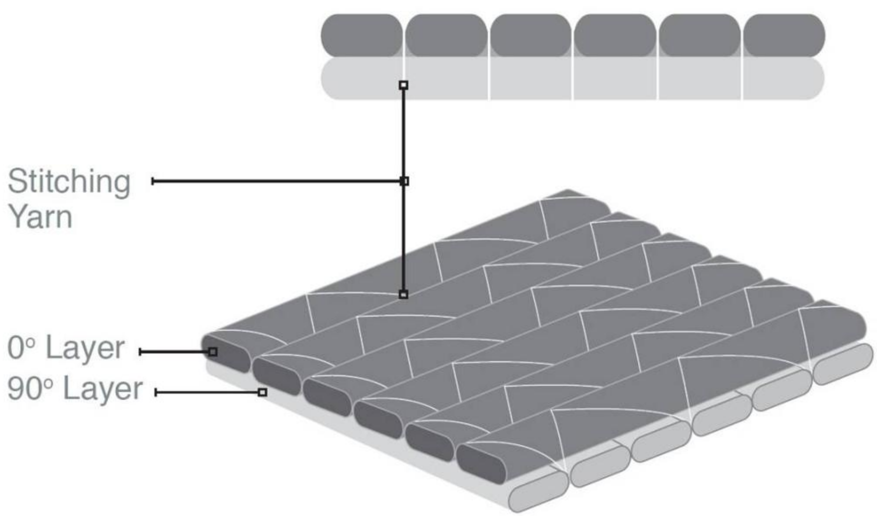


Figure 17. Typical structure of a Non Crimp Fabric (NCF)[74]

Table 5. Unidirectional single layer of T700 NCF/Epoxy material properties Vf=0.64

LongitudinalYoungModulus	E_{11} (GPa)	140.7
TransversialYoungModulus	E_{22} (Gpa)	9.318
Poisson'sratio	ν_{12}	0.3
ShearModulus	G_{12} (GPa)	5.786
Density	ρ (kg/m^3)	1600

Table 6. Orthotropic homogenous layer T700 NCF/Epoxy material properties Vf=0.64 (sublaminate of 0/45/-45/0)

LongitudinalYoungModulus	E_{11} (GPa)	81.2
TransversialYoungModulus	E_{22} (Gpa)	23.7
Poisson'sratio	ν_{12}	0.67
ShearModulus	G_{12} (GPa)	21.9
ShearModulus	G_{23}(GPa)	21.9
ShearModulus	G_{13}(GPa)	10.55
Density	ρ (kg/m^3)	1600

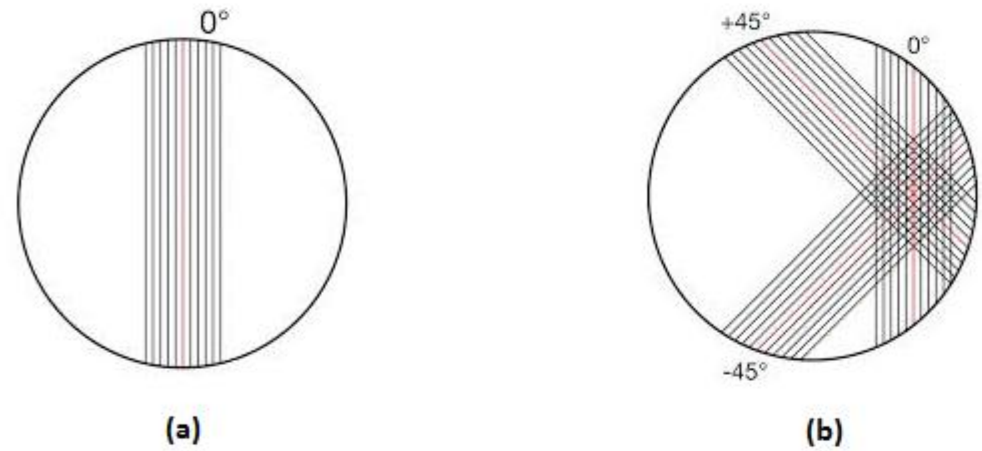
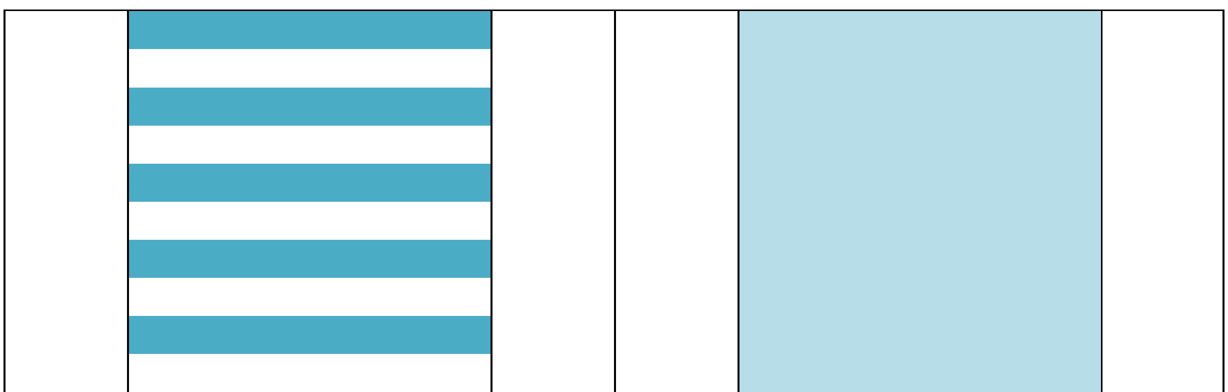


Figure 18. Schematic drawing of NCF oriented as (a) [0],(b) [0/45/-45/0]



[0/θ]	Homogenous single layer
$t_{total}=t_{ply}N$	t_{total}
Ply stiffness	Equivalent sublaminate stiffness
Ply strength	Equivalent sublaminate strength

Figure 19. Design by sublaminates and equivalent properties in design

4.3 Performance Module

Range of an aircraft, maximum flight distance without refueling relies on the rate of fuel consumption of the engine.

The thrust specific fuel consumption can be described in SI units as:

$$\frac{N\ fuel\ /sec}{N\ thrust} \qquad or \qquad \frac{1}{sec}$$

Such assumptions for the mathematical model allow approaching into the real world problem without drastically violating the problem. So, utility of calculus in the simplified form motivates to design and engineering applications.

After the simplifying assumptions have been determined, designer's concern is the change of the aircraft weight over the change in time. The weight of the aircraft decreases by the weight of the burned fuel.
Using distance and then time as the independent variable, these mathematical expressions are obtained:

$$\frac{dx}{dW} = -\frac{V}{c}\frac{C_L}{C_D}\frac{1}{W} \tag{6}$$

$$\frac{dt}{dW} = -\frac{1}{c}\frac{C_L}{C_D}\frac{1}{W} \tag{7}$$

A well known Breguet equation for range is acquired by integrating these physical statements with other established relations. From these differential equations, it is observed that the range of an aircraft depends on the sum of these small differential weight changes. Consequently, combining these equations, the total range can be stated as:

$$X_{destination} - X_{takeoff} = Range = \int_{W_{final}}^{W_{initial}} \frac{V}{tsfc} \frac{C_L}{C_D} \frac{1}{W}\, dW \tag{8}$$

As seemed in the range formula, there are several variables inside of the integral sign and the relationship of each of these variables to weight needs to be defined in advance. The most suitable assumption is taking two of the three critical parameters (ρ, V, C_L/C_D) constant over the range of the integration. [64]

Three different cruise programs are most common for range calculations.
These are;

- constant altitude-constant lift coefficient flight
- constant airspeed – constant lift coefficient flight
- constant altitude – constant airspeed flight

General form is also called as Breguet Range Equation is most common of the cruise-climb program (which V and C_L is assumed as constant) because the mathematics is simpler and the errors could be ignored.[75]

$$\begin{aligned} Max \quad & f(x) \\ subject\ to \quad & 4 < v_1 < 6 \\ & 15 < v_2 < 17 \\ & 38 < v_3 < 42 \end{aligned}$$

v_1: tip chord (tc)

v_2: root chord (rc)

v_3: half span length (b)

Reference wing geometry parameters and bounds of design variables are given in Table 7.

