Design Optimization of a Regional Transport Aircraft with Hybrid Electric Distributed Propulsion Systems

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Thesis submitted to the faculty of the Virginia Polytechnic Institute and State University in partial fulfillment of the requirements for the degree of

Master of Science
In
Aerospace Engineering

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May 24, 2018
Blacksburg, VA

Keywords: Hybrid Electric, Wing-Propeller Interaction, Design Optimization, Regional Transport, Aircraft Design

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ABSTRACT

In recent years, there has been a growing shift in the world towards sustainability. For civil aviation, this is reflected in the goals of several organizations including NASA and ACARE as significantly increased fuel efficiency along with reduced harmful emissions in the atmosphere. Achieving the goals necessitates the advent of novel and radical aircraft technologies, NASA’s X-57, is one such concept using distributed electric propulsion (DEP) technology.

Although practical implementation of DEP is achievable due to the scale invariance of highly efficient electric motors, the current battery technology restricts its adoption for commercial transport aircraft. A Hybrid Electric Distributed Propulsion (HEDiP) system offers a promising alternative to the all-electric system. It leverages the benefits of DEP when coupled with a hybrid electric system. One of the areas needing improvement in HEDiP aircraft design is the fast and accurate estimation of wing aerodynamic characteristics in the presence of multiple propellers. A VLM based estimation technique was developed to address this requirement.

This research is primarily motivated by the need to have mature conceptual design methods for HEDiP aircraft. Therefore, the overall research objective is to develop an effective conceptual design capability based on a proven multidisciplinary design optimization (MDO) framework, and to demonstrate the resulting capability by applying it to the conceptual design of a regional transport aircraft (RTA) with HEDiP systems.
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GENERAL AUDIENCE ABSTRACT

Recent years have seen a growing movement to steer the world towards sustainability. For civil aviation, this is reflected in the goals of key organizations, such as NASA and ACARE, to significantly improve fuel efficiency, reduce harmful emissions, and decrease direct heat release in the atmosphere. Achieving such goals requires novel technologies along with radical aircraft concepts driven by efficiency maximization as well as using energy sources other than fossil fuel. NASA’s all-electric X-57 is one such concept using the Distributed Electric Propulsion (DEP) technology with multiple electric motors and propellers placed on the wing. However, today’s all-electric aircraft suffer from the heavy weight penalty associated with batteries to power electric motors. In the near term, a Hybrid Electric Distributed Propulsion (HEDiP) system offers a promising alternative. HEDiP combines distributed propulsion (DiP) technology powered by a mix of two energy sources, battery and fossil fuel. The overall goal of the present study is to investigate potential benefits of HEDiP systems for the design of optimal regional transport aircraft (RTA).

To perform this study, the aerodynamics module of the Pacelab Aircraft Preliminary Design (APD) software system was modified to account for changes in wing aerodynamics due to the interaction with multiple propellers. This required the development of the Wing Aerodynamic Simulation with Propeller Effects (WASPE) code. In addition, a Wing Propeller Configuration Optimization (WIPCO) code was developed to optimize the placement of propellers based on location, number, and direction of rotation. The updated APD was applied to develop the HERMiT 2E series of RTA. The results demonstrated the anticipated benefits of HEDiP technologies over conventional aircraft, and provided a better understanding of the sensitivity of RTA designs to battery technology and level of hybridization, i.e., power split between batteries and fossil fuels. The HERMiT 6E/I was then designed to quantify the benefits of HEDiP systems over a baseline Twin Otter aircraft. The results showed that a comparable performance could be obtained with more than 50% saving in mission energy costs for a small weight penalty. The HERMiT 6E/I also requires only about 38% of the mission fuel borne by the baseline. This means a correspondingly lower direct atmospheric heat release, reduction in carbon dioxide and NOx emissions along with reduced energy consumptions.
This Thesis is dedicated to my wonderful family
ACKNOWLEDGEMENT

The conception of this report was only possible thanks to the encouragement and guidance of several people.

First, I would like to thank my family, K. Rajkumar, R. Revathi and Pruthvi S. Prakash who have been my strongest motivation. Their incessant love, cheer and always think positive attitudes have helped me immensely during stressful times. The constant support and encouragement they handed out definitely kept me going.

I would like to thank my advisory committee, Dr. Pradeep Raj and Dr. Seongim Choi from Virginia Tech and Dr. Darcy L. Allison from US AFRL, for advising me with this work, through the valuable inputs and countless suggestions they have provided.

I cannot thank enough, my Advisor Dr. Pradeep Raj, our discussions ultimately ended up defining every aspect of this Thesis. I am also very grateful for the immeasurable amount of knowledge I gleansed from Dr. Raj along with his constant support and guidance. Dr. Raj has been a source of inspiration during the years I have spent in Virginia Tech and has definitely helped me strengthen my zeal for flying machines.

I would also like to thank Alexander Schneegans and Mathias Emeneth for facilitating access to Pacelab APD. I am very much obliged to Mathias, who has been incessantly supportive to the project which could have not been completed without his abundant technical input.

I am much obliged to Dr. Rakesh Kapania and the Institute for Critical Thinking and Applied Sciences (ICTAS) for their very kind support to this Thesis.

Many thanks to Nathaniel Blaesser for his kind help with VSPAero, Rikin Gupta for his help with FLOPS and to the very helpful graduate community for creating a wonderful environment for research. I am also very grateful to my wonderful friends, Deep Doshi, Karan Kothari and Ravishwar Karthikeyan for keeping things interesting.
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CHAPTER 1

INTRODUCTION

1.1 Motivation

The recent years have seen a growing movement to steer the world towards sustainability. For civil aviation, this is reflected in the goals of several organizations such as NASA [1] and ACARE as significantly improved fuel efficiency along with reduced harmful emissions and direct heat release in the atmosphere. These goals are supported by the airline industry as well owing to the positive impact of ultra-efficient technologies on the operational cost and the environment.

![NASA Targeted Improvements in Subsonic Transport System-level Metrics](image)

Figure 1.1 NASA Subsonic Transport Technology Goals. Used under Fair Use 2018
Achieving such ambitious goals necessitates the advent of novel technologies and systems along with radical aircraft concepts driven by efficiency maximization as well as utilization of energy sources other than fuel. Now is the right time to consider an amalgamation of alternate fuel technologies with radical designs, to explore innovative propulsion and airframe design concepts. Contributions to the maturation of conceptual design tools and methods for such unconventional concepts would prove to be indispensable.

1.2 Recent Technological Advances

The conception of the X-57 Maxwell [2] by NASA, shown in Figure 1.2, is one such unconventional design that leverages alternate fuel sources by using a distributed electric propulsion (DEP) system. It effectively leverages the scale invariant performance of electrical motors. The DEP of X-57 is a modern implementation of the traditionally infeasible distributed propulsion concept that was hither before considered impractical.

The purpose of this study is to explore the merit of one such concept, heavily inspired by the Maxwell, in the light of modern environmental and operational requirements while enabling an effective multidisciplinary design optimization and conceptual design approach.

To achieve NASA goals presented in Figure 1.1, centered on the environment and backed by forums for sustainable aviation research such as those held by AIAA, many advancements in technologies, associated with Distributed Electric Propulsion, have been introduced since the 2010s. Researchers have looked at
several facets and challenges emerging from such a consideration, from feasibility and prospective conceptual design studies to advancement in aerodynamic and electric propulsion performance estimation techniques.

Borer et al. [3] and Stoll et al. [4] elucidate the benefits of utilizing distributed propulsion in what is now the X-57. The presence of multiple propellers increases the dynamic pressure over the wing and thus can act as high lift systems during takeoff. Shutting down for cruise, this allows for a lifting surface design optimally tuned for cruise performance.

Although the concept of distributed propulsion is in existence since the 1920s, application of this technology was restricted by the complexities in distributing traditional propulsors along with penalties in terms of weight, efficiency and cost. These penalties can be overcome by using electric motors with their scale invariant performance, independence of power generation to altitude and faster response time. Electric motors also are lighter and more compact than traditional powerplants for the same performance. With the offering of environmental benefits in the form of reduced operating noise, lower harmful gas emission and reduced fossil fuel consumption can be realized, electric propulsion does indeed hold a lot of potential.

This brings about the consideration of alternate energy sources, to power the motors, such as the utilization of partially electric and fully electric sources. Automotive vehicle manufacturers, such as Tesla [6], Toyota [7] and Nissan [8] among others, have already paved the way with several commercially available products utilizing alternate sources of energy. There has also been a galvanized effort in the aviation industry to
adopt battery technology and scalable motor technology. Enormous benefits associated with electric aircraft propulsion systems in terms of efficiency are highlighted in Figure 1.3 obtained from the works of Hepperle [5]. Utilization of such battery stored energy extends to the user nearly twice the energy conversion efficiency while at the same time curtailing the nefarious impact of traditional fuel powered air vehicle operations on the environment.

Moore and Fredericks [9] in their work strongly promote the consideration of feasible inclusion of electrically powered aircraft in near – imminent aviation networks. Innovations that could systematically utilize projected electrical storage technologies, which have been traditionally predicted to require vast improvements to be a direct alternative for fossil fuels, when involved with innovations in other fronts such as DEP have been discussed. They also clearly articulate the potential of dramatic reductions in energy costs of their LEAPTech concept due to its innovative DEP system in making a very convincing case for continuing to pursue electrical aircraft designs for not just general aviation, but also for commercial transport aircraft. Gohardani [10] also indicates, notwithstanding the power density limitations, an effective application of DEP could be revolutionary. Combined investigations encompassing this and different levels of aircraft electrification could indeed prove to be worthwhile.

1.3 Challenges for All-Electric Aircraft

The advent of fully electric powered aircraft, while a potential boon to the aviation industry, is restricted by the current battery technology. While opportunities are being widely explored for complete electrification, full-fledged adoption of such technologies in the commercial transport business is constrained.

A substantial limitation of electrical energy is the heavy weight penalty that is associated with energy storage for all electric propulsion adoption in aerospace application.

The limitations and prospects of electrification are currently limited to ultralights and maybe general aviation aircraft. Nevertheless, several technology demonstrators such as the Pipistrel Alpha Electro and JAXA Feather institute that exclusive electrical energy utilization cannot be precluded from contemporary aviation.
The Jet – A fuels have an energy density of about 11.5 kWh/kg whereas the best commercially available battery technology of today, Lithium-Ion batteries are capable of achieving an energy density of 0.265 kWh/kg. Although several advancements in battery technologies have shown promising results in laboratory settings, widespread commercial application of such breakthroughs have yet to materialize.
A recent study by Patterson [9] shows that an all-electric general aviation aircraft using 2050 battery technology would result in half the range of the Cirrus SR22, a conventional hydrocarbon based aircraft, for comparable gross weight of approx. 3,600 lbs.

At least twice the contemporary battery specific energy, i.e., about 0.500 kWh/kg is required for practical usage. Clearly, further radical improvements in battery technology are required for an all-electric variant of an existing design to be competitive in the marketplace.

An order of magnitude increase in specific energy to about 1.2 kWh/kg is needed for a commercially viable aircraft. This would push feasible application of electric propulsion to well beyond 2050 timeframes. A more near-term solution might result from a combined implementation of DEP and hybrid electric propulsion systems which would be a more promising alternative.

A Hybrid Electric Distributed Propulsion (HEDiT) system offers a promising alternative to the all-electric system. It leverages the benefits of DEP when coupled with a hybrid electric system. The benefits of pursuing such an avenue materializes as significantly reduced emissions and fuel burn along with reduced direct atmospheric heat release, key tenets when tackling the issue of environmental responsibility. Also, observed in parallel is the reduction of total energy consumption and total energy costs.

1.4 Hybrid Electric Propulsion Systems

Several Hybrid-Electric architectures can be utilized for the purpose of HEDiT. A thorough investigation of such architectures are presented in the works of Felder [14], Cinar et al. [15] and Strack et al. [16].

Figure 1.8, Figure 1.9, and Figure 1.7 show some of the architectures, i.e., series, parallel, and turboelectric architectures.
The estimation of efficiencies for the hybrid electric architecture is shown in the Figure 1.11 and Figure 1.12.

The efficiencies of the architectures were calculated using simple component estimates, assuming equal power contribution by both energy sources at sea level. The parallel architecture has an efficiency of 51% while the series architecture has an efficiency of 47%, the breakdowns are shown in the efficiency estimation figures. It is interesting to notice that the range of efficiencies falls between the efficiencies of fully electric (~73%) and turboprop (~39%) architectures.
Similarly, the turboelectric architecture, has an efficiency of 31%. This value is less than an air breathing architecture, as there are associated generation and conditioning losses, also the energy source is solely hydrocarbon fuel.

A parallel hybrid electric architecture is mechanically complex due to the coupling of the electric motor and turbine engine. The series architecture the most efficient option with no added complexities while provides several of the benefits of a fully electric system and it was strongly considered.

However, the issue arises with the design of the controller as matching or coupling the response time of motor and turbine could be difficult. Further consideration of architectures using a higher percentage of existing technology would be prudent.
1.5 Focus on Conceptual Design

The alternative hybrid-electric architecture that is explored is the Parallel Power Split (PPS) architecture due to the simplicity associated with this architecture. Also, for a 50% hybridization PPS has an efficiency of 55%. Figure 1.10 illustrates the Parallel Power Split architecture. For the purpose of this research the PPS architecture is used.

A solution to enable the consideration of such Hybrid Electric powered commercial transport aircraft in the immediate future, through maturation of robust MDO and conceptual design capabilities with the accounting of multiple levels of hybridization, would be indispensable.

1.5 Focus on Conceptual Design

The quality of data during the conceptual design phase is of prime importance. Prior studies dating back to the 1990s have shown that more than two-thirds of the total Life Cycle Cost (LCC) would have been committed during the conceptual design phase. Figure 1.13 depicts the cumulative percent of LCC against the product development phases. Also, nearly three-fourths of the manufacturing to be incurred by the product would be a direct determination based on the decisions made during the conceptual design phase.

Therefore, sound decisions early on lead to higher quality of the end product.

Figure 1.13 Dependence of cumulative Life Cycle Cost commits along with actual cost on produce lifecycle development phase (Reference [17]). Used under Fair Use 2018
Note also that design changes in the conceptual and preliminary design phases cost much less than in later phases.

The availability of quality data would undoubtedly enable better informed choices at the initial phases thereby reducing the possibility of a major revamp of design at later stages of the Product Life Cycle.

Robust, quick and accurate conceptual design tools, therefore, are necessary to make sound decisions. With a plethora of available analysis tools, the primary task would be to identify capabilities that would suit the aforementioned description. As is always with Aircraft, tradeoff is the key, trading high fidelity and high speed methods to find an optimum fit is the need of the hour.

As such, this work extends a strong focus on scoping out various methods for analysis and optimization.

### 1.6 Research Scope

This research is primarily focused on the need to develop a mature conceptual design capability for HEDiP aircraft design. Such capability is necessary to investigate potential benefits of optimal regional transport aircraft with hybrid electric distributed propulsion systems.

A mature conceptual design capability that is effective and economic will be based on a proven multidisciplinary design optimization (MDO) framework. The resulting capabilities will demonstrate the need for conceptual design of a regional transport aircraft (RTA) with a HEDiP system.

Distributed Propulsion and Hybrid Electric technologies, as identified in Sections 1.2 and Section 1.4, offer significant benefits in multiple realms. These benefits include aerodynamic performance improvements leading to gross weight reduction; aero – acoustic benefits leading to low noise; augmented controllability; fuel efficiency improvements for reductions in environmental impact; better propulsive efficiency and improved maintainability among several others.

One of the key requirements is fast and accurate estimation of wing aerodynamic characteristics in the presence of multiple propellers. For the method to be suitable for MDO, it must have rapid turnaround time and also be able to capture major wing–prop interaction effects with sufficient accuracy.
In addition, the requirement of tools to optimize the HEDiP configurations, based on the number, location and characteristics of the distributed propellers has been identified.

