Aeroelastic and Aeroacoustic Modelling of Rotorcraft

by

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Carleton University

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Ottawa-Carleton Institute for
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A handwritten signature in black ink, consisting of a large loop followed by a horizontal stroke and a vertical line extending downwards.

Dr. F. Nitzsche, Thesis Supervisor

A handwritten signature in black ink, appearing to read 'J. Beddoes'.

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Abstract

In the present work, SMARTROTOR, a code for aeroelastic and aeroacoustic modeling of rotor blades, is developed. SMARTROTOR is created by interfacing aerodynamic, aeroacoustic, and structural dynamic components. The aerodynamic component is GENUVP, an existing unsteady panel code with a particle-wake model. The new aeroacoustic component, which is developed within the current work as an addition to GENUVP, is based on a solution of the Ffowcs Williams-Hawkings equation for thickness and loading noise. The structural component is an existing non-linear beam element model of the rotor blades based on a mixed variational intrinsic formulation, with inclusion of the effects of integral twist actuation. The aerodynamic and structural components are interfaced to form a closely-coupled aeroelastic code that solves in the time-domain. Aerodynamic, hub vibration, aeroelastic deflection, and aeroacoustic validation results are presented for a model of the HELINOISE BO105 scale rotor. The current version of SMARTROTOR can be used for future investigation of the noise reduction capability of active twist rotor blades.

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List of Symbols

a	Global rotating reference frame, Chapter 4
$\vec{A}$	Vector velocity potential, Chapter 2
$\overline{\partial A}$	Virtual action at the ends of the space and time domain, Chapter 4
b	Undeformed beam reference frame, Chapter 4
B	Deformed beam reference frame, Chapter 4
c	Speed of sound, Chapter 3
C	Matrix of influence coefficients, Chapter 2
C	Rotation matrix, Chapter 4
C^{ab}	Transformation matrix from b to a, Chapter 4
C^{Ba}	Transformation matrix from a to B, Chapter 4
C^{Bb}	Transformation matrix from b to B, Chapter 4
C_N	Normal force coefficient, Chapter 5
C_p	Pressure coefficient, Chapter 2
C_T	Rotor thrust coefficient, Chapter 5
D	Flowfield domain, Chapter 2
$\vec{D}$	Deformation tensor, Chapter 2
$D_\omega(t)$	Vorticity flowfield domain, Chapter 2
f	Equation of body surface, Chapter 3
f_a	Aerodynamic force per unit length, Chapter 4
$f_{u_i}, f_{\psi_i}, \dots, f_{M_{i+1}}$	Beam element functions, Chapter 4

F_B	Internal force vector, Chapter 4
$F_i^{(a)}$	Actuation internal force vector, Chapter 4
F_S	Matrix operator for the beam non-linear equations, Chapter 4
$H()$	Heaviside function, Chapter 3
H_B	Angular momentum vector, Chapter 4
I	3×3 Inertia matrix, Chapter 4
$J(t)$	Total number of vortex particles at time t, Chapter 2
K	Kinetic energy density per unit span, Chapter 4
l	Force per unit area on fluid, Chapter 3
Δl_i	Length of i-th spanwise beam element, Chapter 4
l_r	Radial force per unit area on fluid, Chapter 3
$LFSL$	Low frequency summary level, Chapter 5
m	Blade mass per unit length, Chapter 4
m_a	Aerodynamic moment per unit length, Chapter 4
$\tilde{m}_{\xi}^{\xi}$	Sub-matrix of first mass moments of inertia, Chapter 4
M	Mach number
M_B	Internal moment vector, Chapter 4
$MFSL$	Mid frequency summary level, Chapter 5
$M_i^{(a)}$	Actuation internal moment vector, Chapter 4
M_r	Radial mach number, Chapter 3
n	Local outward normal unit vector
np	n times per revolution
N	Number of blades