Objective is maximization of range, so Breguet range equation is used.

For constant velocity (V) and lift coefficient(C_L):

$$Range = \frac{V}{tsfc}\frac{C_L}{C_D}\ ln\frac{W_i}{W_f} \tag{9}$$

where V stands for the design cruise speed and tsfc is the thrust specific fuel consumption (units: $Kg.s^{-1}.N^{-1}$) which depends on the driving force system. W_i and W_f are the initial and final weights for the cruise flight stage, respectively.

For this design, optimization parameters were defined according to the cruise condition: 13000 m altitude conditions with a free-stream velocity V = 150 m/s and Mach number M = 0.85. The weight W_{rest} except fuel and wings was taken constant initially consistent with the reference.

$$W_i = W_w + W_{fuel} + W_{rest} \tag{10}$$

$$W_f = W_w + W_{rest} \tag{11}$$

$W_{w:}$ wing weight

W_{fuel}= fuel weight

W_{rest} = structural, body and payload

$W_{rest}$is initialized based on the reference aircraft A 380.

Table 7. List of design variables for range optimization

Design variables	Reference values	Lower bound	Upper bound
v_1(tc) [m]	4.9	4	6
v_2(rc) [m]	16.3	15	17
v_3 (b) [m]	39.9	38	42

4.4 Integration between Structural and Performance Modules

The most critical issue in multidisciplinary optimization is to establish communication, data flow and integration among the different disciplines. Response surface methodology which involves design of experiments and regression techniques is implemented for this purpose.

4.4.1 Design of Experiments

Design variables are named as factors, values of design variables are named as levels, and objective functions are called as observations within the context of Design of Experiments (DOE).Three level full factorial design is used in this study (see Figure 20).

Factors of interest for the experimentation (here the wing geometry parameters) were selected firstly. The selection corresponds to the design space with the intention that the experimentation produces feasible and robust response of interest. Limits of design variables and corresponding middle point were chosen for three-levels of each factor. The following adaptation function is used for better fit and it is mapped the design domain and coded domain.

$$x_i = \frac{v_i - [\max(v_i) + \min(v_i)]/2}{[\max(\mathrm{v_i}) - \min(\mathrm{v_i})]/2} \qquad (12)$$

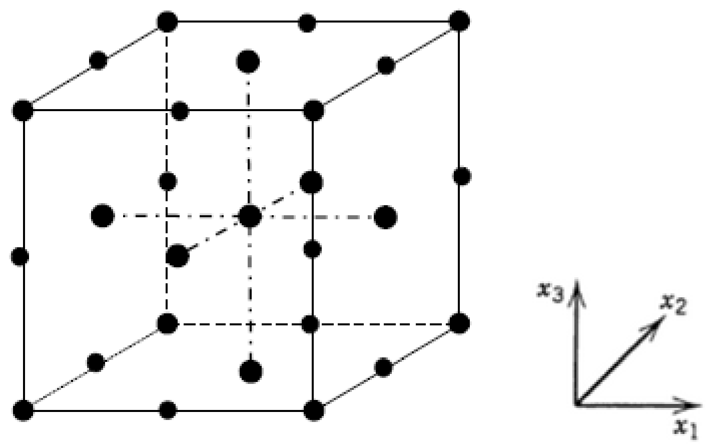


Figure 20. Three-level full factorial experimental design for three coded-configuration variables

4.4.2 Multivariable Regression

Due the fact that the formula of the relationship between the response and the independent variables is unspecified in most RS problems, an approximation to a response function y in terms of predictor variables x_i's is estimated. The response model is generally written as

$$y = F(x_1, x_2, \dots, x_n) + \varepsilon \tag{13}$$

where ε is an error term.

Clarifying the optimum operating conditions for the system or establishing a region of the factor space in which operating requirements are fulfilled are the ultimate goal of Response Surface Methodology. For more comprehensive learning, books of Khuri and Cornell Myers [76], Montgomery and Anderson-Cook [77], and Box and Draper [78] can be suggested.

Typically, a low-order polynomial in particular zone of the independent variables is tried. In case of the response is fitted pleasingly by a linear function of the independent variables, then the approximating function is the first-order model

$$y = \beta_0 + \beta_1 x_1 + \beta_2 x_2 + \cdots + \beta_k x_k + \epsilon \tag{14}$$

When curvature in the system comes into question, then it is utilized a polynomial of higher degree, for instance the quadratic mode

$$y = \beta_0 + \sum_{i=1}^{k} \beta_i x_i + \sum_{i}^{k} \beta_{ii} x_i^2 + \sum_{i<j} \sum \beta_{ij} x_i x_j + \epsilon \tag{15}$$

The linear multiple regression model is rewritten in matrix form as

$$Y = X\beta + \varepsilon \tag{16}$$

$$Y = \begin{Bmatrix} y_1 \\ y_2 \\ \vdots \\ y_n \end{Bmatrix}, \quad X = \begin{bmatrix} 1 & x_{11} & x_{12} & \cdots & x_{1k} \\ 1 & x_{21} & x_{22} & & x_{2k} \\ & \vdots & & \ddots & \vdots \\ 1 & x_{n1} & x_{n2} & \cdots & x_{nk} \end{bmatrix} \tag{17}$$

$$\beta = \begin{Bmatrix} \beta_1 \\ \beta_2 \\ \vdots \\ \beta_k \end{Bmatrix}, \ \ and \ \ \varepsilon = \begin{Bmatrix} \varepsilon_1 \\ \varepsilon_2 \\ \vdots \\ \varepsilon_n \end{Bmatrix} \tag{18}$$

and the coefficient vector b can now be expressed using the Least Square error method as

$$b = (X^T X)^{-1} X^T Y \tag{19}$$

sum of squares of the residuals SSE is following

$$SS_E = Y^T Y - b^T X^T Y \tag{20}$$

where σ is the error of Y. The estimated value of σ is

$$\sigma^2 = \frac{SS_E}{n-k-1} \tag{21}$$

The adjusted coefficient of multiple determination is used to assess performance of the approximation of the response surface

$$R_{adj}^2 = 1 - \frac{SS_E/(n-k-1)}{S_{yy}/(n-1)} \tag{22}$$

Sum of squares

$$S_{yy} = Y^T Y - \frac{(\sum_{i=1}^{n} y_i)^2}{n} \tag{23}$$

The test statistic (such as F-statistic) and its statistical table value (F-distribution) associated with the selected significance level α is used. If the statistic is larger than the table value, the test is considered as "significant" at level α. According to the table called p-value, level of significance is 0.05 by which researcher allows 5% probability of making a mistake, each factor having a p-value higher than 0.05 was eliminated from the mathematical model.