1.7 Research Objectives

The present research effort has four major objectives.

1. Enhance a Multi-disciplinary Design Optimization framework for conceptual design studies of HEDiP Aircraft.
3. Optimize number, location and rotational orientation of HEDiP Propeller–Wing configuration.
4. Quantify potential benefits of a HEDiP Regional Transport Aircraft Design.

1.8 Research Approach

The approach to achieving the objectives is highlighted in this section.

1. Identify MDO framework and update the existing aerodynamics module to account for the effect of wing – propeller interaction.
2. Identify and modify a suitable aerodynamic code for estimation of wing-propeller aerodynamics.
3. Develop an optimization module and perform a wing propeller configuration optimization.
4. Design a RTA with HEDiP and compare it with a conventional twin turboprops design.

Implementation of the outlined methodology along with a brief description of the decision processes are provided in the following chapters.

1.9 Thesis Outline

The thesis has 6 Chapters including the Introduction. Chapter 2 addresses the identification of a suitable Multidisciplinary Design Optimization frameworks. Chapter 3 highlights the considerations and
approaches to simulate wing-propeller interaction, while underlining the need for this facet of the research and perusing readily available techniques for the same.

Chapter 3 also describes validation studies for the selected aerodynamic technique, and to understand the limits of this software by carrying out characterization studies.

Chapter 4 describes the conception of the Wing Propeller Configuration Optimization (WiPCO) module along with its application.

Chapters 5 addresses the integration of WASPE and the selected MDO framework. A case study is presented to fully appreciate the benefits of HEDiP aircraft through DEP and benefits for HEDiP aircraft design owing to the WASPE-MDO coupling.

The conclusions drawn of the assessment of results from the present study along with areas of further exploration are summarized in Chapter 6.
Access to a low-fidelity propeller-wing aerodynamics prediction tool opens several avenues of exploration, while extremely useful to obtain ballpark estimates for design cases. Another very lucrative opportunity lies with the coupling of this analysis with an optimization module. The ability to quickly and accurately predict general trends for conceptual design could help in exploring a wider design space. Identification of optimum solution(s) for a particular design problem with a certain level of accuracy could help locate regions where deeper investigations can be carried out through high fidelity tools.

Several aircraft conceptual designs invoking the potential of DiP technologies including X-57 Maxwell [2] have emerged. However, the propulsion units are usually equally spaced with the same vertical location of all propeller centers. It has been ascertained that the aerodynamics of wing – propeller configurations can be highly sensitive to location [18],[19]. An interesting investigation would be the identification of an optimum wing – propeller configuration for a particular Distributed Propulsion design. The limitation of WASPE in its inability to accurately estimate vertical location variation restricts the broad problem.

Aerodynamics of such configurations although very fascinating, the designing of the desired regional transport aircraft with HEDiP systems requires a more inclusive multidisciplinary design study. In order to carry out such an analysis, the incorporation of the WASPE method in a Multidisciplinary Design, Analysis and Optimization Framework would be the logical choice.
2.1 MDO Frameworks

Several MDO/ MDAO frameworks exist for aircraft conceptual design studies such as NASA’s OpenMDAO, NASA Flight Optimization System (FLOPS) and Pacelab Aircraft Preliminary Design (APD). Although all these frameworks have been widely used for Aircraft Design, the current study investigates the usability of FLOPS and APD for HEDiP aircraft design studies.

The works of Riggins [20] and Locatelli et al. [21] address the relative merits of Pacelab APD and NASA FLOPS MDO frameworks.

The following criteria were utilized in the selection of the framework in this research.

1. **Capability** of the MDO frameworks including but not limited to compatibility with external codes, data interaction methods, user experience along with geometry modelling, analysis and optimization capabilities and libraries.

2. **Modularity** is another key factor considered for the selection of the Frameworks, the ability to switch between analysis and performance estimation techniques would be indispensable for unconventional aircraft design exploration. Ability to account for eccentric user defined design methodology is also considered in estimating the usability of the programs.

3. **Maturity**: While MDO frameworks are extremely effective at addressing the objectives they were initially designed for, the dynamic and progressive nature of aerospace technologies necessitates rapid incorporation and reflection of recent technological advances.

4. **Ease of Update** is another facet of the above criteria, is the ease of implementation of novel methods by users for either increased fidelity or non-conventional aircraft design such as blended wing body and Distributed Propulsion aircraft designs. The conformance of the frameworks to this notion is also assessed.

5. The presence or capability of the **graphical user interface** is directly correlated the ease of usage of the software and faster tracing of the associated learning curve. While exceedingly convenient, this criterion is not very critical in the decision process.

6. **Reliability**: Diverse investigation avenues while extremely useful, the validity of the results being produced must be taken into consideration as only medium – low fidelity estimations are performed. Wide user base and demonstration of accurate performance estimation for existing designs would help build confidence in the reliability of the frameworks. Another avenue that can
be considered is the community trust in the component analysis techniques, usage of widely accepted methods for individual modules would be highly coveted.

Some additional criteria that were considered for the evaluation were the ready availability of the latest release of the frameworks along with user support.

2.1.1 NASA Flight Optimization System (FLOPS)

The Flight Optimization System (FLOPS) [22] developed by NASA is one of the oldest and widely used codes for conceptual aircraft sizing, preliminary analysis and optimization. Materialized by McCullers [23] in FORTRAN, FLOPS has been under continuous development at the NASA Langley Research Center. Availability of FLOPS, while restricted. A public release version of FLOPS is available for use and potential modification. The FLOPS Public Release Version 9.0.0 consists of primarily 6 modules including Weights, Aerodynamics, Performance scaling of propulsion systems, Performance analysis for specified design mission, Takeoff and landing estimation along with a Program control module. Acoustic estimation, engine cycle performance and cost analysis modules are not provided with this version.

An in depth documentation of the program is included in the FLOPS package with intricate details of the numerous parameters and their behavior. FLOPS uses a text based file transfer for data interaction, the input and output communication is through files.

XFLOPS, an iteration of FLOPS with graphics, while in existence was not available to be leveraged for the benefit of this research. The lack of a dedicated graphical user interface (GUI) in the stock version coupled with possibilities and parameter combinations that could be possible with FLOPS, would make acclimatization a long and arduous process. Further, owing to the nature of the code, a lack of technical support in comparison to commercial software which tend to readily make such assistance available, makes debugging associated with novel method introduction very difficult. The lack of a geometry visualization module necessitates the coupling with other modeling software for this purpose, visualization could help support a designer’s intuition.

The weights are estimated through a method developed in – house utilizing an extensive database available with FLOPS. The weights module uses empirical equations to predict the weight of each item. However for complex wing planforms an analytical wing weight estimation method is used.
FLOPS can accept user defined aerodynamic data. Alternatively, FLOPS utilizes the method developed by Feagin and Morrison [24] for aerodynamic data generation. The Empirical Drag Estimation Technique (EDET) is coupled in FLOPS with an enhanced Reynolds Number estimation module. The augmented profile drag consideration also exploits the T-Prime method for the skin friction drag prediction.

The propulsion module included parses the input flight deck, interpolates between the data and can use nonlinear and linear scaling as required, the data generated is forwarded to mission performance analysis modules.

The primary procedures are either the prescription of desired mission range to beget Maximum Takeoff Weight (MTOW) or conversely define MTOW to predict range achievable by current mission.

### 2.1.2 Pacelab Aircraft Preliminary Design (APD)

Pacelab Aircraft Preliminary Design (APD) [25], in comparison to FLOPS is a relatively recently developed framework. Pacelab APD is the product of the German based company PACE GmbH, now a part of TXT. For this research, PACE provided access to state of the art versions of the code, namely the Pacelab APD 6.2 and Pacelab APD 7 Alpha Release.

Pacelab APD is widely utilized by researcher and innovators worldwide. It is continually updated and includes recent technological advances. An interesting feature of Pacelab is the unique solution engine that allows for a plethora of analysis and design possibilities. The approach is based on declarative design where the user decides the design methodology by prescribing the parameters that should be inputs and those that have to be estimated. Another interesting feature is the ability to rapidly setup design sensitivity studies, which is particularly useful in the design space exploration of radical aircraft designs.

APD offers a powerful GUI with a structured and easy to understand parameter prescription interface, Figure 2.1 depicts the user interface of Pacelab APD 7.
The aerodynamics and weights modules are heavily inspired by the works of Torenbeek [26] and Raymer [27]. APD also leverages its extensive database to accurately predict the weights. APD uses a more accurate aerodynamic performance estimation model in comparison to FLOPS, where the methods used are extensively validated but do not reflect contemporary developments.

APD includes a library of available computational methods which can be tapped for the analysis of the specified Engineering Object or aircraft component.

Performance estimation of HEDiP aircraft require the implementation of modified methods. This is made easy with ready access to the computational library wherein individual methods can altered or simply replaced.

Recent works by Cinar et al. [15] indicate the potential of the Pacelab suite in general possess for tackling the Hybrid Electric problem.

A comparison of FLOPS, OpenMDAO, and APD based on the criteria in Section 2.1 is depicted in the figure of merit matrix shown in Table 2.1.

**Figure 2.1 Pacelab APD 7 Alpha Graphical User Interface**
The figure of merit matrix used a very coarse grading scale, with 1 being poor, 2 being average and 3 depicting good, the framework with the highest score would be the most suitable fit. The results of the FOM matrix suggested the use of Pacelab APD for addressing current design problem involving HEDiP.

The coupling of Pacelab APD and WASPE along with the Data Flow are discussed in Chapter 5.
CHAPTER 3

WING – PROPELLER AERODYNAMICS

The problem of wing mounted propellers has been studied extensively since the 1900s. Kroo [18] and Witkowski et al. [28] indicate that the mutual interaction of the flow between the wing and propeller could result in beneficial aerodynamic characteristics. Miranda and Brennan [29] share interesting insights on the propulsive benefits along with the induced drag reduction obtained by mounting propeller and turbines on the wingtips. A more recent investigation by Veldhuis [19] further adds to the understanding of the aerodynamic interference between a propeller and a wing through detailed. Other works by Patterson [13] and Stoll et al. [4] also provide insights into modern application and analysis techniques for wing-propeller interaction.

The propeller slipstream development addressed in Section 3.4 and the estimation of wing aerodynamics in Section 3.2. The effects of wing–propeller interaction are introduced along with the development of the aerodynamic module WASPE to account for this interaction in Sections 3.1 and 3.5 respectively. Also, validation and characterization of wing–propeller aerodynamics using WASPE are presented in Section 3.7.

3.1 Wing–Propeller Interaction Effects

The presence of a propeller and wing in close proximity to each other alters the isolated performance of both. Several studies in the past by Miranda [29], Witkowski et al. [28], Loth [30] and Veldhuis [19] among others have explored these effects in great detail. Renewed interest in this prior well explored concept arises
from the increasing proclivity of the aerospace community towards e-aviation. Key findings of the prior work are briefly described in this section.

The presence of a wing upstream of a pusher propeller has been understood to have positive contributions to the thrust produced by the pusher propeller. In comparison, the tractor propeller with the wing directly downstream of the propulsion system significantly alters the lift and induced drag characteristics. Several studies by Miranda [29], Veldhuis [19], Epema [31] and Alba [32] have shown that this phenomenon augments the aerodynamic efficiencies of the propeller – wing configurations, with increased lift and diminished induced drag.

The jump in dynamic pressure is primarily due to the axial component of the slipstream which strengthens the incoming freestream velocity, this works in increasing the forces on the wing. This is the primary component that is expected to be leveraged in the X-57 Maxwell Distributed Electric Propulsion aircraft. The change in wing performance due to only the axial velocity can be observed in Figure 3.16.

It is also interesting to note that this performance variation is not just limited to the region of the wing directly downstream of the propeller, the effects also spreads across the span. This is evident even in low fidelity potential theory based applications where the adjacent areas do not have any direct exposure to the slipstream. The possible explanation to this can be obtained by the potential theory, the augmented freestream velocity in the area subjected to the prop wash would mean that a normal velocity of higher magnitude would have to be countered by the wing vortex system. This would naturally mean a greater circulation at these locations to satisfy the boundary conditions and thus more lift. The stronger vortices induce higher velocities at all the control points including the regions not subjected the slipstream, which would once again translate to a higher circulation in these areas. The magnitude of increase is inversely proportional to the distance from the region experiencing the slipstream in accordance with the Biot – Savart law and is evident in the spanwise lift distribution.

As the axial component tends to act in the same direction on both halves of the propeller, i.e., the upward rotating blade and downward rotating blade regions, the sense of propeller rotation has no bearing on the change in wing aerodynamics due to the dynamic pressure increase. However, the performance change due to the axial component could be sensitive to the spanwise location of the propeller.
3.1. Wing – Propeller Interaction Effects

Figure 3.1 Variation in lift distribution due to the presence of a tractor propeller rotating in an inboard up manner (Veldhuis [19]).
Used under Fair Use 2018

The tangential component of the propeller slipstream acting in the circumferential direction, normal to the axial velocity tends to modify the direction of the incoming freestream, and modifies the local sectional angle of attack. The direction of the swirl velocity follows the direction of motion of the propeller blade and an upward going blade induces an upward velocity at the wing. This increases the angle of attack at the sections of wing behind the upward moving blade. Similarly, the downward moving blade causes a local angle of attack decrease and thereby reducing the lift produced there. This antisymmetric behavior observed in the lift distribution can also be observed for the induced drag, although in the opposing directions, with some exceptions. The upward going blade when modifying the freestream tilts the force vector forward, thereby creating a negative drag. The effect of the downwash from the downward going blade is to tilt the force vector backward, in case of positive lift and thus have a positive induced drag contribution. Veldhuis [19] argues that as the forward tilted vector is strengthened while the backward tilted vector is weakened, a net reduction in the induced drag is observed in the presence of a propeller.

Although the presence of a wing immediately downstream of a propeller prescribes changes in the propeller performance, as the wing modifies the upstream flow conditions, the current study leverages a “One-way interaction” model. The function of the slipstream of the propeller is to modify the apparent upstream velocity seen by the wing from its freestream value. Thus the action of the propeller slipstream on the wing can be observed in a piecewise fashion, the jump in dynamic pressure over the wing and the altering of the local sectional angle of attack.
The Wing-Propeller interaction can be modeled by a multitude of techniques, understanding the suitability of these methods to the current research must be addressed. Potential analysis techniques capable of wing-propeller aerodynamic simulation are investigated based on the wing aerodynamic estimation capabilities.

### 3.2 Wing Aerodynamic Estimation Techniques

Prediction of finite wing aerodynamic properties dates back to the early 20th century, initially established by Ludwig Prandtl [33] as the extension of the infinite wing results to a finite wing.

Currently, however, aerodynamic performance of the lifting surface can be estimated by several techniques at varying levels of precision and associated costs.

Wind tunnel testing is typically considered to be the most desirable for accurate estimation of wing aerodynamic characteristics. The expense in terms of resources and time, make it unsuitable for design trade studies and for applications based on optimization techniques. In contrast, computational methods can be fast and well suited for optimization studies.

Computational aerodynamics approaches have become increasing indispensable in modern day aircraft design and analysis. The increasing ease of availability and affordability along with rapidly developing computational capability has only further strengthened such dependence. Expedited assessment of the performance of several diverse design configurations at varying conditions while levying a significantly damped resource penalty has drawn the proclivity of modern aircraft designers.

Development and analysis of airfoils, wings and entire aircraft, evaluation of stability and control, optimization routines involving multiple objectives among others are some examples of the areas where contemporary computational approaches can be useful. The work of Cummings et al. [34] offers an in-depth explication of computational aerodynamics.