N	Number of panel elements, Chapter 2
N	Number of beam elements, Chapter 4
p'	Total acoustic pressure, Chapter 3
p'_L	Loading acoustic pressure, Chapter 3
p_{ref}	Reference acoustic pressure at the threshold of hearing, Chapter 3
p'_T	Thickness acoustic pressure, Chapter 3
P	Arbitrary point for potential calculations, Chapter 2
P_B	Linear momentum vector, Chapter 4
r	Vector from source to observer, Chapter 3
r	Distance to evaluation point, Chapter 2
S	Body surface, Chapter 2
$[S]$	General 6×6 cross section stiffness matrix, Chapter 4
SPL	Sound pressure level
t	Time
T_{ij}	Lighthill stress tensor, Chapter 3
u	Flowfield velocity as a function of position and time, Chapter 2
u_a	Displacement of arbitrary point on beam reference axis, Chapter 4
U	Potential energy density per unit span, Chapter 4
v_a	Initial velocity of an arbitrary point on the a frame, Chapter 4
v_n	Normal velocity, Chapter 3
V_B	Linear velocity vector, Chapter 4
$\overline{\partial W}$	Virtual work of applied loads per unit span, Chapter 4
x	Observer position, Chapter 3

x_l	Beam reference axis curvilinear coordinate, Chapter 4
X	Vector of unknown structural variables, Chapter 4
y	Source position, Chapter 3
Z_j	Position of the j-th particle, Chapter 2
Δ	Identity matrix, Chapter 4
ε	Blob cut-off length, Chapter 2
ϕ	Scalar velocity potential, Chapter 2
γ	Generalized force strain, Chapter 4
κ	Generalized moment strain, Chapter 4
μ	Dipole intensity, Chapter 2
θ	Rotation vector in terms of Rodrigues parameters, Chapter 4
ρ_o	Density of undisturbed fluid, Chapter 3
σ	Source intensity, Chapter 2
τ	Source (retarded) time frame, Chapter 3
ω	Vorticity, Chapter 2
ω_a	Initial angular velocity of an arbitrary point on the a frame, Chapter 4
Ω	Vorticity, Chapter 2
Ω_B	Angular velocity vector, Chapter 4
ξ	Local coordinate within each beam element, Chapter 4
ψ	Azimuth angle, Chapter 1
$()_{ret}$	Retarded (source) time, Chapter 3

$(\hat{})$	Unit vector
$(\hat{})$	Beam boundary value, Chapter 4
$(\vec{})$	Vector
$(\tilde{})$	Dual matrix operator, Chapter 4

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Chapter 1 Introduction

Since first introduced in the early 20th century, rotorcraft have become invaluable for numerous civilian and military applications. A rotorcraft's vertical takeoff and landing ability, combined with unique maneuvering characteristics allow operations that could not be accomplished with fixed-wing aircraft. An estimated 45,000 rotorcraft are used worldwide for medical transport, search and rescue, fire fighting, law enforcement, offshore transport, surveillance, and attack [1].

Unfortunately, the aerodynamic principles that allow rotorcraft to be mission versatile also result in excessive noise and vibration. The noise and vibration produced by rotorcraft severely restricts their allowable missions. Vibration and noise produce restrictions due to passenger and pilot comfort, fatigue, and performance considerations [2] [3]. Vibration results in fatigue problems leading to increased cost due to reduced component life. In addition to restrictions due to pilot and passenger comfort levels, rotorcraft noise also results in severe operating limits for rotorcraft due to noise effects on the surrounding environment. Rotorcraft operations are limited in many urban areas due to the noise pollution they produce. The excessive noise is also undesirable for military rotorcraft operations for stealth reasons.

In an effort to alleviate some of the limitations due to noise and vibration, industry and academia have embarked on several research projects to understand the mechanisms that produce these disturbances and investigate methods to address them. Starting from the early days of rotorcraft operation, numerous full-scale and wind tunnel reduced-scale tests have been conducted to build databases of experimental data for rotorcraft vibration and noise research [4] - [11]. Such collections of data are invaluable in understanding the mechanisms responsible for rotor noise and vibration. In addition, extensive work has been conducted to develop analytical models for rotor blade dynamics and vibration [12] - [15] and acoustics [16] - [18]. This work has resulted in a good understanding of the mechanisms that produce the disturbances.

One of the major causes of rotorcraft noise and vibration is the unsteady aerodynamic environment that the rotor creates and the aeroelastic interaction between the rotor blades. It has been said that a rotor blade encounters almost every type of aerodynamic phenomena in one azimuthal revolution [19]. Figure 1.1 shows a schematic diagram of the typical unsteady aerodynamic environment surrounding a main rotor in forward flight. Starting from $\psi = 0^\circ$ (with counter clockwise rotation) the rotor blade first encounters a region of blade-vortex interaction (BVI). BVI is caused by interaction between the rotor blades and the strong tip vortices produced by the preceding blades. The tip vortices cause a strong three-dimensional induced velocity field, which results in rapidly varying airloads, and causes the blades to vibrate. BVI is a particularly strong contributor to rotorcraft noise and vibration, especially in the descent flight scenario [7] [8]. Although not depicted, BVI is also encountered on the retreating side by the blade.