Here the response y is the optimal structural wing box weigh as a function of the planform parameters.

CHAPTER 5

5. Results and Discussion

The case study problem is a swept wing structure with carbon epoxy skins, aluminum webs and spar cabs subjected to elliptic pressure loading. The wing is illustrated in Figure 11. The wing parameters are commenced based on the reference A380-like wing and remained constant except the selected planform design variables during the optimization. The reference wing problem had been previously examined for tailoring wing structures under off-design conditions in Reference [79]. Upper and lower skins are assumed identical. Each station has uniform thickness. The skins are assumed to be made up of 0° and 45° T 700 NCF carbon epoxy laminates.

5.1 Validation of Homogenized Laminate Approach

In order to compare and validate the homogenized laminate approach two cases differentiated by the use of material definitions in the structural analyses were carried out.

In Case I, homogenized laminate properties of (0/45) sublaminate were applied. Each element was treated as if a homogeneous shell with the generated equivalent properties for stiffness matrix. PCOMP and MAT8 card, which defined in MSC Nastran were converted to equivalent PSHELL. In literature, detailed information about this conversion was presented in *Sensitivity and Optimization of Composite Structures using MSC Nastran*. [80]

```
 Material Record : equivalent 0/45 ply_t700_NCF_si_064
MAT8     1        78.5+9  13.9+9   .36     10.6+9  10.6+9   5.3+9   1.6+4


Material Record : equivalent 0/45-45/0 ply_t700_NCF_si_064
MAT8     1        81.2+9  23.7+9   .67      21.1+9  21.1+9 10.55+9   1.6+3
         28.-6    1.5-6           775.+6   750.+6  378.+6 180.+6  298.+6
```

Figure 21. Input blocks of anisotropic and orthotropic material properties

Stress limits mentioned in section 5.3 were defined within input file for MSC Nastran as the following:

```
 Material Record : equivalent 0/45 and 0/-45 (-S) ply_t700_NCF_si_064
 X, X', Y, Y', S, S'
DCONSTR DCID   RID      LALLOW  UALLOW
------12------23------34------45------56------67------78------89------9X
DCONSTR 200       101      -967.+6  645.+6
DCONSTR 200       102      -236.+6   93.+6
DCONSTR 200       103       -90.+6  205.+6
DCONSTR 200       103      -205.+6   90.+6

DCONSTR 200       201      -967.+6  645.+6
DCONSTR 200       202      -236.+6   93.+6
DCONSTR 200       203       -90.+6  205.+6
DCONSTR 200       203      -205.+6   90.+6



 Material Record : equivalent 0/45/-45/0 ply_t700_NCF_si_064
 X, X', Y, Y', S, S'
DCONSTR DCID   RID      LALLOW  UALLOW
------12------23------34------45------56------67------78------89------9X
DCONSTR 200       101      -775.+6  750.+6
DCONSTR 200       102      -378.+6  180.+6
DCONSTR 200       103      -298.+6  298.+6

DCONSTR 200       201      -775.+6  750.+6
DCONSTR 200       202      -378.+6  180.+6
DCONSTR 200       203      -298.+6  298.+6
```

Figure 22. Input blocks of failure criteria of T700 NCF material

The design constraint was the maximum deflection at the tip of wing equal to 2 m. The outcomes were given according to the objection function and the tip deflection for the number of iterations.

In Case II, outputs of optimization in Case I is verified, as special layer-by-layer output is regenerated by the PCOMP option along with MAT8 card reading the individual ply properties and detailed stacking sequence for ply-by-ply analysis .[81] That is homogenized laminate optimal thicknesses found in the Case I, were converted into the ply-by-ply equivalent of the laminates. The orientation angles were specified with respect to the x reference co-ordinate. So, material oriented at 0° had fibers lying through spanwise. The skins are denoted by QUAD4 membrane elements and the webs are denoted by SHEAR panel elements.

Homogenized laminate approach and ply-by-ply analysis were compared and MSC Patran visual deformation results regarding MSC Nastran solution were given as in Figure 23 and Figure 24.

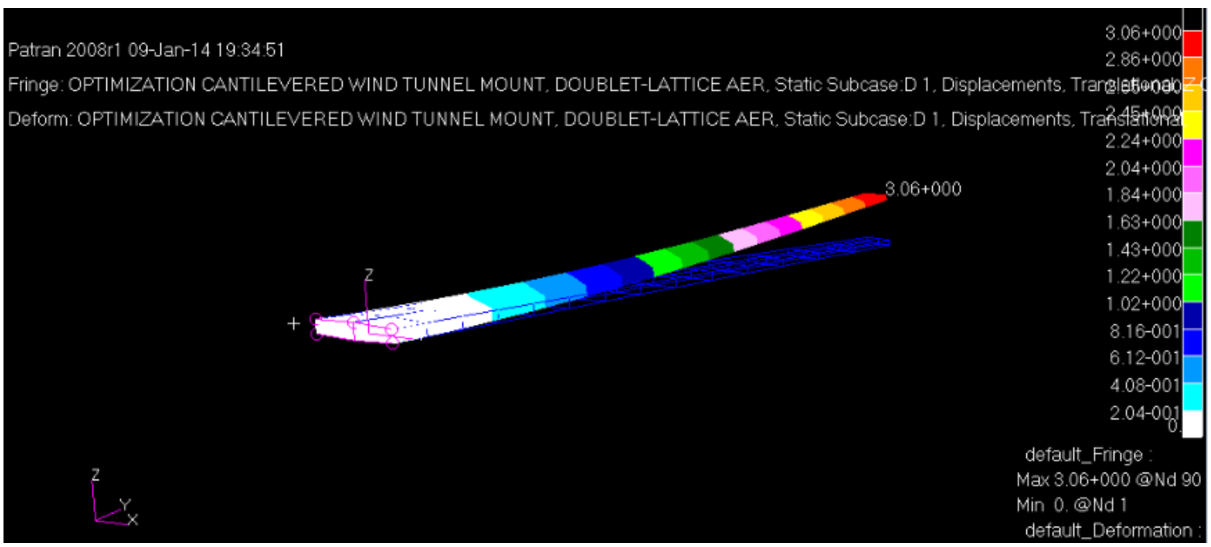


Figure 23. Case I (optimization with equivalent properties using homogenized laminate approach) - Deformation scale in z direction