Numerical solution methods of a set of governing equations such as the Navier-Stokes and Euler equations fall under the purview of Computational Fluid Dynamics (CFD). A detailed discourse on CFD techniques and applications can be found in the works of Anderson [35] and Tannehill et al. [36]. The high computational costs and associated slow turnaround time of CFD restrict its consideration in design exploration studies. Thus a more inexpensive technique is required.
This brings about the consideration of the Lifting Line Theory (LLT). In accordance to the theory, also known as the Prandtl’s Lifting Line Theory, the finite wing is replaced by a system of bound horseshoe vortices creating a vortex sheet. The bound segments extend along the span of the wing and the trailing segments extend infinitely downstream, as shown in Figure 3.2, representing the streamwise shedding of vorticity as necessitated by Kelvin’s circulation theorem – accounting for the spanwise variation of circulation.

Meticulous descriptions of the theory can be perused in the aerodynamic texts by Karamcheti [37], Sadraey [38] and Bertin and Cummings [39], only a few important takeaways are presented here.

Potential flow theory being the core of Prandtl’s work, begets results that do not account for the viscous effects of the flow field. Therefore, only lift and induced drag can be estimated. This Lifting-Line Theory (LLT) is only pertinent to straight finite wings with high aspect ratios and incompressible flow regimes. Although this method has been extensively utilized, a severe handicap of this rapid and reasonable technique is the inability to predict aerodynamic stall.

Aerodynamic models with a higher fidelity in comparison to the classical LLT are required to analyze lifting surfaces with low aspect ratios and sweep. This leads us to another facet of potential flow based solutions, the lifting surface methods – an extension of the lifting line methods, to include spanwise and
chordwise placement of singularities. Such methods while more computationally expensive than LLT –
with each typical run taking several minutes, is still significantly faster than CFD methods.

Similar to LLT, the assumption of ideal flow results in inviscid solutions. The lifting surface is discretized
into a grid of trapezoidal panels lying on the mean camber surface, with either source, vortex or doublet
singularities associated with each panel.

Widely used among these methods is the vortex lattice method, where the wing is represented by a mesh of
horseshoe vortices. The velocities induced by all the vortices at specified control points along with the
Neumann boundary conditions – owing to the impenetrability of the lifting surface, lead to the estimation
of circulation. The consequent application of the Kutta – Joukowskii theorem results in the estimation of the
aerodynamic forces. VLM solutions can be used to approximate the lift, induced drag and coefficients of
moment of a wide variety of lifting surfaces with taper, sweep, dihedral and twist.

Airfoil camber effects can be duly accounted for in VLM, although similar to LLT the influence of thickness
is neglected and the aerodynamic stall is also not predicted.

Limited capabilities associated with the fast LLT estimations and high cost of resources concomitant with
higher order CFD, lead to the selection of the Vortex Lattice Method (VLM). The VLM has the right trade-
off between accuracy, capability and speed for use in design exploration in early studies.

### 3.3 Vortex Lattice Method Package Selection

A plethora of Vortex Lattice Method codes has been developed for use in aircraft aerodynamic performance
analysis, most of which have been validated for several design cases. Several VLM codes were considered,
including Athena Vortex Lattice (AVL) [40], Tornado VLM [41], VSPAero [42], XFLR5 [43] and QuadAir
[44]. Many of these codes have been widely used by the aerospace community and have also been
sufficiently validated for several test cases.

For licensing, the ability to integrate the VLM software with commercial MDO frameworks was considered
tantamount. The decision matrix for the selection of the VLM code, based on attributes, is presented in
Table 3.1. The criteria against which the codes were compared for suitability with the current study are
included.
### Table 3.1 Decision Matrix for VLM code selection

<table>
<thead>
<tr>
<th>Attributes</th>
<th>AVL</th>
<th>QuadAir</th>
<th>VSPAero</th>
</tr>
</thead>
<tbody>
<tr>
<td>Programming Language</td>
<td>FORTRAN</td>
<td>MATLAB</td>
<td>C++</td>
</tr>
<tr>
<td>Rotor/ Disk integration</td>
<td>No</td>
<td>No</td>
<td>Yes</td>
</tr>
<tr>
<td>Physics Conformity</td>
<td>Widely Demonstrated</td>
<td>Limited Demonstration</td>
<td>Widely Used</td>
</tr>
<tr>
<td>Execution Speed (1 Run)</td>
<td>Very Fast</td>
<td>Moderate</td>
<td>Fast</td>
</tr>
<tr>
<td>Learning Curve</td>
<td>Steep</td>
<td>Moderate</td>
<td>Above Moderate</td>
</tr>
<tr>
<td>Versatility/ Features</td>
<td>Good</td>
<td>Poor</td>
<td>Very Good</td>
</tr>
<tr>
<td>Ease of Adding New Features</td>
<td>Moderate</td>
<td>Easy</td>
<td>Hard</td>
</tr>
<tr>
<td>Ease of VnV/ Debugging</td>
<td>Moderate – Hard</td>
<td>Moderate</td>
<td>Very Hard</td>
</tr>
<tr>
<td>Community Trust</td>
<td>High</td>
<td>Low – Moderate</td>
<td>Moderate – High</td>
</tr>
<tr>
<td>Licensing</td>
<td>GNU GPL</td>
<td>N/A</td>
<td>NOSA</td>
</tr>
</tbody>
</table>
The AVL, Tornado VLM and XFLR5 packages while being very capable, were attached with a GNU General Public License [45], an interpretation of which proved to be unacceptable for the aforementioned use.

The careful deliberation of the above selection criteria initially led to the selection of VSPAero, the aerodynamic analysis module of the OpenVSP suite developed by NASA and released as open source. The most coveted features included existing propeller wing analysis capabilities which could be enhanced to utilize user supplied propeller geometric data, along with the mechanism to interface with external applications.

A simple characterization of VSPAero was conducted for its existing propeller-wing aerodynamic prediction capabilities. A typical propeller-wing configuration was modeled where only half the wing was considered, as shown in Figure 3.3, with certain key parameters of the geometry documented in the Table 3.2. The results obtained from this study are depicted in the Figure 3.4, where the coefficients of lift and drag are plotted against the propeller location along the semi wing span.

### Table 3.2 Geometric parameters of the VSPAero Propeller-Wing model

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Wing Chord (C)</td>
<td>2 m</td>
</tr>
<tr>
<td>Wing Semi-Span (b)</td>
<td>10 m</td>
</tr>
<tr>
<td>Airfoil</td>
<td>NACA0012</td>
</tr>
<tr>
<td>Propeller Diameter (Φₚ)</td>
<td>3 m</td>
</tr>
<tr>
<td>Advance Ratio (J)</td>
<td>0.8</td>
</tr>
<tr>
<td>Thrust Coefficient (Tₖ)</td>
<td>0.4</td>
</tr>
<tr>
<td>Propeller Vertical Location (Z - Axis)</td>
<td>0 m</td>
</tr>
<tr>
<td>Propeller Streamwise Location (X - Axis)</td>
<td>-0.5 m</td>
</tr>
<tr>
<td>Propeller Spanwise Location (Y -Axis)</td>
<td>-1 m to -10 m</td>
</tr>
</tbody>
</table>

Figure 3.3 Propeller - Wing model used for spanwise variation study in VSPAero, the arrow indicates the direction of the propeller location variation.
The results show erratic trends as evident in the Figure 3.4. The method, therefore, is not suitable for any kind of optimization study. Also, noticeable sensitivity to the predicted results are observed when the propeller location was varied, even by 1% of the wing span at certain locations.

![Figure 3.4 Aerodynamic behavior of propeller spanwise location variation, as apparent from VSPAero, a few noticeable outliers are highlighted](image)

The exhibited characteristics prove to be well beyond the expected behavior for such a variation and was concluded as less than satisfactory for use in the current application.

The necessity of a robust aerodynamic estimation software led to the development of Wing Aerodynamic Simulation with Propeller Effects (WASPE). For this development, QuadAir, developed by Roberto A. Bunge at Stanford University, was selected, which utilizes the MATLAB platform. QuadAir while lacking the capability to account for propeller slipstream, rated well for the implementation of new methods and modifications. As such, propeller effects would have to be incorporated in QuadAir.

Section 3.4 addresses the different methods available for propeller slipstream development along with the computational simulation software for the slipstream simulation.
3.4 Propeller Slipstream Modeling

The propeller induced velocities consist of axial and tangential components that can be estimated through experimental methods and CFD techniques. The axial induced velocities are produced along the axis of the propeller and cause changes in the dynamic pressure over the region of the wing blanketed by the slipstream. Axial induced velocities generally tend to attain minimum values, approaching null, around the tip and root of the propeller blades, while achieving higher magnitudes near the radially centric portion of the blade, as seen in Figure 3.5. The generation of the axial component of the induced velocity can be attributed to the momentum jump across the propeller while creating thrust. Several traditional theories, such as the actuator disk theory, provide a good approximation of the increase in the scale of the axially induced velocities from far upstream – which are near zero, to far downstream conditions along the streamtube of the propeller.

![Figure 3.5 Sample Propeller Induced Velocities (Patterson [13]). Used under Fair Use 2018](image)

Tangential components of the propeller induced velocities, also referred to as the swirl induced velocities, are perpendicular to the propeller axis in the propeller plane. Unlike axial velocities, the swirl components change the local angle of the freestream directly aft of the propeller. The magnitudes of the swirl velocities tend to be highest close to the root and diminish to near zero at the propeller tip and hub. The development of the swirl induced velocities is, however, assumed to be near instantaneous at the propeller plane, with
insignificant distance to attain the far downstream values and no occurrence of the tangential components upstream of the propeller.

Although the distribution and scale of the propeller slipstream velocities are highly dependent on the propeller geometry and operating conditions, typical profiles of the induced velocities normalized by the freestream value are depicted in Figure 3.5.

While it is interesting to note the degree of accuracy that can be achieved through experimental and high-fidelity methods, the pursuit of such avenues however, is not well suited for conceptual design investigations owing to the high cost and long turnaround time.

As such, low-fidelity methods with low computation costs are thereby preferred for conceptual design studies and can be effectively utilized for the development of the slipstream. Techniques such as Momentum (Actuator Disk) Theory, Blade Element Momentum Theory and Goldstein Solution briefly discussed in Sections 3.4.1, 3.4.2 and 3.4.3, could provide propeller data for initial design assessment studies.

### 3.4.1 Momentum Theory

Analytical propeller performance estimation techniques have proven to be extremely important for initial propeller performance estimation. Widely used among these are the Actuator Disk Theory and the Blade Element Momentum Theory, which leverage fundamental principles to assess propeller performance. Established by Rankine [46] for marine applications in the late 19th Century, the Actuator Disk Theory, also referred to as the Classical Momentum Theory is a simplified theory for the rapid estimation of propeller characteristics.

Energization of the flow, based on Newton’s laws, by the actuator disk is assumed to be the approximation to an actual propeller. This infinitesimally thin disk is assumed to consist of an infinite number of blades which do not generate aerodynamic drag. Shape of the propeller blade is also, however, not taken into consideration.

A general overview of this theory is presented, a more detailed explication of this theory can be located in the works of Glauert [47] and McCormick [48].
Figure 3.6 depicts the actuator disk model of the propeller, a large control boundary with a wall height several times the radius of the actuator disk is considered. The energy added to the freestream reveals itself as an impulsive surge in pressure across the actuator disk.

![Figure 3.6 Schematic of the Actuator Disk Model](image)

Application of continuity necessitates that the mass flow through the disk after simplification be given as

$$\dot{m} = \rho \pi r_p^2 V$$ \hspace{1cm} Eq. 3-1

Where $r_p$ depicts the radius of the propeller/actuator disk. Ideal flow is assumed with the flow being incompressible. Considering the assumption equating far upstream and downstream static pressures to the atmospheric pressure and applying Bernoulli’s theory, we arrive at the jump in pressure across the actuator disk

$$\Delta P = \frac{1}{2} \rho \left( V_{\text{exit}}^2 - V_\infty^2 \right)$$ \hspace{1cm} Eq. 3-2

The underlying theory imposes a constancy of velocity across the actuator disk. A rise in velocity and constant static pressure far downstream is explained as the influx of flow perpendicular to the freestream direction. The variation in pressure and velocity along the control volume is depicted in Figure 3.7.
Newton’s second and third laws of motion mandate the generation of a propulsive force, Thrust, which is the change in momentum in the control boundary.

\[ T = \dot{m}(V_{\text{exit}} - V_\infty) \]  
\[ \text{Eq. 3-3} \]

The thrust is assumed to be uniformly distributed about the surface of the actuator disk. This, along with the pressure jump, can be transcribed as

\[ T = \pi r_p^2 \Delta P \]  
\[ \text{Eq. 3-4} \]

Equations Eq. 3-1, Eq. 3-3 and Eq. 3-7 considered together results in

\[ V = \frac{1}{2} (V_{\text{exit}} + V_\infty) \]  
\[ \text{Eq. 3-5} \]

The propeller thrust is

\[ T = 2\pi r_p^2 \rho V_a (V_\infty + V_a) \]  
\[ \text{Eq. 3-6} \]
Where the induced velocity is given as

\[ V_a = V - V_\infty = (V_{\text{Exit}} - V_\infty)/2 \]  \hspace{1cm} \text{Eq. 3-7}

Equation Eq. 3-5 reveals that the velocity slightly aft of the actuator disk is in fact the average of its far upstream and downstream values. This in turn implies that the induced velocity immediately aft would be half the far downstream induced velocity.

The axially induced velocity may be expressed as

\[ \frac{V_a}{V_\infty} = \frac{1}{2} \left( \sqrt{1 - \frac{2T}{\pi r_p^2 \rho}} - 1 \right) \]  \hspace{1cm} \text{Eq. 3-8}

This velocity is of the same magnitude across the entire actuator disk. Furthermore, the momentum theory does not account for any rotational effects in the slipstream generation thereby producing only the axial induced velocity. Further efforts in the development of the classical actuator disk theory indeed have succeeded in accounting for the rotational slipstream effects, although such a discourse would be beyond the scope of this work.

The momentum theory while proving to be a great initial estimation tool, with quick analysis times, would however not be sufficiently accurate for design space exploration due to the limitations listed below.

- Lack of the tangential component in the propeller induced velocities
- Oversight of the presence of the propeller hub
- Inability to account for the drag component and the thrust deterioration at blade tips
Also, the absence of the propeller geometric information would make this tool counter-intuitive for designers.

### 3.4.2 Blade Element Momentum Theory

![Propeller Blade Element Geometry Schematic](image)

Figure 3.8 Propeller Blade Element Geometry Schematic (Modified from McCormick [48]). Used under Fair Use 2018

The Blade Element Theory and the Momentum Theory are simultaneously used in the Blade Element Momentum Theory (BEMT) as a means to curb the limitations of the Actuator Disk model. BEMT accounts for the geometric characteristics of the propeller, with each blade modelled as a number of differential elements with airfoil cross sections of known aerodynamic characteristics, Figure 3.8 shows the propeller blade element. Each propeller element is subjected to the influence of a resultant of the local freestream velocity, the local propeller rotational velocity and the local slipstream velocity. While the former two are definitively known, the latter is dependent on the propeller performance which is in turn relies on the local velocity of which the slipstream velocity is a component. This slipstream velocity can be initially estimated through the Momentum Theory, although only axial component is predicted, it should prove to be a reasonable starting point.