This region is followed by a region of supercritical flow at the tip of the rotor blade caused by the addition of the free-stream velocity to the rotor tip speed. Shock waves can form in this region of high-speed flow. Shock formation causes wave drag and shock induced flow separation, resulting in varying airloads and rotor blade vibration. In addition, shock formation is a source of impulsive noise. On the retreating side ($\psi = 180^\circ$ to $\psi = 0^\circ$), the subtraction of the free-stream velocity from the rotor tip speed results in low-velocity flow. Due to lower velocity on the retreating side, the blade must be pitched at a higher angle of attack, leading to problems with stall. The high rate at which the blade is pitched as it moves around the azimuth (i.e. cyclic control) combined with the high angle of attack on the retreating side results in dynamic stalling of the blade. Similarly, the rapid change in loading due to dynamic stall results in both vibration and noise.

Figure 1.1 The unsteady aerodynamic environment around a typical helicopter rotor in forward flight [20].

Along with understanding the mechanisms of rotorcraft noise and vibration, researchers have focused on developing methods to control these mechanisms and reduce their effect. Many concepts have been proposed, investigated, tested, and implemented for this purpose. These concepts can be generally grouped into two main categories: passive and active control.

Passive control concepts have been used with varied success for alleviation of noise and vibration [21]. Examples of passive techniques for alleviation of vibration include blade-appended pendulum absorbers or pendabs, and dynamic anti-resonant vibration isolators (DAVI). A pendab is a tuned pendulum-weight installed at the root of the rotor blade to reduce the vibratory out of plane shear loads transmitted from the blade to the hub. A pendab is based on the characteristics of a basic two-degree of freedom spring-mass-damper system. In operation, a pendab generates shear loads that are out of phase with the loads generated by the blade, reducing the vibratory shear loads that are transmitted to the hub [13]. An illustration of a pendab is shown in Figure 1.2. A DAVI is a passive concept that reduces the transmission of vertical vibratory loads from the rotor hub to the fuselage. With the DAVI concept multiple tuning masses are installed at the interface between the rotor and the fuselage. These tuning masses generate dynamic loads that are out of phase with the rotor vibratory loads, thereby reducing the vibratory loads transmitted to the fuselage [13]. An illustration of a DAVI is shown in Figure 1.3. Passive techniques for acoustic control include the use of acoustic lining for cabin noise and planform modification for projected noise. Acoustic lining is used to reduce the transmission of acoustic energy to the cabin interior. An example of blade planform

modification is sweeping the blade tips to delay the onset of transonic flow on the advancing side during high-speed flight. The major limitation of passive concepts is the limited range – typically a frequency range – over which they are effective. In addition, the weight efficiency of passive concepts is questionable [22].

Figure 1.2 An illustration of a pendab.

Figure 1.3 A schematic diagram of a DAVI installation.

Active techniques for noise and vibration control are an attractive alternative to passive techniques because of their ability to be effective over a larger range. Higher

harmonic control (HHC) is an example of an active control technique for reduction of both noise and vibration. Helicopters have a swashplate to cyclically change the blade pitch once per revolution (or $1p$) of the main rotor. HHC uses actuators to change the orientation of the swashplate with respect to the main rotor shaft, changing the pitch of all the blades at higher frequencies Np and $(N \pm 1)p$, where N is the number of rotor blades. HHC has been demonstrated to be effective in reducing both noise and vibration produced by the main rotor [7]. Active techniques have been applied for cabin noise reduction through the use of distributed sensors and actuators. Accelerometers are typically used for vibration sensors and microphones for noise sensors. Examples of actuators for both noise and vibration are dual-point force actuators, inertial actuators, and microphones. The actuators are used to generate vibrations and noise that are out of phase with the original disturbances, thereby reducing them [22] [23].