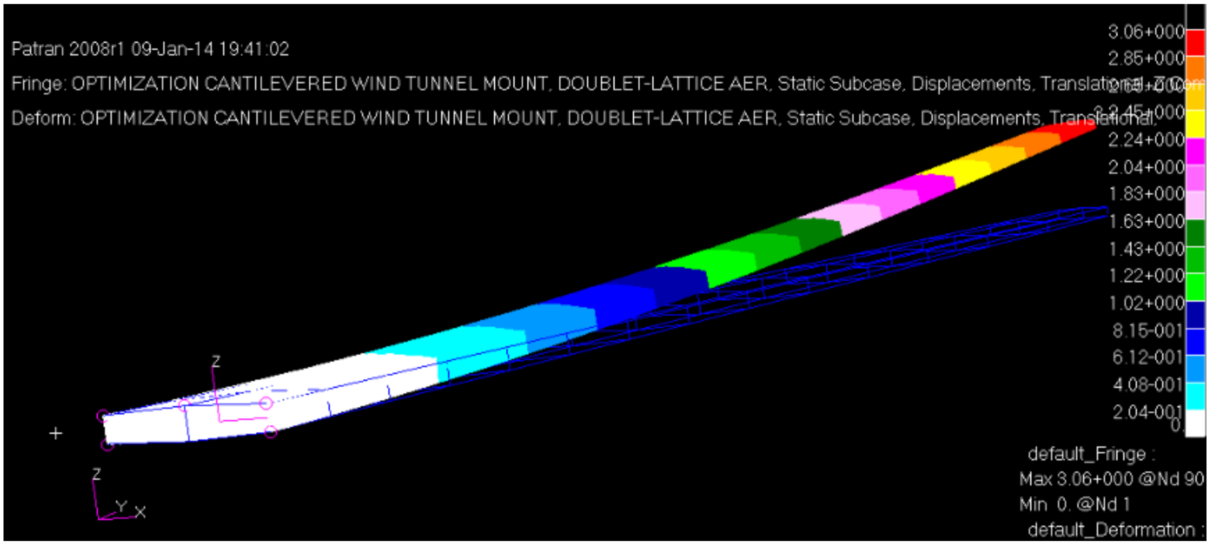


Figure 24.Case II (ply-by-ply structural analysis for verification)- Deformation in z direction

In addition to the results of maximum deformation points at node 90, all 90 points for displacement vectors are compared and high degree of similarity of two cases with regards to displacement data were obtained.

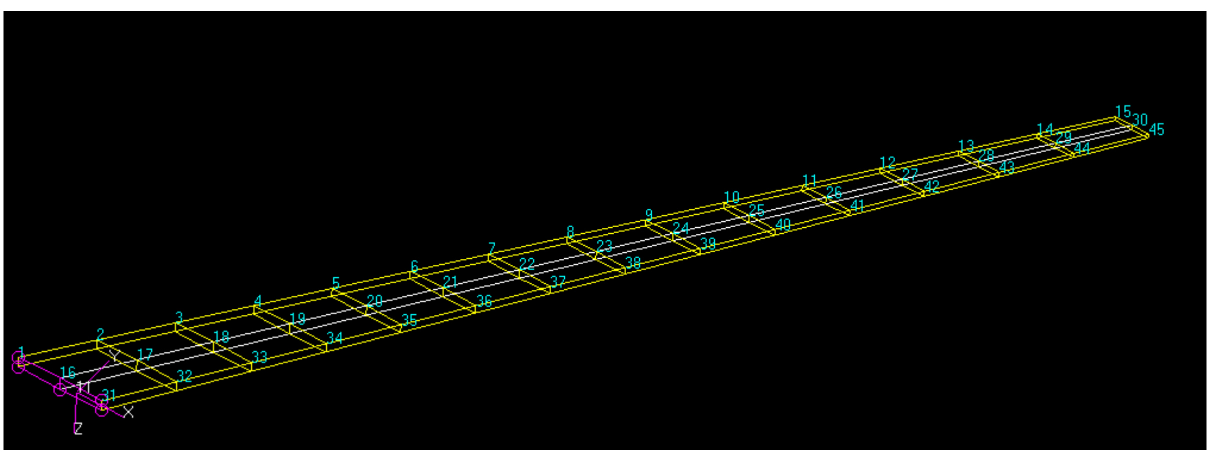


Figure 25.Node numbers from 1 to 45 for upper skin in structural model

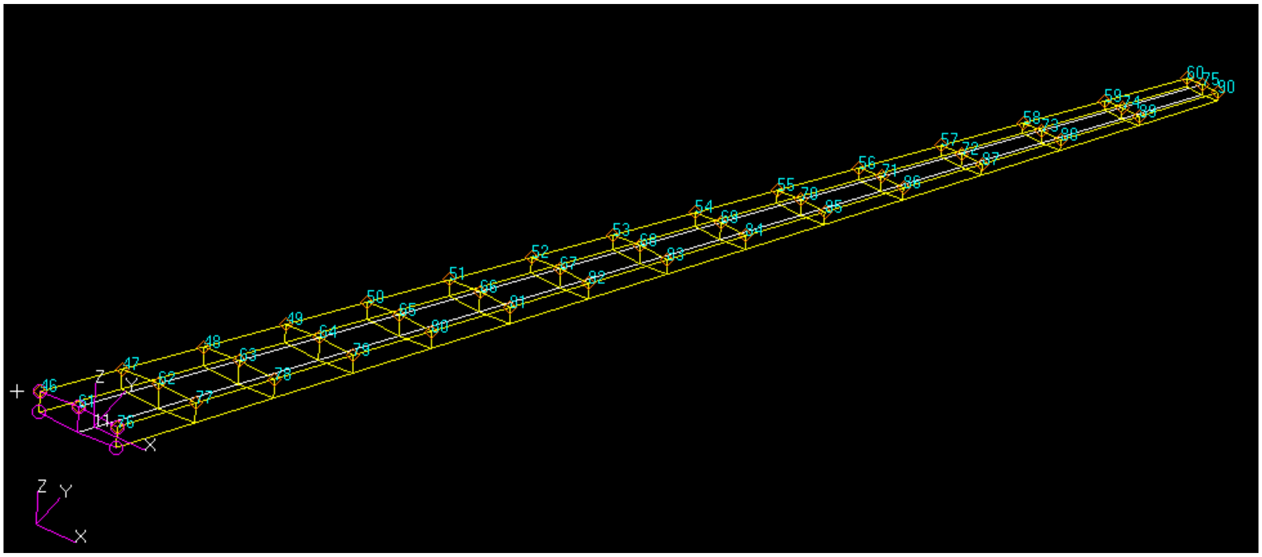


Figure 26. Node numbers from 45 to 90 for lower skin in structural model

There were 90 points for displacement vector. According to the data taken from 30. point, optimization and ply by ply results are given below:

Optimization for 0/-45

T1	T2	T3	R1	R2	R3
1.031756E-02	2.361159E-02	3.008326E+00	9.794798E-02	-4.195225E-02	0.0

Ply by ply for 0/-45

T1	T2	T3	R1	R2	R3
1.032095E-02	2.360512E-02	3.005776E+00	9.703874E-02	-3.914716E-02	0.0

Figure 27. Comparison translation and rotation displacement vector in 3 dimension for node 30 (middle of the tip chord)

5.2 Results within the MDO Framework

Response surface methodology provided a beneficial approach to integrate the design of structural weight optimization within the performance optimization. Data coming from structural optimization runs based on the planforms from the design of experiments was used for fitting a mathematical model. (For 27 experiments, homogenized laminate properties of [0/45/-45/0] sublaminate were applied. - see Figure 22 and Figure 23) Least Square Method is utilized for Response Surface model by using JMP 7 software package. The data from the structural optimization results are given as in following Table 8:

Table 8. Design domain factors and observations

Run Order	Tip chord v_1, (m)	Root chord v_2, (m)	Half span length (v_3), m	Weight Response, y (kg)
1	4	15	38	7,25E+03
2	4	15	40	8,56E+03
3	4	15	42	1,01E+04
4	4	16	38	7,04E+03
5	4	16	40	8,27E+03
6	4	16	42	9,70E+03
7	4	17	38	6,89E+03
8	4	17	40	8,04E+03
9	4	17	42	9,41E+03
10	5	15	38	7,33E+03
11	5	15	40	8,63E+03
12	5	15	42	1,01E+04
13	5	16	38	7,13E+03
14	5	16	40	8,35E+03
15	5	16	42	9,76E+03
16	5	17	38	6,98E+03
17	5	17	40	8,13E+03
18	5	17	42	9,46E+03
19	6	15	38	7,44E+03
20	6	15	40	8,65E+03
21	6	15	42	1,02E+04
22	6	16	38	7,25E+03
23	6	16	40	8,46E+03
24	6	16	42	9,97E+03
25	6	17	38	7,10E+03
26	6	17	40	8,25E+03
27	6	17	42	9,56E+03

Table 9. Coded domain factors, responses and predicted response

Exp.No.	**x_1**	**x_2**	**x_3**	**Weight Response**	**Predicted Response**
1	-1	-1	-1	7251,2	7259,907
2	-1	-1	0	8556,6	8551,019
3	-1	-1	1	10076	10063,24
4	-1	0	-1	7043	7074,074
5	-1	0	0	8269,2	8288,519
6	-1	0	1	9696	9724,074
7	-1	1	-1	6888,7	6888,241
8	-1	1	0	8044,4	8026,018
9	-1	1	1	9407,5	9384,907
10	0	-1	-1	7332,3	7311,852
11	0	-1	0	8630,8	8602,963
12	0	-1	1	10142	10115,19
13	0	0	-1	7134,8	7141,852
14	0	0	0	8349,5	8356,296
15	0	0	1	9763,4	9791,852
16	0	1	-1	6983,1	6971,852
17	0	1	0	8127,4	8109,63
18	0	1	1	9458,1	9468,518
19	1	-1	-1	7443	7408,241
20	1	-1	0	8654,1	8699,352
21	1	-1	1	10233	10211,57
22	1	0	-1	7248,2	7254,074
23	1	0	0	8457,1	8468,519
24	1	0	1	9971,6	9904,074
25	1	1	-1	7103	7099,907
26	1	1	0	8245,7	8237,685
27	1	1	1	9560,5	9596,574

Approximated β parameters were obtained at the end of analysis:

$$y = 8356{,}3 + 90x_1 - 246.67x_2 + 13125\mathrm{x}_3 + 22{,}22\mathrm{x}_1^2 + 15.83\mathrm{x}_1\mathrm{x}_2 - 76{,}67\mathrm{x}_2\mathrm{x}_3 + 110{,}56\mathrm{x}_3^2 \tag{24}$$

This equation was substituted in W_w and the planform optimization was performed by MATLAB for the problem formulation in Eq. 10 and 11. (see Appendix B)

Table 10. Summary of Fit

RSquare	0.999
RSquareAdj	0.999
Root Mean Square Error	29.898
Mean of Response	8444.815
Observations (or Sum Wgts)	27

Table 11. Analysis of Variance

Source	DF	Sum of Squares	Mean Square	F Ratio
Model	7	32992090	4713156	5272.527
Error	19	16984	894	Prob> F
C. Total	26	33009074		<.0001*

Table 12. Parameter Estimates

Term	Estimate	Std Error	t Ratio	Prob>|t|
Intercept	8356.2963	12.86618	649.48	<.0001*
x1	90	7.047097	12.77	<.0001*
x2	-246.6667	7.047097	-35.00	<.0001*
x3	1325	7.047097	188.02	<.0001*
x1*x1	22.222222	12.20593	1.82	0.0845
x1*x2	15.833333	8.630896	1.83	0.0823
x2*x3	-76.66667	8.630896	-8.88	<.0001*
x3*x3	110.55556	12.20593	9.06	<.0001*

Table 13. Sorted Parameter Estimates

Term	Estimate	Std Error	t Ratio	t Ratio	Prob>|t|
x3	1325	7.047097	188.02		<.0001*
x2	-246.6667	7.047097	-35.00		<.0001*
x1	90	7.047097	12.77		<.0001*
x3*x3	110.55556	12.20593	9.06		<.0001*
x2*x3	-76.66667	8.630896	-8.88		<.0001*
x1*x2	15.833333	8.630896	1.83		0.0823
x1*x1	22.222222	12.20593	1.82		0.0845

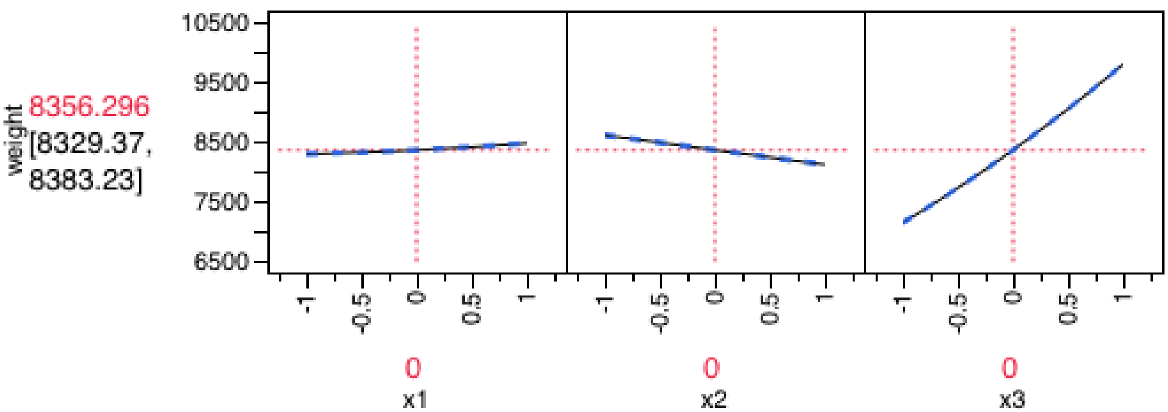


Figure 28. Prediction Profiler

Full Factorial DOE data, optimal and reference geometry for range is given in Table 14.

According to the reference wing, optimal wing configuration provides about 6% better range performance.

Table 14. Calculated range values for experiments, reference wing and optimum geometry

	Tip (v_1)	Root (v_2)	Half span (v_3)	Range	Normalized Range
1	4	15	38	8099110	0,999208
2	4	15	40	8105531	1
3	4	15	42	8066572	0,995194
4	4	16	38	7904971	0,975256
5	4	16	40	7928010	0,978099
6	4	16	42	7905554	0,975328
7	4	17	38	7721567	0,952629
8	4	17	40	7760330	0,957412
9	4	17	42	7753577	0,956578
10	5	15	38	7865556	0,970394
11	5	15	40	7877169	0,971826
12	5	15	42	7844454	0,96779
13	5	16	38	7680350	0,947544
14	5	16	40	7707782	0,950929
15	5	16	42	7690808	0,948835
16	5	17	38	7505087	0,925922
17	5	17	40	7547508	0,931155
18	5	17	42	7545538	0,930912
19	6	15	38	7637537	0,942262
20	6	15	40	7653964	0,944289
21	6	15	42	7627145	0,94098
22	6	16	38	7460614	0,920435
23	6	16	40	7492106	0,92432
24	6	16	42	7480310	0,922865
25	6	17	38	7292920	0,899746
26	6	17	40	7338706	0,905395
27	6	17	42	7341247	0,905708
reference	4,9	16,3	39,9	7679878	0,947486
optimal	4	15	39,2622	8108704	1,000391

5.3 Discussion and Concluding Remarks

Multidisciplinary design optimization entails high computational cost and difficulties of information flow among the disciplines. These issues can be addressed by the use of approximations such as response surfaces tailored within the individual disciplines. This study presents an example of RSM in an MDO framework. Response surface was to provide an optimal structural wing weight equation by which essential information for optimizing range of the airplane can be predicted.