The resolution of the elemental lift and drag in the direction opposing the air inflow, results in the estimation of the elemental thrust force.
This equation is then integrated radially from root to tip and scaled with number of propeller blades to estimate the propeller thrust. However, application of the momentum theory to an annular disk of radial length \( dr \) at the location of the considered element would result in a thrust prescribed by

\[
dT = \rho V_{\infty}^2 L c (C_t \cos \theta - C_d \sin \theta) dr \quad \text{Eq. 3-9}
\]

Simultaneous comparison of elemental thrust equations leads to the discovery of the elemental axially induced velocity, an interference flow component represented as a factor of the freestream velocity

\[
\frac{V_a}{V_\infty} = 1 + a \quad \text{Eq. 3-11}
\]

Similarly, comparison of the torque on the annular segment will yield the tangential factor, at the elemental location, defined with respect to the propeller rotational velocity

\[
\frac{V_t}{\Omega} = a' \quad \text{Eq. 3-12}
\]

Thus iterating through these steps will yield the radial slipstream profile following which the propeller performance is updated until convergence is attained. While BEMT provides for a much better slipstream profiling, especially with the tangential velocity prediction, a few shortcomings adversely affect the accuracy of the slipstream. The primary concern revolves around the independence of the performance of each annular segment with that of the rest of the propeller and the subsequent autonomy of the slipstream at a location to its immediate vicinity. Tip losses and wake expansion are also not accounted for.

3.4.3 Goldstein Method

The application of the potential solution to the propeller problem was addressed by the works of Betz [] and was further developed by Goldstein [49] for propellers with two and four blades. The method assumes the propeller blades to be rapidly rotating finite wings with significant twist, and implements Prandtl’s theory to this system. Each blade is substituted with a linear system of bound vortices, representing a
circulation distribution across the blade, similar to the Lifting Line Theory. Every segment of the blade is rendered as a bound vortex of strength $\Gamma$ and, this circulation vanishes at both ends of the propeller blade. In accordance with potential flow theory and Prandtl’s assertions such a model would be accompanied by the presence of a trailing vortex surface which is shed from the *rotating wing*. The presence of the freestream velocity coupled with the rotational behavior of the blade imparts a screw-like or helicoidal shape to the trailing vortex surface, shed by every segment along the blade, the strength of which are denoted by

$$\Gamma_T = -\frac{\partial \Gamma}{\partial r}$$

Goldstein solved for the ideal propeller through the use of Watson’s Bessel functions, which preclude the solution from a generalization. An approximation was postulated earlier by Prandtl with higher degrees of accuracy associated with greater number of blades and shorter pitch of the helicoidal vortex surface.

The exact solution presented by Goldstein for a finite number of blades does not account for profile resistance, further explanations of the exact solution are detailed in the works of Goldstein [49] and McCormick [48]. The associated induced velocities obtained are given by

$$V_{axial} = -\frac{w(\mu)}{\sqrt{1 + \mu^2}}$$

and,

$$V_{tang} = \frac{w\mu^2}{\sqrt{1 + \mu^2}}$$

Where $w$ prescribes the advance velocity of the trailing vortex system parallel to the rotor axis and $\mu$ is the representation of the ratio of propeller rotational and freestream velocities.

The ability to account for finite propeller blades and blade geometry would indeed be very beneficial. Also, with the use of the vortex system would mean that the blade section performance would no longer be independent of the characteristics of its adjacent sections.
3.4.4 Computational Simulation Propeller Slipstream

The quest for a robust and quick propeller slipstream development tool that could be effectively employed in conceptual design and optimization studies lead to the consideration of the widely used software XROTOR.

XROTOR, a propeller analysis and design tool developed by Drela [50] of MIT, is a suitable source of propeller induced velocity estimation for the scope of this research. An open source FORTRAN code, XROTOR provides for circumferentially-averaged time independent results to the unsteady propeller problem. Axisymmetric distribution of the axial and swirl induced velocities at a required refinement of radial stations, with acceptable fidelity can be obtained for low computational expenses.

The propeller analysis code also accounts for the effects of geometric characteristics of the propeller being modelled – including the number of blades, twist angles and chord distributions from the root to tip, while accounting for the presence of a propeller hub where induced velocities are modelled as non-existent.

The Potential Formulation method of XROTOR is an extension of the Goldstein solution to indiscriminate blade numbers and arbitrary radial load distributions is used for the computation of the required velocities with required radial refinement. Individual blades are modelled as lifting lines, with bound vortices placed radially along the span, along with helically developing trailing vortices being extended from each blade, thereby solving for a rigid wake of helicoidal nature.

The slipstream velocities are modelled in a cylindrical propeller streamtube, thereby neglecting the contraction and shape of the developing slipstream. While the suggested method on the XROTOR platform provides for several desirable features, it also has some key shortfalls. These key shortfalls include absence of viscous effects and the lack of compressibility effects.

XROTOR can be used to simulate co-rotating and contra-rotating propellers. Such a capability could prove to be indispensable for the purpose of design studies.

3.4.5 Swirl Recovery Modeling

In addition to the variation in the wing aerodynamic performance in the presence of an upstream propeller, accounted for in the “One – Way Interaction” model leveraged by the current study, a deviation from the
isolated propeller performance and slipstream development due to the presence of the lifting surface has also been indicated in the works of Witkowski et al. [28], Kroo [18], Veldhuis [19] and Alba [32].

A strong assertion is made relating the induced upwash directly upstream of the wing to the alteration of the local angles of attack on the propeller blades – consequently modifying the propeller characteristics. Wikowski et al. prescribe that such an influence on downward traversing blade(s), would tend to increase the local sectional angle of attack, strengthening the sectional lift and accordingly the thrust produced. Conversely, an antisymmetric behavior can be observed on the upward moving blade(s), with reduced sectional local angle of attacks along with subsiding sectional lift and thrust.

The wing, behaving similar to a turbine stator recovers rotational energy from the propeller slipstream. As such, the estimation of the propeller induced slipstream velocities, in the presence of a downstream wing requires the correction of the overestimated tangential components of the slipstream velocities. The works of Veldhuis [19] and Alba [32] address in depth the reduction of the rotational velocities. This reduction is dependent on several contributing factors such as the relative distance of propeller placement, wing characteristics and operating conditions among others.

Several recovery factors have been suggested to account for the above phenomenon, although keeping in mind the simplistic nature of the propeller-wing aerodynamics module being pursued for conceptual design, a constant Swirl Recovery Factor of 0.5, as suggested by Veldhuis[19] is thereby employed for the proposed model in this Thesis.

The accounting of the SRF while insufficient to account for an all-encompassing “Two – way interaction” – seeks to enhance the results being obtained by the utilized “One – Way Interaction” model using an easily modeled full interaction feature.

3.5 Wing Aerodynamic Simulation with Propeller Effects (WASPE)

Realistic estimations of the wing – propeller interaction effects, introduced from early phases of conceptual design being the primary motivation of this study along with the lack of a suitable open source platform for the same, necessitates the conception of an in-house aerodynamic simulation tool with the aforementioned capabilities, as described in Section 3.1.
WASPE is envisioned as a modification of a VLM software, by integrating the propeller slipstream data. The slipstream data may be from a software such as XROTOR, user supplied experimental data, or analytical data. Also, beneficial would be the ability to estimate the change in the moment coefficients. The VLM tools considered and the thought process behind the selection of the preferred tool is detailed in the Section 3.3.

WASPE along with facilitating the effects of propeller presence, significantly modifies the program with altered data flow and varied methods for the estimation of forces, moments and circulation. Compressibility corrections, spanwise distribution of the aerodynamic coefficients and estimation of the stability derivatives for asymmetric configurations were also enabled. Data transfer thorough file exchange for user ease along with effective communication with external modules was introduced, further details on the implementations are discussed in the following sections.

3.5.1 Baseline VLM Formulation

The traditional VLM formulation represents a lifting surface as a grid of superimposed horseshoe vortices placed on the mean surface. The bound segment of each horseshoe panel is located at the quarter chord point of the associated trapezoidal panel, with the trailing legs extending far downstream as depicted in the Figure 3.9.

![Figure 3.9 The system of horseshoe vortices representing a finite lifting surface (Anderson [51]). Used under Fair Use 2018](image)

Also, each of these panels contain a “collocation point” at which the boundary conditions are satisfied. These collocation points are generally located at the three quarter chord point of each of the panels. The
velocities induced at these collocation points must be determined in order to satisfy the boundary conditions. The no cross flow boundary condition, enforced on each panel, ensures there is no flow across the representative surface of the wing and is given by

\[ V \cdot n = 0 \quad \text{Eq. 3-16} \]

Here \( V \) is the velocity at the control points on the panels and \( n \) is the normal vector of the panels. Any vortex filament of a fixed constant strength introduces a nearby flow field, with the velocity induced by a differential length of the segment at any point in the flow space is determined by the application of the Biot–Savart law

\[ dV = \frac{\Gamma}{4\pi} \frac{dl \times r}{|r|^3} \quad \text{Eq. 3-17} \]

Where \( dl \) is the differential length and \( r \) is the distance between the element and point of application as shown in Figure 3.10.

![Figure 3.10 Representative Vortex Filament (Anderson [51]). Used under Fair Use 2018](image)

The induced downwash velocity at a panel \( i \) due to the horseshoe vortex \( j \) is of the form

\[ w_i = I_{C_{i,j}} \Gamma_j \quad \text{Eq. 3-18} \]

\( I_{C_{i,j}} \) is the influence coefficient which would be used to construct the Aerodynamic Influence Coefficient matrix. The total velocity induced on panel \( i \) is the sum of velocities induced by all \( N \) horseshoe vortices and is given by
The velocities induced at every control point are calculated assuming unit vortex strengths and as such result in a set of linear equations of the general form of Eq. 2-15. These velocities represent the influence coefficients as the strengths are unity and thus form the Aerodynamic Influence Coefficient (AIC) matrix. The set of linear equations can be written as

\[(V_{\text{induced}} + V_\infty) \cdot n = 0\]  

Eq. 3-20

Where \(V_{\text{induced}}\) represents the velocities induced by the wing vortex lattice system in the presence of a freestream, this equation can be re-written in the terms of the AIC matrix and circulation \(\Gamma\) as

\[\Gamma = (AIC)^{-1}(-V_\infty \cdot n)\]  

Eq. 3-21

The circulation of each panel is estimated to ensure that the Neumann boundary conditions are satisfied and the actual wing induced velocities can be obtained using these actual strengths.

Once the panel strengths are estimated, the application of the Kutta – Joukowski theorem results in the estimation of the force on the panel, this is given by

\[dF = \rho \Gamma V \times dl\]  

Eq. 3-22

The panel lift and induced drag are resolved from the equation Eq. 3-22 to give

\[l = \rho \Gamma (u_\infty + u_{\text{induced}})\]  

Eq. 3-23

\[d_{\text{ind}} = \rho \Gamma (w_{\text{induced}})\]  

Eq. 3-24

These values can be integrated to give the wing coefficients of lift and induced drag. Similarly the moments can be estimated about any point \(p\) as

\[dM = dF \times r_p\]  

Eq. 3-25
The pitch, roll and yaw moments about point $p$ are given in order as

$$m_y = l \times x_p \quad \text{Eq. 3-26}$$

$$m_x = l \times y_p \quad \text{Eq. 3-27}$$

$$m_z = d_{ind} \times y_p \quad \text{Eq. 3-28}$$

The works of Margason and Lamar [52] along with that of Cummings et al. [34] can be perused for more information of the Vortex Lattice Method.

### 3.5.2 WASPE Formulation for Propeller – Wing Interaction

WASPE requires a few modifications from the traditional formulation to represent lifting surfaces subjected to the slipstream of the tractor propeller(s). The propeller – wing interaction effects as described in Section 3.1 can be simulated using a modified Vortex Lattice Method, with changes in the boundary conditions and force estimation. This section details the changes to the baseline VLM formulation along with other required treatments to assess the one-way propeller – wing aerodynamic interference.

The key change to the VLM formulation is in the boundary conditions. Here the velocity at panel control points is altered by adding the induced velocities generated by the upstream propeller(s). This velocity is given as

$$V = V_{induced} + V_{\infty} + V_{propeller} \quad \text{Eq. 3-29}$$

This change in velocity must be accounted for while determining the panel circulations. Augmenting the freestream velocity with that of the propeller(s) requires modification of the input numerical or experimental slipstream in order to obtain compatible data.

Also, the presence of multiple propellers is done in WASPE by transforming the radial slipstream data in the propeller reference plane into Cartesian velocity components in the WASPE plane of reference. WASPE identifies the control points affected by the slipstream of a propeller and determines the velocity change at
these points through a linear interpolation of the propeller slipstream data, and is repeated for all the propellers, parsing of the slipstream data for integration with WASPE is discussed in Section 3.5.3.

The change in velocity by components is given by

\[
V_x = u_{induced} + u_\infty + V_{axial} \quad \text{Eq. 3-30}
\]

\[
V_z = w_{induced} + w_\infty + V_{tangential} \quad \text{Eq. 3-31}
\]

This change in velocity is reflected in the change circulation in order to satisfy the Neumann boundary conditions, the equation to estimate the panel circulations is

\[
\Gamma = [AIC]^{-1}(-V_\infty \cdot n - V_{propeller} \cdot n) \quad \text{Eq. 3-32}
\]

The lift and induced drag now become

\[
l = \rho \Gamma (V_{axial} + u_\infty + u_{induced}) \quad \text{Eq. 3-33}
\]

\[
d_{ind} = \rho \Gamma (V_{tangential} + w_{induced}) \quad \text{Eq. 3-34}
\]

Although the formulation of the moments is not modified, the change in the forces would translate into a change in the moments as well. The capabilities of WASPE are documented in Section 3.13.

3.5.3 Slipstream Data Parsing

The input slipstream data is represented by Figure 3.11, with each data point containing the axial and tangential velocities at the respective radial location. This input set is associated with a certain radial refinement depending on the source of the slipstream, i.e., the number of discrete radial locations where induced velocity information is available.
The input is converted into another data set with the same refinement used by the lifting surfaces of WASPE. The panel locations are identified by tracking the first spanwise panel, the control point of which is under the influence of the propeller slipstream.

This is determined by the following equation

\[
R_1 = \text{Round} \left( \frac{P_y - \text{real} \left( \sqrt{r_p^2 - P_z^2} \right)}{\Delta b} \right) \quad \text{Eq. 3-35}
\]

The values \(P_y\) and \(P_z\) are the spanwise and vertical locations of the propeller, \(r_p\) is the radius of the propeller and \(\Delta b\) is the width of the equally spaced panels. Similarly, the last panel to experience the slipstream is identified by

\[
R_2 = \text{Round} \left( \frac{P_y + \text{real} \left( \sqrt{r_p^2 - P_z^2} \right)}{\Delta b} \right) \quad \text{Eq. 3-36}
\]

If either of the panel locations are estimated to occur outside the lifting surface limits, such as the case of wingtip mounted propellers, the panel location(s) violating this limit are reset to the panel closest to the
limit, such as the wingtip panel. The panels subjected to the slipstream are identified as every spanwise and chordwise panel between the identified first and last panels $R_1$ and $R_2$ as shown in Figure 3.12.

Figure 3.12 WASPE: Identifying panels subjected to prop wash, yellow panels are seen by WASPE as under propeller influence while white are perceived as not under influence

The respective control point locations are projected on to the propeller plane, the velocities experienced on them are then determined using a simple linear interpolation scheme.

The normalized radial location in the propeller reference plane $P_L$ of the control points are given by

$$P_L = \sqrt{p_z^2 + \left(\sqrt{r_p^2 - p_z^2} - (R_t \Delta b)\right)^2}$$

Eq. 3-37

The quantity $R_t$ is the calibrated panel location, which is set to 0 at the center of the propeller disk.

Figure 3.13 shows the additional slipstream velocity as seen by WASPE, where the slipstream with 30 data points is represented by two 5-point curves.
3.5. Wing Aerodynamic Simulation with Propeller Effects (WASPE)

3.5.4 Compressibility Corrections

WASPE can be effectively used to estimate the aerodynamic performance of lifting surfaces in incompressible flow regime (typically $M \leq 0.3$).

However, for compressible subsonic flow the predicted results tend to deviate from the actual characteristics.

In order to account for changes in the altitude and velocity, reasonably simple factors of correction could be introduced to the incompressible solution; these estimations of compressibility are postulated as the
Compressibility Corrections. Several correction approaches were considered, with three of the most suitable methods discussed for the purpose of this study.