With the recent introduction of smart materials, a new generation of noise and vibration control concepts has emerged, notably the use of individual blade control (IBC). IBC differs from HHC in that each blade is controlled individually in the rotating frame, increasing the number of control variables, and allowing greater control authority [24]. Using smart materials for IBC has several advantages, including reduced weight and power requirements [22], the possibility of modal control, and high-bandwidth actuation. Several notable smart material IBC concepts have been pursued to date, including the Smart-Spring root actuation, trailing-edge flap actuation, and integral-twist actuation. The Smart-Spring blade root actuation is a concept proposed by Nitzsche for dynamically modifying the impedance boundary condition of the blade root [25] [26]. By

dynamically modifying the blade root boundary conditions, the blade can be controlled to reduce its response to unsteady airloading. Trailing-edge flap actuation involves using smart-materials to deflect a trailing-edge flap at the outer portion of the blade [27] - [29], as shown in Figure 1.4. The trailing edge flap can also be used to move the blade away from the vortex core of preceding blades, reducing the magnitude of BVI response. Deflection of the trailing-edge flap results in the generation of loading that is out of phase with the unsteady airloads that the blade naturally encounters. The trailing edge flap concept has been successfully tested on model scale rotors, and is in development for a full-scale demonstration [28]. The integral twist concept [30] - [32] uses layers of active composite material embedded in the blade at $\pm 45^\circ$ to cause a twist deformation along the length of the blade as shown in Figure 1.5. Similar to the trailing edge concept, integral twist actuation generates out of phase loading that reduces the natural unsteady airloading, and can be used to move the blade away from a vortex core.

Figure 1.4 A schematic diagram of the trailing-edge flap actuation concept.

Figure 1.5 A schematic diagram of the integral twist actuation concept.

1.1 Motivation

Further research and design efforts for using smart materials to attenuate rotor noise and vibration requires the development of new computational tools. Computational tools provide a time and cost efficient means for conducting parametric and design studies. New analytical theories can be tested and verified prior to committing to expensive and time-consuming experiments. Computational trials can be used to establish parameters and limitations for experiments. After experimentation, computational results can be used for comparison and validation.

The class of code required to model smart rotor blades is in general demand in the rotorcraft industry. Since rotor blades are very flexible structures, accurate aerodynamic and structural analysis requires consideration of aeroelastic effects. As previously mentioned, rotorcraft aerodynamics is generally an unsteady problem; therefore through

aeroelastic coupling the structural analysis is also dynamic in nature. The ideal rotorcraft aeromechanical analysis code would combine high-resolution aerodynamic and structural dynamic models, have the ability to model smart IBC concepts, and have a component capable of providing acoustic results – a so-called *comprehensive* rotorcraft aeromechanical analysis.

Currently there are very few comprehensive rotorcraft aeromechanical analysis codes available. Three of the most notable codes in this class are DYMORE, CAMRAD II, and CHARM. DYMORE is a finite element based analysis for non-linear flexible multi-body structures [33] [34]. Using DYMORE's extensive library of rigid and flexible members, linear actuators, and numerous joint definitions, a sophisticated model of a rotorcraft structure can be developed, accounting for virtually the entire vehicle. A drawback to DYMORE is that its built-in aerodynamic component is a relatively simple lifting-line model with dynamic inflow, which is unable to capture aerodynamic loading oscillation at frequencies higher than Np . CAMRAD II is one of the most comprehensive rotorcraft aeromechanical analyses currently available [35]. Similar to DYMORE, CAMRAD II uses a multi-body dynamics approach for modelling rotors including consideration of multiple load paths, control system flexibility, vibration control devices, and advanced blade geometry. A lifting-line theory coupled with a free-wake analysis forms the aerodynamic component of CAMRAD II [36]. In comparison to DYMORE, CAMRAD II has a less sophisticated structural model and more-sophisticated aerodynamics. Neither DYMORE nor CAMRAD II have built-in acoustic components, or built-in modelling of smart IBC concepts. CHARM is a rotorcraft aeromechanics code that

combines very sophisticated aerodynamics with a simple blade structural dynamic model [37][38]. CHARM has primarily been used for studying rotor/wake/body aerodynamic interactions for complete rotorcraft configurations. By using “fast vortex” and “fast panel” schemes, CHARM is able to model complete rotorcraft configurations including wake effects with moderate computational cost. CHARM also includes built-in coupling with an aeroacoustic code to provide acoustic results. The main drawback to CHARM is its simple structural dynamics model: a linear finite-element based modal analysis that does not include built-in modeling of smart blade concepts.

As suggested by Wang [19], there is still room for the development of additional rotorcraft aeromechanics codes. Wang describes Sikorsky’s experience with three different rotorcraft aeroelastic codes for research, design, and development. Having three different codes gives Sikorsky the advantage of being able to crosscheck results from the different codes. In addition, having multiple codes allows Sikorsky to constantly improve each code based on experience with the other codes.