Wing geometry parameters, root chord length, tip chord length and span, were adopted as the design variables that are also the variables for the approximation function within the entire design space. Three-level full factorial experimental design was considered. The methodology began with generating optimal structural design for each planform geometry (for 27 wing planforms by the experimental design) subjected to the stress and tip displacement constraints for given static loading, boundary conditions.

As for the structural optimization of composite wing planforms, homogenized laminate approach and associated equivalent properties were used. Homogenized laminates approach allowed using single continuous thickness variables for each assigned laminate domain replacing the ply-by-ply description of the laminated structure within the structural analyses. Comparison of the homogenized laminate approach and ply-by-ply analyses for a reference wing design showed the results are very close.

Next, the generated optimal structural weight data was utilized to construct the RS weight equation as a function of the tip chord, root chord, and half span length parameters for the optimal composite wing-box structural weight. Accuracy of the RS was very good, the maximum error was less than 1% for displacement vector in z direction.

The range optimization was then completed by genetic algorithm implemented in MATLAB, the optimal range was found at the vicinity of one of the 27 DOE configurations showing about 6% increase compared to the reference design.

In conclusion, response surface methodology provided a scan of the design space and beneficial approach to integrate different disciplines. By the RS based weight equation, design optimization of the wing structure was incorporated within the performance optimization. It is also important to note that presented MDO framework can be implemented into larger scale problems easily.

Appendix A

Input - LAMAR's LAMDES

```
wingoptimization - rafaelpereira
1.000     0.0       11.62     845.88    0.0       0.00.0
3.000     0.0       0.00.00.00.00.0
0.0       0.00.0       1.0
-30.4843  -39.90    0.0       1.0
-35.3843  -39.90    0.0       1.0
-16.30    0.0       0.0       1.0
1.0   1.0 50.0 0.85 0.42 25.0 0.0005
0.8       0.0       1.0       0.0       1.0       0.0       0.0
0.0       0.00.0
```

Appendix B

MATLAB Script for Optimization

```
x0=[4,15,38];
lb=[4;15;38];
ub=[6;17;42];

options = optimset('Algorithm','active-set','DiffMaxChange',1,'Tolfun',1e-
10,'Display','iter-detailed');
[x,fval]=fmincon(@range_A380_jmp,x0,[],[],[],[],lb,ub,[],options)
```

```
lb=[4;15;38];
ub=[6;17;42];


options = gaoptimset('PopulationSize',1000,'Tolfun',1e-16,'Tolcon',1e-
16,'PlotFcns',{@gaplotbestf,@gaplotstopping})
[x,fval]=ga(@range_A380_jmp,3,[],[],[],[],lb,ub,[],options)
```

```
function R=range_A380_jmp(x)

tc=x(1); % tip chord
rc=x(2); % root chord
b=x(3); % half span length


S=(rc+tc)*b %m^2
M=0.85; dimensionless
AR=(b*2)^2/S
e=0.74; % from virginia tech range A380 matlab files ranges data
g=9.81; %m/s^2
tsfc=1.6e-4; %N/N/s
V=150; %m/s
Cl=0.6; % expected CL which is defined in LAMDES
Cd0=0.0163; % for M=0.85 from virginia Tech slides

Cd=Cd0+(Cl^2)/(pi*AR*e)
LD=Cl/Cd
Vtsfc=V/tsfc


W=8356.2963+90*(x(1)-10/2)-246.6667*(x(2)-32/2)+1325*((x(3)-
80/2)/2)+22.222222*(x(1)-10/2)*(x(1)-10/2)+15.833333*(x(1)-10/2)*(x(2)-32/2)-
76.6667*(x(2)-32/2)*((x(3)-80/2)/2)+110.55556*((x(3)-80/2)/2)*((x(3)-80/2)/2)
Wi=220000+4*W
Wf=120000+4*W
lnWiWf=log(Wi/Wf)
R=-((V/tsfc)*(Cl/Cd)*log(Wi/Wf));
```

Appendix C

Tsai-Wu Failure criteria

For utilizing materials effectively and design to fulfill the mission properly, failure criteria are needed. One of the well known failure criteria the von Mises criterion which is used for isotropic materials and also other most commonly used criteria are also empirical like von Mises. They are derived from maximum stress, maximum strain, or a stress or strain quadratic invariant. Criteria for isotropic materials like von Mises and Tresca are normally valid to yielding, on the other hand criteria for used composite material such as Tsai-Wu are used to the ultimate because of yield criteria is not valid for composites. For the reason that composite materials are clearly anisotropic and can fail in several different modes subjected to their loading condition and the mechanical properties of the material, Failure criteria are considerably further sophisticated for composite materials than for metals they are resilient and strong. For detailed information about failure criteria, books of Tsai, S. W.[73], Dowling, N. E.[82], and Kaw, A. K. [83]may be useful.

In this study Tsai-Wu[84] failure criterion is applied in within the context of MSC Nastran packages by the help of MSC Laminate Modeler for analysis of a composite wing box. MSC Laminate Modeler makes available rapid calculation of the strength of a structure in comparison with the present industry norms. PCL functions[85] can be also utilized to described custom criteria for complex applications.

This failure theory is based on the total strain energy failure theory and closely correlates with experimental data. Theory says that a lamina is failed in condition:

$$H_1\sigma_1 + H_2\sigma_2 + H_6\tau_{12} + H_{11}\sigma_1^2 + H_{22}\sigma_2^2 + H_{66}\tau_{12}^2 + 2H_{12}\sigma_1\sigma_2 < 1 \qquad (25)$$

The components H_1, H_2, H_6, H_{11}, H_{22}, and H_{66} of the failure theory are obtained by the strength parameters of a unidirectional lamina. They are given below:

$$H_1 = \frac{1}{(\sigma_1^T)_{ult}} - \frac{1}{(\sigma_1^C)_{ult}}; H_2 = \frac{1}{(\sigma_2^T)_{ult}} - \frac{1}{(\sigma_2^C)_{ult}}; H_6 = 0 \tag{26}$$

$$H_{11} = \frac{1}{(\sigma_1^T)_{ult}\,(\sigma_1^C)_{ult}};\ H_2 = \frac{1}{(\sigma_2^T)_{ult}\,(\sigma_2^C)_{ult}};\ H_{66} = \frac{1}{(\tau_{12})_{ult}^2} \tag{27}$$

$$H_{12} = -\frac{1}{2}\sqrt{\frac{1}{(\sigma_1^T)_{ult}\,(\sigma_1^C)_{ult}\,(\sigma_2^T)_{ult}\,(\sigma_2^C)_{ult}}} y = F(x_1, x_2, \dots, x_n) + \varepsilon \tag{28}$$

As σ refers with subscript of *ult* to the ultimate stress, *T* and *C* superscripts corresponds compressive and tensile. While τ_{12} means shear stress in the 1-2 plane, subscripts of 1 and 2 are directions of stresses for tensile and compression stresses.