The earliest of these corrections proposed in the 1900s to treat incompressible data, the Prandtl-Glauert correction method, has been widely used for subsonic flows in the compressible regime. This treatment relies on the linearized perturbation velocity potential equation, with application extended only to subsonic flows. The theory fails when applied to transonic flows, e.g., with freestream Mach numbers $> 0.7$.

Readers are directed to the informative texts by Cummings et al. [34], Bertin and Cummings [39] and Anderson [51], a brief synopsis of the methods implemented in WASPE is provided in this section.

The Prandtl – Glauert treatment leverages the assumptions of thin airfoils and small angles.

This is implemented through the Prandtl – Glauert equation given by

$$\beta^2 \phi_{xx} + \phi_{yy} = 0 \quad \text{Eq. 3-38}$$

Where the correction $\beta$ is defined as

$$\beta = \sqrt{1 - M_{\infty}^2} \quad \text{Eq. 3-39}$$

The Prandtl – Glauert equation can be subjected to spatial transformations in order to obtain the Laplace’s Equation in the transformed space representing an incompressible flow to which the subsonic compressible flow is mapped. The pressure coefficient can be written as

$$C_{P,PG} = -\frac{1}{\beta} \left( \frac{2\hat{u}}{V_{\infty}} \right) = \frac{C_{P0}}{\sqrt{1 - M_{\infty}^2}} \quad \text{Eq. 3-40}$$

Aerodynamic forces and moments are directly dependent on the pressure distribution, being integrals of the same. The Prandtl – Glauert Rule given by the equation above can also be extended to forces and moments which are generated on account of surface pressures, as such the viscous component of drag cannot be corrected this way. The application of the Prandtl – Glauert correction factor is illustrated in the equations below.
\[ C_{L, PG} = \frac{C_{L, Incompressible}}{\sqrt{1 - M^2_{\infty}}} \quad \text{Eq. 3-41} \]

Also,

\[ C_{m, PG} = \frac{C_{m, Incompressible}}{\sqrt{1 - M^2_{\infty}}} \quad \text{Eq. 3-42} \]

A comparison between the compressible and incompressible results is presented in Figure 3.14.

The speed of sound utilized to calculate the freestream Mach number at a particular altitude is given by

\[ V_S = 331.3 \sqrt{1 + \frac{T_A}{273.15}} \quad \text{Eq. 3-43} \]
The temperature variation with the altitude is assumed to be piecewise linear and is stored as a data table for WASPE to access, this variation is depicted in the Figure 3.15 below.

![Figure 3.15 Piecewise linear approximation of the Temperature variation with Altitude](image)

It is important to note that the Prandtl-Glauert corrections are valid only for slender bodies and only provide reasonable predictions for relatively thin airfoils at lower values of free-stream Mach numbers.

While the Prandtl-Glauert corrections are widely utilized, an effort was made to improve the accuracy and range of application. One such improvement was proposed by Karman and Tsien [53]. The proposed method utilizes a Specific Heat Ratio \( \gamma = -1 \), which does not represent any gas, but instead depicts the usage of a linear pressure – density variation.

The modification to the correction factor as proposed by Karman and Tsien is determined by
\[ C_{P,KT} = \frac{C_{P0}}{\sqrt{1 - M_\infty^2} + \frac{M_\infty^2}{1 + \sqrt{1 - M_\infty^2}^2} C_{P0}} \]  

Eq. 3-44

From equation Eq. 3-44 it is clear that for very small values of \( C_{P0} \), the Karman-Tsien rule reduces to the Prandtl-Glauert rule. Several studies indicate the higher level of correlation with experimental data by utilizing the Karman-Tsien rule for wider applications in comparison to the original Prandtl-Glauert rule.

Another improvement to the Prandtl-Glauert theory was proposed by Laitone [54], A strong assertion was made advocating for the utilization of the local Mach numbers in lieu of the freestream Mach numbers. This would help in obtaining a more accurate estimate of the local compressibility effects on any finite body. Also, unlike the Karman-Tsien method, Laitone utilizes the exact specific heat ratio at the operating conditions. The compressibility correction factor obtained following Laitone’s approach is given by

\[ C_{P,LT} = \frac{C_{P0}}{\sqrt{1 - M_\infty^2} + \frac{M_\infty^2}{\sqrt{1 - M_\infty^2}^2} \frac{C_{P0}}{2} \left(1 + \frac{\gamma - 1}{2} M_\infty^2\right)} \]  

Eq. 3-45

Although Laitone’s rule better estimates the effects of compressibility, determination of the specific heat ratio at various operational conditions would be necessary, owing to the dependence of the specific heat ratio on local temperature. This would require the use of another data table similar to the one being used for temperature. Therefore, the Karman-Tsien rule which utilizes a correction factor independent of the specific heat ratio is preferred. WASPE can leverage any of the three methods at the discretion of the user, while the Karman – Tsien approach is left as a default recommended setting.

3.6 WASPE: Momentum Theory

The slipstream estimated from the momentum theory is used as the input for WASPE, where a simple rectangular wing is buffeted by the constant axial velocity of an actuator disk. The axial induced velocity is estimated to be about 10% of the freestream velocity for this case. The resulting lift distribution is revealed in Figure 3.16.
The demonstrated behavior appears to diverge from what has been expected of the aerodynamics of a wing downstream of a propeller, justifying the need for the improved slipstream development model.

### 3.7 WASPE Validation and Characterization

Utilization of WASPE for the purpose of this research would require sufficient amount of confidence in the validity of the results being produced. To demonstrate this, WASPE results are compared with available published data. Also, the characterization of WASPE grid sensitivity is carried out. Sections 3.8 and 3.9 present the results.

Further, an understanding of the capabilities of the code through in depth characterization can be helpful to identify areas of application with high, medium, low or no confidence and as such highlight and prioritize areas where improvements and additional development is necessary.

### 3.8 Grid Sensitivity Study

The purpose of this grid study is to characterize the sensitivity of the results to grid density.
The grid variation study involved the consideration of five chordwise refinements ranging from a coarse 10 chord divisions to a fine refinement of 30. Similarly, five spanwise refinement points were chosen with the least number of span divisions equal to 30 and the maximum being 90 divisions. The resulting 25 data points, with a broad range of 300 to 2700 panels. The analyzed cases are tabulated in Table 3.3.

<table>
<thead>
<tr>
<th>Chordwise</th>
<th>Panels</th>
<th>30</th>
<th>45</th>
<th>60</th>
<th>75</th>
<th>90</th>
</tr>
</thead>
<tbody>
<tr>
<td>10</td>
<td>300</td>
<td>450</td>
<td>600</td>
<td>750</td>
<td>900</td>
<td></td>
</tr>
<tr>
<td>15</td>
<td>450</td>
<td>675</td>
<td>900</td>
<td>1125</td>
<td>1350</td>
<td></td>
</tr>
<tr>
<td>20</td>
<td>600</td>
<td>900</td>
<td>1200</td>
<td>1500</td>
<td>1800</td>
<td></td>
</tr>
<tr>
<td>25</td>
<td>750</td>
<td>1125</td>
<td>1500</td>
<td>1875</td>
<td>2250</td>
<td></td>
</tr>
<tr>
<td>30</td>
<td>900</td>
<td>1350</td>
<td>1800</td>
<td>2250</td>
<td>2700</td>
<td></td>
</tr>
</tbody>
</table>

A half wing with a single tractor propeller, rotating inboard up, is considered. A symmetry boundary condition is used at the root of the wing. The geometry used for this study is depicted in Figure 3.17.
3.8.1 Wing Lift Coefficient (C_L)

The variation of coefficient of lift with number of panels on the wing is seen below in Figure 3.18. Initial grid refinements result in minute corrections, the magnitude of these correction decrease with further refinement until a constancy of six significant digits is achieved. The optimal number of wing panels based on speed and accuracy tradeoff would be around 1800.

![Graph showing the variation of coefficient of lift with number of panels on the wing](image_url)

Figure 3.18 Dependence of the Wing Lift Coefficient (C_L) on the number of wing panels, only 6 points are shown for figure clarity

The range of the wing coefficient of lift (C_L) is between 0.287 for the coarsest grid and 0.285 for the finest solution. Also with the recommended the coefficient of lift is 0.28547 with about 0.12% difference from the finest value.
3.8.2 Wing Pitching Moment Coefficient ($C_{M,C/4}$)

The change in pitching moment coefficient about the quarter chord is depicted in Figure 3.19. No variation in the coefficient of quarter chord pitching moment was observed with panel refinement, this would indicate that a lower refinement can be utilized without any significant impact on accuracy.

The invariance of the wing pitching moment coefficient at quarter chord is such that the result produced is -0.003707 at 300 panels and -0.003703 at 2700 panels. Similarly, the change in the normalized rolling moment coefficient on refinement resulted in extremely small changes.

![Figure 3.19 The variation of the wing coefficient of pitching moment about quarter chord on number of panels, 6 out of 25 data points shown for simplicity](image)

3.8.3 Wing Induced Drag Coefficient ($C_{D,i}$)

The next avenue of investigation is along the lines of the lift induced drag prediction. Several techniques can be utilized to predict induced drag, WASPE uses the force resolution method as elucidated in the work of Karamcheti [37], but can also utilize the LIDRAG [55] program which is based on the Trefftz plane approach.
3.8.3.1 Force Resolution Method (FRM)

The variation of the wing induced drag coefficient with grid density as produced by WASPE is shown in Figure 3.20. The results show considerable scatter, a steep decrease in the $C_{D,1}$ value is observed on increasing the number of spanwise panels while keeping the number of chord divisions constant. However, increasing chordwise panels while keeping spanwise panels at 30 results in steep increments in the drag produced.

![Figure 3.20 Variation of Wing Induced Drag Coefficient with number of panels. Piecewise linear variations can be associated with a fixed number of span panels indicated in the numbered boxes](image1)

![Figure 3.21 Variation of Induced Drag Coefficient with spanwise and Chordwise paneling](image2)
Modifying both the division parameters simultaneously reveals the range of $C_{D,i}$ values that lie between 0.0011 for the 90 X 10 distribution and 0.0176 for the 30 X 30 case, a 93% difference with respect to the latter value. The drag predictions from the force resolution method are overly sensitive to grid density to be useful for design studies.

### 3.8.3.2 Force Resolution Method Improved Results

Granting that the results obtained from Force Resolution Method in WASPE presented in Section 3.5.2 are from being in the realm of satisfactory, a careful inspection of the data presented in Figure 3.20 reveals a pattern that can potentially exploited to produce desirable results with an adequate level of confidence.

Figure 3.22 indicates that along a specific arrangement the induced drag coefficients indeed produce a desired behavior. The data points that form this configuration are highlighted in the figure below.

![Figure 3.22 WASPE induced drag coefficients, highlighted points possess desired grid variation characteristics](image)
A close examination of the number of spanwise and chordwise panels of these points reveal that they are in a specific ratio of 3:1 respectively. This observation spurred a desire to pursue a deeper exploration of the effects, if any, of varying the paneling ratio of the grid.

Several ratios were investigated to ascertain the best possible candidate to utilize for satisfactory results. The convergence of the wing coefficient of lift and coefficient of induced drag in the presence of a single propeller are presented in Figure 3.23. The induced drag coefficient varies from 0.0046 for 300 panels to 0.0053 for 2700 panels, with the values converging as panel number was increased, the difference between the results obtained for 2700 panels and its adjacent case, with 2250 panels was about 0.18% and thus considered acceptable.

![Figure 3.23 The convergence profile of wing CL and CDi, a total of 11 data points are utilized with a constant paneling ratio of 3:1 span to chord divisions](image1)

![Figure 3.24 Comparison of WASPE normal, LIDRAG and WASPE 3:1 results, as can be seen, WASPE 3:1 and LIDRAG results are in close agreement to each other](image2)

A total of 40 cases were considered with panel ratios varying between 20:10 to 40:10, for a propeller on and propeller off case and two angles of attack. Figure 3.24 compares WASPE force resolution method fixed ratio results with those produced by LIDRAG and WASPE FRM baseline. As evident from the plot, on increasing the number of panels both LIDRAG and WASPE FRM 3:1 seem to converge.
The WASPE panel ratio that provides best results would be the 3:1 ratio. The tabular comparison of the results obtained is shown in Table 3.4.

Table 3.4 Tabular insight into the behavior of LIDRAG and WASPE 3:1, 4 data points are shown

<table>
<thead>
<tr>
<th>Paneling</th>
<th>No. of Panels</th>
<th>C_L</th>
<th>LIDRAG C_D(i)</th>
<th>3:1 WASPE C_D(i)</th>
<th>LIDRAG ‘e’</th>
</tr>
</thead>
<tbody>
<tr>
<td>48 X 16</td>
<td>768</td>
<td>0.287</td>
<td>0.005467</td>
<td>0.004959</td>
<td>0.90029</td>
</tr>
<tr>
<td>42 X 14</td>
<td>588</td>
<td>0.287</td>
<td>0.005523</td>
<td>0.004947</td>
<td>0.89106</td>
</tr>
<tr>
<td>36 X 12</td>
<td>432</td>
<td>0.289</td>
<td>0.005613</td>
<td>0.004831</td>
<td>0.88855</td>
</tr>
<tr>
<td>30 X 10</td>
<td>300</td>
<td>0.289</td>
<td>0.005771</td>
<td>0.004609</td>
<td>0.86428</td>
</tr>
</tbody>
</table>

The table reveals that even though the estimation are slightly different, they seem to be converging in the similar direction. The variation range of LIDRAG is about 5.5% and for WASPE 3:1 the deviation is around 7%.

The consideration of the force and moment coefficients suggest that an optimum tradeoff between accuracy and speed would require around 1800 panels, with around a 3:1 span to chord panel ratio. Thereby a grid of 75 X 25 panels is recommended.

3.8.3.3 Trefftz Plane Results

The Trefftz plane results are obtained by using the code LIDRAG developed by Ives [55] based of the method introduced by Glauert [47] coupled with WASPE.

LIDRAG is based on a method extended from the Lifting Line Theory where a single dimensional discretization, i.e., spanwise, takes place in comparison to a span and chord dependent grid in VLM.
LIDRAG is an induced drag computation code that is capable of handling single planar lifting surfaces. The program takes an input in the form of spanload distribution, the coefficients of the Fourier series – which Glauert relates to the circulation, are projected through a Fast Fourier Transform.

The resulting outputs includes the integrated coefficient of lift and also the Oswald efficiency factor ‘e’, in the Trefftz plane which can then be used to compute the lift induced drag by the formula

$$C_{D,i} = \frac{C_L^2}{\pi e AR}$$

Eq. 3-46

The drag predictions using LIDRAG are depicted in Figure 3.25 along with those of WASPE from the force resolution method.

![Figure 3.25 The comparison of LIDRAG and WASPE results for wing coefficient of induced drag, the variation in the results of LIDRAG is minimal in contrast to WASPE](image)

Further details of the comparison are presented in Table 3.5
Table 3.5 Tabular Comparison of LIDRAG and WASPE results, 4 data points are considered.

<table>
<thead>
<tr>
<th>Chordwise Panels</th>
<th>Spanwise Panels</th>
<th>No. of Panels</th>
<th>$C_L$</th>
<th>Trefftz $C_D(i)$</th>
<th>FRM $C_D(i)$</th>
<th>Trefftz ‘e’</th>
</tr>
</thead>
<tbody>
<tr>
<td>25</td>
<td>45</td>
<td>1125</td>
<td>0.287</td>
<td>0.005568</td>
<td>0.009232</td>
<td>0.88348</td>
</tr>
<tr>
<td>15</td>
<td>45</td>
<td>675</td>
<td>0.287</td>
<td>0.005568</td>
<td>0.004975</td>
<td>0.88348</td>
</tr>
<tr>
<td>25</td>
<td>30</td>
<td>750</td>
<td>0.289</td>
<td>0.005771</td>
<td>0.014261</td>
<td>0.86428</td>
</tr>
<tr>
<td>15</td>
<td>30</td>
<td>450</td>
<td>0.289</td>
<td>0.005771</td>
<td>0.007772</td>
<td>0.86428</td>
</tr>
</tbody>
</table>

Another limitation associated with the use of LIDRAG for the current application is the lack of sectional induced drag distribution estimation. Also the program is limited to a maximum of 50 spanwise stations, which could limit the refinement that can be used when a multi propeller–wing configuration is to be analyzed, where 50 stations could be too little to properly account for propeller effects as discussed in Section 3.1.

The finding in this section ascertains that either of the two options can be utilized to sufficiently estimate induced drag. The benefit of using WASPE with a fixed ratio over LIDRAG is the ability to account for chordwise paneling associated with VLM over only spanwise discretization as is the case with LLT. There is no limitation on the maximum grid refinement in WASPE in comparison to LIDRAG. Also, WASPE can predict the spanwise drag distribution, which is not the case with LIDRAG, which only provides a span efficiency factor. LIDRAG could also be limited to general cases where wing propeller interaction anomalies are not evident in the induced drag, such as a wing at its zero lift angle of attack under the influence of propeller(s) slipstream, as shown in Figure 3.31.

While both WASPE FRM and LIDRAG have their benefits, the choice of the solution method for induced drag is left to the discretion of the user.
3.9 Validation with Available Data

WASPE has been validated by comparing its predictions to experimental data. Lack of experimental data for multiple propeller wing aerodynamics restricted the validation to a single wing propeller case.

The experimental data presented by Veldhuis [19] is compared to WASPE results. The Figure 3.26 shows the geometry utilized for this comparison. Further details of the geometry can be seen in the work of Veldhuis.

![Geometry diagram](image)

Figure 3.26 The geometry utilized for the comparison of WASPE results with that of literature

A single propeller – wing configuration is considered; the direction of propeller rotation is inboard up. The sectional lift coefficient distribution is depicted in the Figure 3.27. The computed and measured data compare well.
3.9. Validation with Available Data

Validation with Available Data

Figure 3.27 Comparison WASPE sectional coefficient of lift to experimental data (experimental data from Veldhuis [19])

The behavior of the distribution is also in accordance with the theory of propeller–wing interaction, which suggests that the upward rotating blade causes an increase in lift while the downward moving blade tends to reduce this.

Figure 3.28 Comparison of WASPE and Experimental wing lift coefficient (experimental data from Veldhuis [19])
The variation of the wing lift coefficient with angle of attack is also compared with measured data, Figure 3.28 illustrates the same.

A very good agreement between the experimental and WASPE results is observed.

For wing induced drag coefficient there is no known method to separate induced drag from total drag for experimental studies, therefore there is no data to compare the distribution. As such the focus is on understanding the behavior portrayed by the distribution is in accordance to the expected behavior.

Figure 3.30 depicts the distribution of the sectional coefficient of induced drag for the inboard up rotating propeller-wing configuration.

Although, induced drag distribution cannot be compared the wing induced drag coefficient can be estimated as the difference of total drag and zero lift drag. Results from Veldhuis are used for comparison. The measured and computed results are in good agreement.
The distributions show that the induced drag reduces in the area where the blade is moving up and the converse happens when the blade is moving downward, following the explanation in Section 3.1. However, in case of zero angle of attack for a wing with symmetrical wing sections, an anomalous behavior can be seen in Figure 3.31, where the induced drag in all the sections is reduced and negative. This is because non zero lift is produced only due to the presence of the propeller slipstream. The upward rotating blade causes a rise in lift and also tilts the lift vector forward causing a negative drag. Generically the downward rotating blade typically tilts the lift vector backward, however, as the lift itself is negative, the action of the propeller is to tilt this negative vector backward, causing a negative contribution in the direction of drag or a positive influence in the opposite direction.

Coefficient of pitching moment for propeller – wing configurations is also difficult to validate with available data. However, several experimental investigations for wing sections exist, and the work of Abott [56] contains data of several such sections. Also, the behavior of the coefficient of pitching moment against angle of attack can be analyzed to determine conformance to expectation.
Figure 3.32 The coefficient of pitching moment about different points varying with angle of attack

Figure 3.32 shows the variation of wing coefficient of pitching moment about the quarter chord, leading edge and center of gravity, which is at 37.5% of the mean aerodynamic chord lengths from the leading edge.

As expected, the pitching moment coefficient about quarter chord, assumed to be the aerodynamic center, is invariant with the change in angle of attack. Further the curves representing the coefficient of pitching moment about leading edge and center of gravity have negative and positive slopes as is expected of a simple wing.

Experimental comparison of these coefficients for NACA 0012 and NACA 4412 is presented in Figure 3.33, where experimental data is obtained from Abott [56].

The finite wing results from WASPE are converted to the infinite section results using aspect ratio corrections.
The comparison for both circumstances was done at varying angles of attack in the regions where flow would remain attached, as VLM would not be able to account for the separation physics. A very high degree of agreement can be seen between the computed and experimental results for both the cases.

The positive results obtained from this section indeed indicate that WASPE can be effectively and with a good degree of confidence be used for propeller–wing aerodynamic interaction modelling. Further work would involve the identification of the design sensitivities WASPE can predict.

### 3.10 Propeller Location Variation

Benefits of tip mounted propeller systems have been investigated vigorously in the past [30], [29]. The significant gains in this variation of propeller location have also been discussed by Patterson [13] and Veldhuis [19]. Therefore, an ability to model this change in characteristics would be indispensable. This section probes the ability of WASPE to ascertain such variation.
Table 3.6 Spanwise characterization geometry and analysis conditions

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Wing Chord (C)</td>
<td>2 m</td>
</tr>
<tr>
<td>Wing Semi-Span (b)</td>
<td>10 m</td>
</tr>
<tr>
<td>Propeller Diameter (Φ₁)</td>
<td>3 m</td>
</tr>
<tr>
<td>Propeller RPM</td>
<td>750</td>
</tr>
<tr>
<td>Propeller Vertical Location (Z - Axis)</td>
<td>0 m</td>
</tr>
<tr>
<td>Propeller Streamwise Location (X - Axis)</td>
<td>0.5 m</td>
</tr>
<tr>
<td>Propeller Spanwise Location (Y -Axis)</td>
<td>1.65 m to 10 m</td>
</tr>
<tr>
<td>Paneling (S X C)</td>
<td>60 X 10</td>
</tr>
</tbody>
</table>

A single inboard up propeller – wing configuration is considered for this study, spanwise variation of the propeller location with respect to the wing and the effect this has on the aerodynamics are of prime interest. A total of 25 locations are considered, further details of the analysis are presented in Table 3.6.

Figure 3.34 Spanwise lift coefficient distribution, Y=1.65 m

Figure 3.35 Spanwise lift coefficient distribution for a wingtip mounted propeller, Y=10 m

The effect of the propeller location when at the wing tip is depicted in Figure 3.35, also the effect of the propeller location when at the most inboard point physically possible with a 10% clearance, is shown in Figure 3.34.
The variation of $C_L$ with spanwise location are shown in Figure 3.36. The coefficient of lift increases as the propeller moves outboard, with a greater gradient of increase as the propeller traverses the end of the wing, towards the wing tip. This can be explained by looking at the lift distribution curves, the outboard downward rotating portion of the propeller responsible for angle of attack reduction, where reduced lift is evident, is gradually moved until the penalizing section is outside the wing, when the propeller is located at the wingtip. A similar trend has been reported by Veldhuis [19].

A kink is visible right when the propeller is located on the wing tip, on nuanced inspection of the underlying variables suggested that when the propeller center is located slightly inboard of the wingtip, this could be due to the data parsing and averaging techniques utilized. Further studies with a finer grid are required to better understand this inconsistency.

WASPE in its current form is unable to account for vertical variations in the propeller locations, while undoubtedly a promising feature. Further development is necessary to account for this aspect of the propeller – wing interaction.
3.11 Multi – Rotor Aerodynamics

WASPE can account for the presence of multiple unique and non-intersecting propellers in its analysis, however for the purpose of a lucid demonstration a case where the wing is buffeted by the slipstream of two tractor propellers is considered. Figure 3.37 depicts the geometry used for the multi-propeller study.

Lack of sufficient and available experimental data for multiple propeller – wing configurations curtail the ability to validate such configurations. Nonetheless, the characterization and validation of WASPE from the previous section was deemed sufficient to proceed with utilization of WASPE in HEDiP aircraft design studies.

![Figure 3.37 The geometry of the two propeller - wing configuration](image)

Naturally, a higher number of propellers necessitate a finer grid to properly account for the nuances of the propeller slipstream. The two propeller – wing sectional coefficient of lift is depicted in Figure 3.38.

The regions directly behind the propeller are indicated by the dotted lines, the magnitude of the difference between propeller off and propeller on cases is indicated by the solid line.
Even though WASPE accounts for only a one-way interaction, some effects of the propellers on each other seem to be evident. This leak can be observed on the region directly behind the inboard propeller where the sectional coefficient of lift of the segment of the wing, directly behind the downward rotating portion of the propeller, is significantly raised in comparison to the outboard propeller. The significance of the augmentation of lift is the desirable characteristic that is targeted to be leveraged, this augmentation is directly dependent on the propeller slipstream behavior.

The multi propeller – wing capability is extensively used in the Pacelab APD – WASPE coupling for propeller driven aircraft configurations.

### 3.12 WASPE Asymmetric

The significance of distributed propulsion on the stability and control of the aircraft led to the development of WASPE Asymmetric.

Figure 3.39 illustrates the geometry used for the WASPE Asymmetric study.
An iteration of WASPE capable of determining the aerodynamic characteristics of asymmetric wing – propeller configurations, WASPE Asymmetric can account for uneven distribution of propellers as well as aerodynamic effects of differential propeller thrust.

A demonstration of this capability is depicted in Figure 3.40. The full wing is modelled with two propellers located symmetrically on each half of the wing. The asymmetry is introduced in the form of change in rotational direction. Instead of both propellers having an inboard up direction of rotation, one propeller has an inboard up orientation while the other is associate with an inboard down direction.

The result of this asymmetry is clearly visible in Figure 3.40. This also reinforces the understanding that an inboard up propeller creates a higher lift augmentation than the outboard up case, as is evident in the asymmetric case.

Figure 3.39 Geometry for WASPE asymmetric study
3.13 WASPE Current Capabilities and Improvements

This demonstrates the capability to utilize WASPE Asymmetric for several operating conditions with DiP. Stability and control augmentation by controlling the power output of individual propulsion system could be investigated. The coupling between roll, pitch and yaw needs to be thoroughly investigated for such benefits to be utilized to reduce the size of, or maybe even eliminate, the control surfaces. The potential associated with having distributed propulsion systems on contemporary aircraft is certainly enormous.

3.13 WASPE Current Capabilities and Improvements

The capabilities of WASPE are listed below

- Estimation of the effects of multiple propellers on wing aerodynamics
  - Symmetric and asymmetric configurations and/or flight conditions
  - Wing lift, drag, pitching moment and rolling moment coefficients and distribution across the span
Effect of spanwise locations of propellers on wing aerodynamic data

- Compressibility Corrections
  - Prandtl–Glauert
  - Laitone
  - Karman–Tsien
- Kutta–Joukowski and Trefftz Plane estimates for Drag

The current limitations of WASPE are

- No shocks or viscous effects
- Current version cannot model effect of vertical and streamwise propeller locations
- Mutual interference of wing and propellers is not modeled

Thus WASPE is indeed adequate for HEDiP Aircraft conceptual design studies
CHAPTER 4

AERODYNAMIC WING DESIGN OPTIMIZATION WITH DISTRIBUTED PROPULSION SYSTEMS

One of the objectives of this research is to document the development of an optimization module capable of accounting for propeller location variation, which can be quickly modified to account for the other design variables in the wake of WASPE improvements. Aerodynamic based optimization of the configuration could help in further coaxing out a better aero performance from any DiP aircraft utilizing propellers.

4.1 Wing – Propeller Configuration Optimization (WiPCO)

The access to a low fidelity code capable of accounting for propeller – wing interaction effects allows for several design optimization studies.

The works of Miranda and Brennan [29], Kroo [18] and Veldhuis [19] clearly indicate the sensitivity of aerodynamic performance to the propeller location.

Armed with these considerations, an optimization of the placement of propellers based on location, number and direction of rotation could be leveraged to give a more efficient design. This lead to the contemplation of the generation of the Wing Propeller Configuration Optimization (WiPCO) module.
Application of WiPCO is limited to spanwise location because the streamwise and vertical have potential to be used in WiPCO, however owing to the current inability of WASPE to model vertical variation of the propeller, they are not currently included. On further improvements in WASPE this design variable set can be readily utilized for a wider design space exploration.

The following section describe the optimization formulation in more detail.

4.2 Optimization Methodology

The optimization approach considers two optimization methods, the Sequential Quadratic Programming (SQP) method and the Genetic Algorithm (GA) method. Both of these methods have been leveraged in WiPCO, an in-depth explanation to these approaches can be obtained in the work of Arora [57].

The Sequential Quadratic Programming method is one of the solutions to the constrained Nonlinear Programming Problem (NLP). This iterative approach uses semi-second order information of the problem functions to obtain an improved rate of convergence as compared to a linear approximation.

The Genetic Algorithm is an approach inspired by the natural selection process, this involves the generation of an initial population of candidate solutions to the optimization problem. This population is allowed to mutate and crossover to progress towards a better solution every generation. This evolution continues until the convergence criteria are satisfied.

4.3 Optimization Formulation

The primary aim of the optimization module was the identification of the optimum configuration to maximize aerodynamic efficiency. A reduction in wing area is estimated based on the improvement in the aerodynamic characteristics of the optimized configuration in comparison to the baseline. Two optimization approaches are considered and their merits and limitations are discussed in this section. The purpose of WiPCO is to be used supplementary to the MDO studies, where the MDO optimized design could be further fine-tuned with inputs from WiPCO. Decisions on the number of electric motors to be used along with their optimum placement can be enhanced based on the obtained optimization results.
4.3.1 Outcome of Optimization and Objective Function

The primary outcome of the envisioned optimization problem through WiPCO is the Area Reduction Potential (ARP). The improvement in the aerodynamic performance would mean that a smaller lifting surface would suffice to carry out the same design mission.

\[
ARP = \frac{S_{\text{baseline}} - S_{\text{optimized}}}{S_{\text{baseline}}} = 1 - \frac{\text{Lift}_{\text{optimized}}}{\text{Lift}_{\text{baseline}}}
\]  

Eq. 4-1

The objective driving the optimization is the aerodynamic efficiency represented by the lift – to – induced drag ratio, this parameter is targeted to be maximized.

\[
F_1(X) = \frac{C_{l,\text{Wing}}}{C_{Dl,\text{Wing}}}
\]  

Eq. 4-2

The Sequential Quadratic Programming (SQP) and the Genetic Algorithm (GA) approaches are utilized for optimization in WiPCO. The results of both the methods are presented in Section 4.3.4.

4.3.2 Design Variables

The design variables considered in WiPCO are given as

1. Number of Propellers (N)
2. Rotational Orientation of Individual Propellers (D_N)
3. Spanwise Location of Individual Propellers (Y_N)

The total number of design variables would depend on the number of propellers, and is determined as 2N+1. The design variables for a sample case are depicted by Figure 4.1.
No. of Design Variables = 2N + 1

Figure 4.1 Design Variables illustrated for a 2 propeller case where N = 2.
Inputs beginning with Y have numeric inputs and those beginning with D have integer inputs

It’s interesting to note that the number of active design variables can vary between iterations. The first design variable depicting number of propellers in an iteration dictates the active or passive status of the associated design variables. Taking advantage of the symmetric nature of the problem, only one half of the wing is considered. The optimization constraints for this optimization problem are described in Section 4.3.3.