As a result, for this quadratic approximation, strength values of the lamina, elasticity modulus and Poisson's ratio of the fibers and the lamina, fiber fraction are needed.

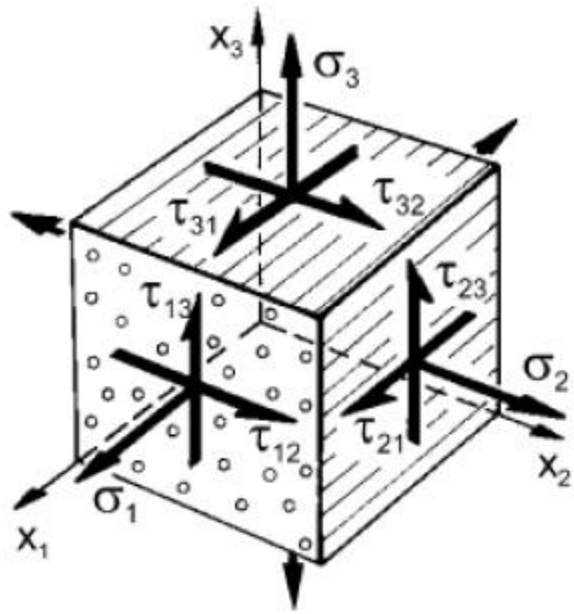


Figure 29. Stresses acting on a UD lamina [86]

Appendix D

Classical Lamination Theory

A structural laminate can consistently handled from the basic building block by the classical lamination theory (CLT). It makes practical precise simplifying assumptions, so makes possible three dimensional elasticity problem reduce to a solvable two dimensional mechanics of deformable body problem.

A unidirectional lamina is inclusive of the orthotropic material category. If the lamina is thin and any out-of-plane loads are not performed on it, plane stress conditions for the lamina may be assumed.

Therefore, Hooke's Law in three dimensions is reduced to two dimensions, and used for unidirectional lamina. We can assume,

$$\sigma_3 = 0, \tag{29}$$

$$\tau_{31} = 0, \tag{30}$$

$$\tau_{23} = 0, \tag{31}$$

$$\varepsilon_3 = S_{13}\sigma_1 + S_{23}\sigma_2, \tag{32}$$

$$\gamma_{23} = \gamma_{31} = 0. \tag{33}$$

For an orthotropic plane strains can then be written as

$$\begin{bmatrix} \varepsilon_1 \\ \varepsilon_2 \\ \gamma_{12} \end{bmatrix} = \begin{bmatrix} s_{11} & s_{12} & 0 \\ s_{12} & s_{22} & 0 \\ 0 & 0 & s_{66} \end{bmatrix} \begin{bmatrix} \sigma_1 \\ \sigma_2 \\ \tau_{12} \end{bmatrix} \tag{34}$$

Where Qij are stiffness reduced matrix coefficients, the stress–strain relationship can be written in terms of reduced stiffness matrix as

$$\begin{bmatrix} \sigma_1 \\ \sigma_2 \\ \tau_{12} \end{bmatrix} = \begin{bmatrix} Q_{11} & Q_{12} & 0 \\ Q_{12} & Q_{22} & 0 \\ 0 & 0 & Q_{66} \end{bmatrix} \quad (35)$$

$$Q_{11} = \frac{E_1}{1 - v_{12}v_{21}}, Q_{12} = \frac{v_{12}E_2}{1 - v_{12}v_{21}}, Q_{22} = \frac{E_2}{1 - v_{12}v_{21}}, Q_{66} = G_{12} \quad (36)$$

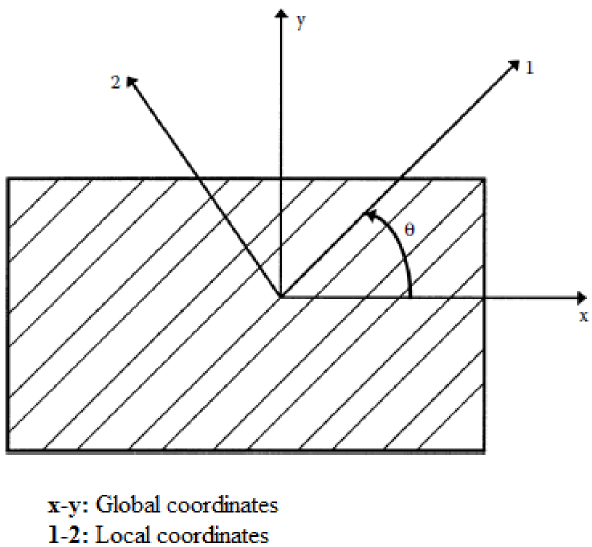


Figure 30. Global and local coordinates for an angled lamina

Composite lamination applications commonly does not contain only unidirectional laminae as a consequence of their low stiffness and strength properties in the transverse direction thus, laminates generally are placed according to an orientation. Composite lamination theory deals with the stress-strain relationship for an angle lamina. Hooke's law can be simplified by the help of material symmetry and the selected orientation of the reference coordinates.

$$\begin{bmatrix} \sigma_x \\ \sigma_y \\ \tau_{xy} \end{bmatrix} = \begin{bmatrix} \bar{Q}_{11} & \bar{Q}_{12} & \bar{Q}_{16} \\ \bar{Q}_{12} & \bar{Q}_{22} & \bar{Q}_{26} \\ \bar{Q}_{16} & \bar{Q}_{26} & \bar{Q}_{66} \end{bmatrix} \begin{bmatrix} \varepsilon_x \\ \varepsilon_y \\ \gamma_{xy} \end{bmatrix} \quad (37)$$

Accordingly, computation of transformations is critical in analyses of stress and strain, and the tensorial nature of stress and strain is obviously seen in their transformation properties. Other physical statements such as moment of inertia and curvature, also transform to stress and strain by a similar approach.

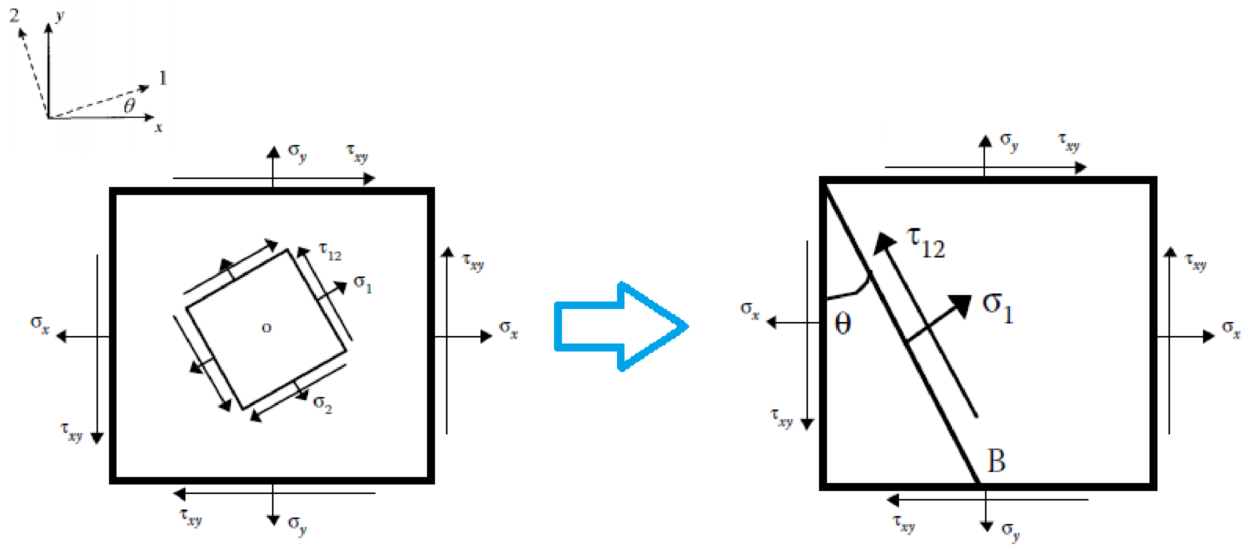


Figure 31. Transformation of stresses between local and global axes

$$\sigma_1 = \sigma_X Cos^2\theta + \sigma_y Sin^2\theta + 2\tau_{xy} Sin\theta Cos\theta \tag{38}$$