4.3.3 Optimization Constraints

The bounds and constraints are put in place to ensure that nonphysical solutions are avoided. The basic premise of the constraints is to ensure that the placement of the propellers is physical feasible. This set of constraints basically impose a criterion of non-intersection as described later in this section. Also, the optimization module ensures that the lift produced or the coefficient of lift of the combined system is either greater or the same as the baseline configuration.
A bound that is imposed on the problem is that the propellers may not cover more than nine-tenths of the half wing span. The formulation of the constraints is given below.

\[
g_1(X) = \Delta \text{Lift} \leq 0 \tag{4-3}
\]

\[
g_2(X) = \sum_{i=1}^{N} 2 * r_i - 0.9 * \left(\frac{b}{2}\right) \leq 0 \tag{4-4}
\]

The generalized non-intersection constraints are given as

\[
g_3(X) = r_1 - Y_1 < 0 \tag{4-5}
\]

\[
g_4(X) = Y_1 - \left(\frac{b}{2}\right) + \sum_{i=2}^{N-1} 2 * r_i + r_N + r_1 \leq 0 \tag{4-6}
\]

\[
g_k(X) = Y_{(k-1)} + r_{(k-1)} + r_k - Y_k \leq 0, \text{where} \ 2 \leq k \leq N \tag{4-7}
\]

\[
g_l(X) = Y_l - \left(\frac{b}{2}\right) + \sum_{i=(l+1)}^{N-1} 2 * r_i + r_N + r_l \leq 0, \text{where} \ 2 \leq l \leq N - 1 \tag{4-8}
\]

\[
g_{(2N+2)}(X) = Y_N - \left(\frac{b}{2}\right) \leq 0 \tag{4-9}
\]
At any given iteration of the problem, the total number of constraints to be satisfied are $2N+2$. It is important to note that the number of active constraints is not dependent on maximum number of propellers, but the number of propellers for that particular iteration.

The following set of Figures illustrates the non-intersection constraints for an example involving 3 propellers where all the non-intersection conditions are satisfied.

The first step is to set the constraints for the placement of the first propeller, in Figure 4.2.

![Figure 4.2 Step 1: Range of feasible spanwise locations for placement of the 1st propeller](image)

This constraint ensures that the placement is physically possible and also ensures that in the extreme case, where the first propeller is placed in its outward most location, there is sufficient room for the placement of the remaining 2 propellers.

Step 2 shown by Figure 4.3, is for the placement of the second propeller, note that the first propeller is already placed.

![Figure 4.3 Step 2: Range of feasible values for placement of the 2nd propeller, considering that the 1st propeller location is known](image)
The range of physically feasible propeller placement locations for propeller 2 would depend on the location of the first propeller, and the sum of the radii of the first and the second propellers. The satisfaction of the non-intersection constraints for the propeller 2 is associated with the above depicted range.

Step 3 identifies the feasible location for placement of the third propeller, given that two propellers are already in place, as shown in Figure 4.4.

![Figure 4.4 Step 3: Range of feasible values for placement of the 3rd propeller, considering that the 1st and 2nd propeller locations are known](image)

Its formulation is similar to that of Step 2. It provides for the satisfaction of the non-intersection constraints for the third and final propeller. However, the only difference is in the constraint for the maximum allowed outboard location. The outward most feasible location is now equivalent to the half span allowing for the case incorporating tip–mounted propellers.

The final Step 3 A of the example is illustrated in Figure 4.5, and this shows the successful feasible placement of all three propellers.

![Figure 4.5 Step 3A: Illustration of a feasible propeller placement for a 3 propeller case, satisfying the non–intersection constraint](image)
The future consideration of the vertical and streamwise propeller location design variables should not impact the enabling of these non-intersection constraints.

4.3.4 Optimization Results

WiPCO was run on several different settings to understand the behavior of the defined problem. Initial runs included setting the maximum allowable propellers to 1, 2 and 3, while allowing other design variables to proceed unhindered. Accounting for the difficulty in addressing discrete design variables in sequential quadratic programming, they were manually varied. The Genetic Algorithm based solver was allowed to proceed unhindered.

4.3.4.1 Sequential Quadratic Programming

The results for the SQP are shown in Figure 4.6 for the 1, 2 and 3 propeller cases, the converged optimum is presented in the following Table 4.1.

---

Figure 4.6 SQP Results for 0-3 Propellers
### 4.3. Optimization Formulation

#### Table 4.1 Tabulated Results for SQP optimization for 0-3 Propellers

<table>
<thead>
<tr>
<th>No. Of Propellers</th>
<th>Location 1 (in m)</th>
<th>Location 2 (in m)</th>
<th>Location 3 (in m)</th>
<th>( \frac{C_L}{C_D(s)} )</th>
</tr>
</thead>
<tbody>
<tr>
<td>0</td>
<td>---</td>
<td>---</td>
<td>---</td>
<td>82.64</td>
</tr>
<tr>
<td>1</td>
<td>0.628372</td>
<td>---</td>
<td>---</td>
<td>95.32</td>
</tr>
<tr>
<td>2</td>
<td>0.632011</td>
<td>0.394648</td>
<td>---</td>
<td>101.56</td>
</tr>
<tr>
<td>3</td>
<td>0.630198</td>
<td>0.335765</td>
<td>0.108</td>
<td>111.73</td>
</tr>
</tbody>
</table>

While SQP is faster in execution than GA, the primary issues are the handling of discrete variables and the reliance of the final solution on the initial guess and the finite-difference step size.

The results obtained indicate that the aerodynamic efficiency increases with the number of propellers. However, that would be because the size and characteristics of the propellers are being kept constant, further increase in number of propellers would require the reduction in size.

#### 4.3.4.2 Genetic Algorithm

Genetic algorithms are able to handle discrete variables, and requires no restriction or manual prescription of integer constrained design variables. The same case as above is run in WiPCO using GA, the results are depicted in Figure 4.7.
The converged solution is tabulated in the Table 4.2.

<table>
<thead>
<tr>
<th>No. Of Propellers</th>
<th>Location 1 (in m)</th>
<th>Location 2 (in m)</th>
<th>Location 3 (in m)</th>
<th>$C_L/C_{D(i)}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>3</td>
<td>0.64</td>
<td>0.3721</td>
<td>0.1191</td>
<td>115.15</td>
</tr>
</tbody>
</table>

A population size of 50 was used. The benefits of using the Genetic Algorithm is associated with its lack of dependency on gradients and its ability to account for integer variables. As such, the Genetic Algorithm approach was preferred and can be utilized for the configuration optimization involving any number of propellers.
4.4 Potential Role of WiPCO in HEDiP Aircraft Design

The Wing Propeller Configuration Optimization (WiPCO) module provides interesting insights into the optimal locations for propeller placements. The results from WiPCO can be utilized in the design of a more efficient HEDiP aircraft design, where, instead of equally or randomly spaced propeller locations an aerodynamically optimized configuration would be used. The potential WiPCO – APD interaction is depicted in Figure 4.9.

The initial HEDiP designed on Pacelab APD based on user inputs and requirements, would share wing-propeller configuration information, on user discretion, once with WiPCO, which would again provide Pacelab APD with the optimal configuration information for the initial HEDiP design. Modification of the HEDiP aircraft design to account for WiPCO suggestions would result in the aerodynamic based configuration optimized HEDiP aircraft.
CHAPTER 5

HYBRID ELECTRIC REGIONAL TRANSPORT AIRCRAFT CONCEPTUAL DESIGN STUDY

One of the objectives of this research is to identify the benefits associated with the use of HEDiP technologies in design of aircraft. In this chapter the design of one such aircraft is presented to understand the sensitivity of the design to battery storage and energy source distribution parameters. Also, benefits of the wing-propeller interaction arising out of Distributed Propulsion is investigated. A total of 3 Hybrid Electric Regional Multi-propeller integrated Transport (HERMiT) aircraft are conceptually designed.

This chapter includes the integration of WASPE and Pacelab APD along with the demonstration of the conceptual design capability for a Hybrid Electric aircraft with Distributed Propulsion systems for a typical regional transport mission. The chapter also highlights the key differences between a HEDiP design and a contemporary comparator aircraft.
5.1 WASPE – Pacelab APD Integration

The flow of data in the integration of WASPE with Pacelab APD is shown in Figure 5.1, the procedure allows the use of aerodynamic data from one of two sources, a dynamic data set from WASPE varied with each iteration and a Reference Drag Polar generated with WASPE for the initial estimate, the choice of data source is designated as a user input.

Figure 5.1 Flowchart depicting WASPE-APD integration
The tradeoff between the two ways is accuracy and speed, with each drag polar containing about 50 data points as altitude, speed and angle of attack are varied. The use of a single reference drag polar generated with an initial estimate of the wing geometry would be several times faster, depending on the number of iterations, than a dynamic query for the updated drag polar from WASPE.

The methodology for this study is to obtain a converged solution using the reference drag polar where the geometry is allowed to vary. This is followed by further refinement of the performance characteristics using updated data from WASPE with the converged solution as the analysis case, thereby striking balance between speed and accuracy. This process could be repeated multiple times if required for a higher degree of accuracy.

The data interaction between WASPE and Pacelab APD occurs through a File – In – File – Out (FIFO) mechanism described in the following Section 5.1.1.

5.1.1 WASPE – Pacelab APD Coupling using FIFO Communication

A few features of WASPE that make the integration with Pacelab APD more convenient include the ability of WASPE to undertake batch analysis along with the consideration of altitude and velocity effects.

The FIFO interaction requires the generation of 4 files in Pacelab APD to act as inputs for WASPE and the processing of the drag polar output file generated by WASPE after the analysis.

The input files are listed as

1. Aircraft Geometry Input File
   The first input file includes information regarding the wing geometry along with size, number, location and direction of rotation of any existing propellers. Also included is the information regarding wing paneling, with the number of spanwise and chordwise panels specified.

   A sample input file of a generic aircraft is shown in Appendix A.

2. Analysis Conditions Input File
This input file contains information on the conditions for the required analysis. The freestream velocity, altitude, density and angle of attack of the wing are included as comma separated values and are structured in 3 loops.

- Speed
  - Altitude and Density
    - Angle of Attack

A sample analysis conditions input file is included in Appendix A.

3. Airfoil Input File

The wing airfoil characteristics are provided as X, Y – Coordinate data for unit chord length, the airfoil file is in the Selig Format [58], with the airfoil coordinates originating from the trailing edge which trace the upper surface followed by the lower surface and terminate once again at the trailing edge.

The airfoil coordinate file for NACA 0012 in the specified format is shown in Appendix A.

4. Slipstream Input File

The fourth input file for WASPE contains the propeller input slipstream profile, with the radial location normalized by the propeller radius and the axial and tangential velocities normalized by the freestream. This data can either be from XROTOR, experiments or any other source.

While the general version of WASPE can account for multiple slipstream profiles representing different propellers, the integrated version for simplicity uses one slipstream profile for all the propellers which is scaled by the propeller radius.

The output file contains the coefficients of force and pitching moment appended to the analysis conditions to ensure proper indexing of aerodynamic data in the drag polar. Appendix A, Contains an illustration of the output file.

5.2 Baseline Aircraft: de Havilland DHC – 6 – 400 Twin Otter Aircraft

The Twin Otter aircraft is used as the baseline aircraft for hybridization, the widely used aircraft with more than 900 built, is an excellent case for the HEDiP assessment study.
The Twin Otter, originally developed by de Havilland Canada and now being produced by Viking Air, is a twin turboprop conventional regional transport aircraft powered by Pratt & Whitney single stage free-turbine propulsion systems coupled with three bladed reversible pitch Hartzell propellers. The propulsion systems are mounted on the high wing. The Twin Otter consists of a metal airframe and a tricycle landing gear.

### 5.2.1 Design Specifications

The Twin Otter design specifications are included in the Table 5.1 [59].
Table 5.1 Baseline Twin Otter Specifications

<table>
<thead>
<tr>
<th>Specification</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Mission Payload</td>
<td>19 + 2 PAX (1,474 kg)</td>
</tr>
<tr>
<td>MTOW</td>
<td>5,670 kg</td>
</tr>
<tr>
<td>OEW</td>
<td>3,377 kg</td>
</tr>
<tr>
<td>MLW</td>
<td>5,579 kg</td>
</tr>
<tr>
<td>Span</td>
<td>19.8 m</td>
</tr>
<tr>
<td>Area</td>
<td>41.98 m²</td>
</tr>
<tr>
<td>Wing Loading</td>
<td>135 kg/m²</td>
</tr>
<tr>
<td>Wing Airfoil Section</td>
<td>NACA 63A516 Mod</td>
</tr>
<tr>
<td>Thrust-to-Weight Ratio</td>
<td>0.54</td>
</tr>
<tr>
<td>Max Range with Mission Payload</td>
<td>400 NM</td>
</tr>
<tr>
<td>Ceiling</td>
<td>8138.16 m (26,700 ft.)</td>
</tr>
<tr>
<td>Cruise Altitude</td>
<td>3048 m (10,000 ft.)</td>
</tr>
<tr>
<td>Max. Cruise Speed</td>
<td>182 KT</td>
</tr>
<tr>
<td>Propulsion</td>
<td>2x PT6A-34, 650 SHP</td>
</tr>
<tr>
<td>Max. Fuel</td>
<td>1431 L</td>
</tr>
</tbody>
</table>

5.3 Aircraft Mission

The aircraft design mission for the conceptual design of the HERMiT series is the same as that of the de Havilland Canada DHC – 6 – 400 Twin Otter. The Twin Otter has a maximum range of 400 NM with a 19
passenger load. Demonstration of the capabilities of the HEDiP RTA for the same mission would help quantify the potential benefits.

Figure 5.3 shows the primary mission profile and Table 5.2 contains some of the key parameters for specifying the mission.

![Figure 5.3 HERMiT RTA design mission. Range considered from start to end](image)

<table>
<thead>
<tr>
<th>Table 5.2 Key Mission Parameters for HERMiT</th>
</tr>
</thead>
<tbody>
<tr>
<td>Mission Range</td>
</tr>
<tr>
<td>HERMiT Payload</td>
</tr>
<tr>
<td>HERMiT Cruise Altitude</td>
</tr>
<tr>
<td>Cruise Speed</td>
</tr>
<tr>
<td>Max Payload (For Payload - Range)</td>
</tr>
<tr>
<td>Reserve Mission</td>
</tr>
</tbody>
</table>

The same mission was recreated on Pacelab APD 6.2 in order to validate the performance prediction by the MDAO framework, the twin otter model performance and parameters were in close agreement with the original aircraft specifications.

Also, HERMiT would be designed to accommodate a fuel reserve to enable a 45-minute hold at cruise level, for a fair comparison with the Twin Otter.
5.3.1 Modeling Assumptions and Parameters in Pacelab APD

The sizing of Twin Otter along with HERMiT – 2E and HERMiT – 6E on APD was based on directing certain parameters as fixed input assumptions and certain others as design inputs Table 5.3 highlights some of the key assumptions, inputs and output parameters estimated by Pacelab APD.

<table>
<thead>
<tr>
<th>Assumptions</th>
<th>Design Inputs</th>
<th>Outputs</th>
</tr>
</thead>
<tbody>
<tr>
<td>Thrust-to-Weight ratio</td>
<td>Range</td>
<td>MTOW</td>
</tr>
<tr>
<td>Wing Span</td>
<td>Payload</td>
<td>OEW</td>
</tr>
<tr>
<td>Wing Area</td>
<td>Battery Specific Density</td>
<td>MFW</td>
</tr>
<tr>
<td>Propulsion Power-to-Weight</td>
<td>Hybridization Factor</td>
<td>Wing Loading</td>
</tr>
<tr>
<td></td>
<td>Cruise Characteristics</td>
<td>Battery Energy</td>
</tr>
<tr>
<td></td>
<td></td>
<td>Battery Weight</td>
</tr>
</tbody>
</table>

The Twin Otter as modeled in APD is shown in Figure 5.4.