$$\sigma_2 = \sigma_X Sin^2\theta + \sigma_y Cos^2\theta + 2\tau_{xy} Sin\theta Cos\theta \tag{39}$$

$$\tau_{12} = -\sigma_x Sin\theta Cos\theta + \sigma_y Sin\theta Cos\theta + \tau_{xy} (Cos^2\theta - Sin^2\theta) \tag{40}$$

If above relations are written in pseudovector-matrix form as

$$\begin{bmatrix} \sigma_1 \\ \sigma_2 \\ \tau_{12} \end{bmatrix} = \begin{bmatrix} c^2 & s^2 & 2sc \\ s^2 & c^2 & -2sc \\ -sc & sc & c^2 - s^2 \end{bmatrix} \begin{bmatrix} \sigma_x \\ \sigma_y \\ \tau_{xy} \end{bmatrix} \tag{41}$$

$$[\bar{Q}] = [T]^{-1}[Q][R][T][R]^{-1} \tag{42}$$

T is transformation matrix:

$$T = \begin{bmatrix} c^2 & s^2 & 2sc \\ s^2 & c^2 & -2sc \\ -sc & sc & c^2 - s^2 \end{bmatrix} \tag{43}$$

where c = Cos θ and s = Sin θ

When transformation matrix is inverted:

$$[T]^{-1} = \begin{bmatrix} c^2 & s^2 & -2sc \\ s^2 & c^2 & 2sc \\ sc & -sc & c^2 - s^2 \end{bmatrix} \tag{44}$$

by defining the Reuter's matrix [R] :

$$[R] = \begin{bmatrix} 1 & 0 & 0 \\ 0 & 1 & 0 \\ 0 & 0 & 2 \end{bmatrix} \tag{45}$$

So, the relation between local stresses and global stresses is

$$\begin{bmatrix} \sigma_x \\ \sigma_y \\ \tau_{xy} \end{bmatrix} = [T]^{-1} \begin{bmatrix} \sigma_1 \\ \sigma_2 \\ \tau_{12} \end{bmatrix} \tag{46}$$

and the relation between local strains and global strains is

$$\begin{bmatrix} \varepsilon_1 \\ \varepsilon_2 \\ \gamma_{12} \end{bmatrix} = [R][T][R]^{-1} \begin{bmatrix} \varepsilon_x \\ \varepsilon_y \\ \gamma_{xy} \end{bmatrix} \tag{47}$$

$$\begin{bmatrix} \varepsilon_x \\ \varepsilon_y \\ \gamma_{cy} \end{bmatrix} = \begin{bmatrix} \varepsilon_x^0 \\ \varepsilon_y^0 \\ \gamma_{xy}^0 \end{bmatrix} + z \begin{bmatrix} \kappa_x \\ \kappa_y \\ \kappa_{xy} \end{bmatrix} \tag{48}$$

where ε_0 is the vector of the mid-plane strains; κ is the vector of mid-plane curvatures, z is an arbitrary distance from the mid-plane.

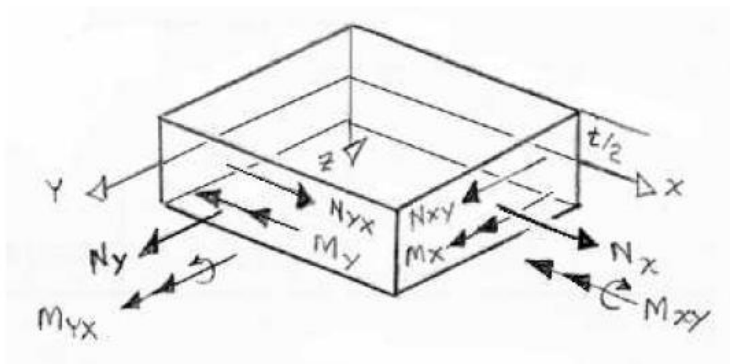


Figure 32. Forces and moments acting on a laminate [87]

N (the vector of resultant forces) and M (the vector of resultant moments) per unit width of the laminate are stated as

$$\{N_x, \quad N_y, \quad N_{xy}\} = \int_{-H/2}^{H/2} \{\sigma_x, \quad \sigma_y, \quad \tau_{xy}\} dz, \tag{49}$$

$$\{M_x, \quad M_y, \quad M_{xy}\} = \int_{-H/2}^{H/2} \{\sigma_x, \quad \sigma_y, \quad \tau_{xy}\} z dz \tag{50}$$

In matrix form, that relates the force and moment resultants to the midsurface strains and curvatures are given as:

$$\begin{bmatrix} N_x \\ N_y \\ N_{xy} \\ M_x \\ M_y \\ M_{xy} \end{bmatrix} = \begin{bmatrix} A_{11} & A_{12} & A_{16} & B_{11} & B_{12} & B_{16} \\ A_{12} & A_{22} & A_{26} & B_{12} & B_{22} & B_{26} \\ A_{16} & A_{26} & A_{66} & B_{16} & B_{26} & B_{66} \\ B_{11} & B_{12} & B_{16} & D_{11} & D_{12} & D_{16} \\ B_{12} & B_{22} & B_{26} & D_{12} & D_{22} & D_{26} \\ B_{16} & B_{26} & B_{66} & D_{16} & D_{26} & D_{66} \end{bmatrix} \begin{bmatrix} \varepsilon_x^0 \\ \varepsilon_y^0 \\ \gamma_{xy}^0 \\ \kappa_x \\ \kappa_y \\ \kappa_{xy} \end{bmatrix} \tag{51}$$

where A_{ij} are the extensional stiffnesses, D_{ij} the bending stiffness and B_{ij} the coupling stiffnesses. The elements of these [A], [B], and [D] matrices can be determined as

$$A_{ij} = \sum_{k=1}^{n} \left[\left(\bar{Q}_{ij}\right)\right]_k (h_k - h_{k-1}), \quad i = 1,2,6; \quad j = 1,2,6 \tag{52}$$

$$B_{ij} = \sum_{k=1}^{n} \left[\left(\bar{Q}_{ij}\right)\right]_k \left({h_k}^2 - {h_{k-1}}^2\right), \quad i = 1,2,6; \quad j = 1,2,6 \tag{53}$$

$$D_{ij} = \sum_{k=1}^{n} \left[\left(\bar{Q}_{ij}\right)\right]_k \left({h_k}^3 - {h_{k-1}}^3\right), \quad i = 1,2,6; \quad j = 1,2,6 \tag{54}$$

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