Figure 5.4 DHC – 6 – 400 Twin Otter in Pacelab APD
The hybridization and re-design of the Twin Otter to subsequently produce the HERMiT series requires the consideration of a Hybrid Electric Architecture. Section 5.4 details the selected architecture and the associated benefits for the same.

### 5.4 Hybridization Methodology for HERMiT

The parallel, series and turboelectric architectures were considered for implementation in HERMiT. Several criteria were considered in the process of selection, some of them are

- **EIS**: Target timeline for the realistic implementation of the architecture, a very important consideration owing to the goal of introducing HERMiT in the 2030 timeframe
- **Efficiency**: The coupling of the energy sources along with associated conversion, conditioning or transfer results in loss of useable power, architectures with lower energy losses are looked upon favorably
- **Complexity**: The ease of integrating the fuel and electrical energy along with the ease of designing architecture subsystems would also have to be considered

The parallel hybrid electric architecture requires the coupling of the motor and the shaft of the engine. This requires the design of a complex gearbox subsystem to accommodate operations using simultaneous and individual energy sources. The efficiency of the powertrain as estimated in Chapter 1 is about 47% for equal electrical and fuel power output.

A series hybrid electric architecture on the other hand has associated power conversion losses stemming from the conversion of the chemical energy to electrical energy and associated conditioning, while lacking the complexity of the parallel architecture it has a total estimated efficiency of 51% for equal electrical and fuel power output.

This brings about the consideration of the Parallel Power Split (PPS) Hybrid Electric Architecture illustrated in Figure 5.5.
The key benefits of this architecture are its simplicity along with a high degree of utilization of existing technology. The PPS hybrid electric architecture also does not require the design of any complex mechanical subsystems. This architecture can be implemented using existing technology with each propulsor relying solely on one type of energy source.

The hybridization factor for this architecture at sea level is prescribed as

$$HF = \frac{Engine\ Shaft\ Power}{(Engine\ Shaft\ Power + Motor\ Shaft\ Power)}$$

Eq. 5-1

The efficiency of the architecture is a function of the number of fuel and electric propulsion systems along with the hybridization factor. The efficiency also depends on the altitude because of engine power output being dependent on the same.

The efficiency of the PPS architecture with 2 motors and 2 engines, with a hybridization factor of 0.5 is about 55%.
The general efficiency associated with this architecture is given by

\[ \eta = [(1 - HF) \times 0.39 + HF \times 0.71] \times 100\% \]  

Eq. 5-2

With a higher efficiency and simplicity, the Parallel Power Split Hybrid Electric Architecture is considered to be a good fit for HERMiT.

5.5 Hybrid Electric Regional Multi-propeller integrated Transport (HERMiT)

Twin Otter acting as the baseline, three hybrid electric variants were conceptually designed to understand the general trends that could be posed by such aircraft. Sensitivity studies of Maximum Takeoff Weight (MTOW) to hybridization factor and battery energy density are of prime interest to understand when and to what extent existing aircraft can be hybridized. The effects of varying the number of powerplants or propellers, i.e., distributing the propulsion system and also benefits of propeller wing aerodynamics are investigated.
5.5.1 HERMiT 2E

HERMiT 2E is the hybridized variant with 4 powerplants; 2 electric motors and two engines. The wing geometry was kept the same as the baseline Twin Otter. The target of the design was to match the specified range and payload of the Twin Otter. The benefits of propeller-wing interaction were not accounted for. This case assesses the practicality of developing an aircraft using hybrid electric propulsion without any other benefits for an EIS using technologies in the 2025 timeframe.

Figure 5.7 depicts the three view of HERMiT 2E

![HERMiT 2E Diagram]

Figure 5.7 Three view of HERMiT 2E, Geometry Renderings from Pacelab APD

The inboard powerplants in red depict the air breathing engines and the outboard podded powerplants are electric motors.
Sensitivity studies on HERMiT 2E were carried out on Pacelab APD, with the objective being the MTOW and the design variables being hybridization factor and battery specific energy. The variation of MTOW with battery energy density at a HF of 50% is shown in Figure 5.8. The trend reveals that in the 2025 timeframe, with a battery energy density of about 0.31 kWh/kg, HERMiT 2E has more than 2 times the MTOW of the baseline, making an EIS with 50% hybridization impractical.

![Figure 5.8 HERMiT 2E Sensitivity to battery energy density at HF = 0.5](image1)

![Figure 5.9 HERMiT 2E MTOW sensitivity to hybridization and battery storage advancements](image2)

The sensitivity of HERMiT 2E MTOW to both HF and battery energy density is depicted in Figure 5.9. Note that a design leveraging lower hybridization would be less dependent on energy density. Similarly, higher levels of hybridization show higher sensitivity to battery energy density.

While HERMiT 2E might not be practical for 2025, it still offers better performance than an all-electric variant, as evident from the trend observed in the carpet plot where more electric (higher HF) would mean a greater penalty.
5.5. Hybrid Electric Regional Multi-propeller integrated Transport (HERMiT)

The Payload – Range performance of HERMiT 2E with a 50% hybridization and 0.4 kWh/kg battery energy density is shown in Figure 5.10.

Thus it can be concluded that while HERMiT 2E satisfies all design requirements, it would not be an economically viable replacement to the fuel burning baseline Twin Otter.

5.5.2 HERMiT 6E

The next iteration of the series HERMiT 6E leverages 8 propulsion systems out of which 6 are electric motors. However, the potential benefits due to DiP aerodynamics were not accounted for. The reference wing area and design mission were also matched to those of HERMiT 2E. The conceptual design was carried out under the presumption of technologies predicted to be have matured by the mid-term timeframe of 2025.

Sensitivity studies were performed to understand the behavior of MTOW with variations in HF and battery energy density, similarity in the trends with HERMiT 2E could be observed.

Figure 5.13 shows the comparison between the MTOW sensitivity of HERMiT 2E and 6E to the advancement in battery specific storage capacity.
The MTOW sensitivity chart reveals that a HERMiT 6E design would have about a 15% higher MTOW than the 2E variant and around 3 times the MTOW of the baseline, if 2025 technologies are utilized with a 50% hybridization. The HERMiT 6E indeed sees a reduction in performance when compared to its 2E variant, this penalty is rationalized as being a result of the additional powerplant systems and subsystems without accounting for the relief provided by improved aerodynamic efficiencies due to wing-propeller interaction.

<table>
<thead>
<tr>
<th>HERMiT</th>
<th>MTOW (kg)</th>
<th>Wing Loading (kg/m²)</th>
<th>Energy Density (kWh/kg)</th>
<th>Hybridization Factor</th>
<th>Fuel (kg)</th>
<th>S_{ref} (m²)</th>
</tr>
</thead>
<tbody>
<tr>
<td>2E</td>
<td>7850</td>
<td>187</td>
<td>0.5</td>
<td>0.5</td>
<td>540</td>
<td>41.98 m²</td>
</tr>
<tr>
<td>6E</td>
<td>8700</td>
<td>207</td>
<td>0.5</td>
<td>0.5</td>
<td>630</td>
<td>41.98 m²</td>
</tr>
</tbody>
</table>
A few key features of the HERMiT 2E and 6E designs, for another battery energy density are compared in Table 5.4.

Thus for the HERMiT series, based on the above results, it can be concluded that hybridization alone of the baseline would not be sufficient to be an economically beneficial replacement. Improvements to the design while accounting for the benefits provided by the Wing-Propeller interaction is very much necessitated.

### 5.5.3 HERMiT 6E/I

For the next phase of the study, an improved version of the 6 Motor and 2 Engine Hybrid variant, HERMiT 6E/I was designed which leverages the benefits of wing-propeller interaction effects. The initial estimation uses a reference drag polar generated from WASPE for the HERMiT 6E wing-propeller configuration. Further, the result obtained is refined by using the drag polar from WASPE being run dynamically with APD, this methodology was employed to reduce the number of WASPE calculation cases and thereby saving time considerably. A typical WASPE batch run when coupled with APD requires around 75 calculation cases for each iteration, with the general calculation case requiring about 3.5 minutes of runtime on a workstation, the specifications of which are provided in Table 5.5.

Also, HERMiT 6E/I was designed using Pacelab APD 7.0 which is coupled to WASPE, while HERMiT 2E and 6E were designed on Pacelab APD 6.2.

<table>
<thead>
<tr>
<th>Table 5.5 Workstation Specifications</th>
</tr>
</thead>
<tbody>
<tr>
<td>Operating System</td>
</tr>
<tr>
<td>Processor</td>
</tr>
<tr>
<td>No. of Cores</td>
</tr>
<tr>
<td>Max. Processor Speed</td>
</tr>
<tr>
<td>Installed Memory (RAM)</td>
</tr>
</tbody>
</table>
The aircraft design mission still remained the same, to ferry 19 passengers across a distance of 400 NM. The hybridization factor for the 6E/I variant was 0.3, the wing span was fixed to the HERMiT 6E value, also the mean aerodynamic chord was varied.

Figure 5.14, illustrates the three view of HERMiT 6E/I.

![HERMiT 6E/I three view, geometry rendered on Pacelab APD 7.0](image)

The key design parameters of the HERMiT 6E/I are compared with that of the baseline in Table 5.6.
HERMiT 6E/I was designed with a target Entry-Into-Service (EIS) timeframe of 2030, utilizing technologies with TRL of 6 or higher by 2025.

The result of leveraging the benefits of wing-propeller aerodynamics with distributed propulsion along with Hybrid Electric propulsion systems is a 29% reduction in the lifting surface area when compared to the baseline Twin Otter. Also, the energy cost associated with the operation of the Aircraft per trip is also reduced significantly. The improvements in HERMiT 6E/I come at the slight penalty of MTOW for the same design mission.
5.6 Benefits of HERMiT over Twin Otter

The conceptual design of HERMiT 6E/I demonstrates the clear benefits over an existing conventional regional transport design.

Figure 5.15 shows the HERMiT 6E/I superimposed over the baseline Twin Otter and illustrates the reduction in wing area, the 6E/I lifting surface is highlighted in red.

The enhancement of the lifting surface aerodynamic efficiency translates to a significantly reduced wing area, which would mean a lower thrust to weight ratio could be achieved along with a reduction in required power generation by about 45%. A comparable performance can be obtained with a more than 50% saving in trip energy costs, a very significant improvement for a small MTOW penalty.

The fuel burn is also significantly reduced, with HERMiT 6E/I requiring only about 38% of the mission fuel borne by the comparator. This would mean a correspondingly lower direct atmospheric heat release, reduction in Carbon Dioxide and NOx emissions along with reduced energy consumptions, all of which are
some of the highlighted subsonic technology improvement goals suggested by the NASA Strategic Implementation Plan 2017 [1].

Implementation of HEDiP technologies in future RTAs indeed bring some very interesting improvements both from economic and environmental standpoints.

The demonstration of a projected feasibility and economic practicality of HERMiT 6E/I introduction to the market by 2030, suggests that HEDiP technologies can be utilized to reap several of the benefits All-Electric-Aircraft are associated with, in a not very distant timeline.
CHAPTER 6

CONCLUSIONS AND FUTURE WORK

Design optimization of a regional transport aircraft with hybrid electric distributed propulsion systems was undertaken. The primary goal of this research was to investigate the benefits associated with leveraging unconventional aircraft designs coupled with hybrid electric energy architectures.

Key conclusions are in Section 6.1 and areas of future work in Section 6.2.

6.1 Conclusions

The key conclusions arising out of this study are:

- The conceptual design methods for HEDiP Aircraft design were duly enhanced.
  - Pacelab APD MDO framework found to be most suitable fit for current research based on a qualitative comparison of two other frameworks.
- HEDiP Aerodynamic prediction capability has been improved.
  - The Wing Aerodynamic Simulation with Propeller Effects (WASPE) module was successfully developed and validated.
  - WASPE was successfully coupled with Pacelab APD for HEDiP Aircraft design studies.
- Wing – Propeller Configuration Optimization methodology and code developed.
  - WiPCO module developed to enable potential aerodynamic efficiency improvements of lifting surfaces.
  - SQP and GA optimization approaches utilized.
  - Optimum wing-propeller configuration based on Number, Spanwise Location and Rotation direction identified.
6.2. Future Work

- HERMiT Series Designed and benefits of HEDiP technologies over conventional aircraft demonstrated.
  - Sensitivity of design to battery storage technology and extent of hybridization explored.
  - The 6E/I variant has 50% reduced operating energy costs.
  - DiP technology in 6E/I allows for about 30% reduction in lifting surface area.
  - Allows for a reduced energy consumption and fuel burn, thereby limiting nefarious effects to the environment.
  - Practical and Beneficial implementation of HERMiT 6E/I allows for an EIS in 2030.

Thus, all the research objectives set forth in the beginning of this Thesis, to enable HEDiP aircraft design, have been accomplished.

The conceptual design of the HERMiT 6E/I aircraft, leveraging the benefits of Hybrid Electric Distributed Propulsion systems, indicates the possibility and economic viability of implementation of these more sustainable alternatives in a near- to mid-term timeframe.

### 6.2 Future Work

HEDiP technology implementation in aircraft design opens several avenues of investigation. A few recommendations for future work include

- The investigation of thermal management systems for implementation of HEDiP architectures.
- Characterization of the noise footprint reduction or change as a result of employing electric motors and distributed propulsion configuration.
- Exploration of the capabilities and limitations of other hybrid electric architectures, understanding the sensitivity of the aircraft design to HE architecture type along with more effective and efficient fuel and electric energy coupling techniques.
- WASPE capabilities can be further enhanced to
  - Model wing-propeller mutual interference effect, by accounting for a true two-way aerodynamic coupling
  - Model streamwise and vertical propeller location effects on the lifting surface aerodynamics
  - Simulate control surface effects
- An investigation of aircraft stability and control by coupling traditional control surfaces with differential thrust capability of distributed propulsion configurations.
• The exploration of the use of surrogate models for wing-propeller aerodynamics, which could make for an effective substitute to analysis techniques for use in MDO and conceptual design studies.
REFERENCES


Appendix A

This Appendix contains sample input files for the WASPE-APD coupling

Aircraft Geometry Input File Example

% Sample APD Aircraft Geom Disc

cg=0.16929219972166,0,0
b=19.8
rootc=2.12
tpr=1
swp=0
dih=3.5
tws=0,0
rnum=4
rrad
1,1,1,1
rloc
2.475,4.95,7.425,9.62973
rdir
1,1,1,1
spd=10
chd=5

Analysis Condition Input File Example

40,1.22500001812429,0,0
40,1.22500001812429,3,0
40,0.516720072333047,0,8138.16
40,0.516720072333047,3,8138.16
200,1.22500001812429,0,0
200,1.22500001812429,3,0
200,0.516720072333047,0,8138.16
200,0.516720072333047,3,8138.16

Airfoil Input File Example

Airfoil
   l   0
0.999013364214136   0.0116529195486363
0.996057350657239   0.0150987190360854
0.991143625364344   0.0207988786958524
0.984291580564316   0.0286903683059441
0.975528258147577   0.0386873224815751
0.964888242944126   0.0506832342220234
0.95241352623301    0.0645535277951363
0.938153340021932    0.0801583498559013
<table>
<thead>
<tr>
<th>Slipstream Input File Example</th>
</tr>
</thead>
<tbody>
<tr>
<td>0.0954915028125263 -0.383741109743022</td>
</tr>
<tr>
<td>0.114743378612105 -0.40948172131823</td>
</tr>
<tr>
<td>0.135515686289294 -0.432187721829873</td>
</tr>
<tr>
<td>0.157726447035656 -0.451727182509592</td>
</tr>
<tr>
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</tr>
<tr>
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</tr>
<tr>
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</tr>
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</tr>
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