1.2 Objectives

Following the identified need for comprehensive rotorcraft aeromechanical analysis codes, the specific objectives of the current work are:

1. To develop a general active aeroelastic aeroacoustic rotorcraft code.
2. To validate the code for hingeless rotorcraft using data from the HELINOISE aerodynamic, aeroacoustic, and aeroelastic model scale rotor test.

The primary objective of this work is to develop a comprehensive rotorcraft aeromechanical analysis code – called SMARTROTOR – with aerodynamic and

structural dynamic models of sufficient resolution to capture the major physical phenomena that rotorcraft encounter: wake-induced aerodynamic loading oscillation and non-linear coupled structural dynamic deformation. SMARTROTOR is intended to be general, in the sense that it can be used to model a variety of rotor types (fully-articulated or hingeless), in a variety of flight scenarios. An acoustic component is also included in the code to allow the study of rotorcraft noise. Finally, the code supports modelling of smart IBC concepts. Although the current version of SMARTROTOR only supports modelling of active twist rotors, SMARTROTOR is organized in a modular fashion, to allow modelling of other smart IBC concepts.

SMARTROTOR is developed within the current work as part of two collaborations that provide the aerodynamic and structural components. Through collaboration with the National Technical University of Athens, Greece an existing aerodynamic code is modified within the current work for use as the aerodynamic component of SMARTROTOR. A new aeroacoustic component is developed within this thesis as an extension of the aerodynamic component. The structural component used in this work is a slightly modified version of an existing code developed by a collaborative partner at the Massachusetts Institute of Technology. New aeroelastic coupling interfaces are developed in this thesis to combine the aerodynamic/aeroacoustic and structural components. The final combined code is the SMARTROTOR code.

To validate SMARTROTOR, the HELINOISE experiment is used as a benchmark. The HELINOISE experiment is used because it has an extensive database of aerodynamic, acoustic, and aeroelastic results for a hingeless scale rotor.

The goal for conclusion of this work is the release of a code that can be used for an initial evaluation of the noise reduction capability of active twist rotor blades. The ability of active twist rotor blades to reduce vibration has been proven both computationally and experimentally [39]. As previously discussed, rotorcraft vibration and noise are often the result of the same excitation mechanisms. Thus, with successful vibration reduction, the expected result is the potential for noise reduction by using the same actuation means. The code developed within the current work provides, as a next step, a means to estimate the control authority of active twist rotor blades for noise reduction, and a tool for acoustic control law development.

1.3 Thesis Overview

The present chapter gives an introduction to the problems rotorcraft face due to noise and vibration. An introduction to the mechanisms that cause the noise and vibration is given. An overview of industry and academia attempts to address these problems is presented. The motivation and objectives for the current work are given.

Chapter 2 gives details of GENUVP, the aerodynamic component used in SMARTROTOR. The theory behind GENUVP's use of an unsteady panel method with a particle wake is described and an overview of the code structure is given. Results from several aerodynamic test cases are presented.

Chapter 3 describes a new aeroacoustic component, developed in the current work, which is added to GENUVP as part of the requirements for SMARTROTOR. The aeroacoustic component is based on a solution of the Ffowcs-Williams Hawkins equation for loading and thickness noise, the theory of which is presented. The method

of implementation of the solution into a numeric code is described. Validation results are presented for a model scale rotor in hover and forward flight.

Chapter 4 presents the aeroelastic coupling method used to form SMARTROTOR. The theory, upon which the non-linear beam element structural component is based, is presented. A classic two-dimensional aeroelastic problem is extended to three dimensions, forming the theoretical basis for aeroelastic coupling of the aerodynamic and structural components. A brief description of the aeroelastically coupled code structure is presented. Initial aeroelastic validation results for a non-rotating and a rotating rotor blade are presented.

Chapter 5 contains validation results for analysis of the HELINOISE BO105 model scale hingeless rotor. A brief description of the HELINOISE experiment is given. A description of the model created in SMARTROTOR is given. Aerodynamic loading, hub vibration, aeroelastic deformation, and aeroacoustic results are presented.

Chapter 6 contains a summary, concluding remarks, and recommendations of directions for future work.

Chapter 2 Aerodynamic Component

Computational modelling of rotorcraft aerodynamics is a unique and challenging problem. To properly model a rotorcraft, the interaction between several independently moving bodies must be captured. In addition, the influence of the wake must be accounted for as it has strong influence on the aerodynamics of the rotor. These factors combine to make rotorcraft aerodynamic modelling a difficult and computationally expensive task.

Accurate modelling of rotorcraft aerodynamics is essential for research and development in several related disciplines. Aeroelastic analysis of rotor blades requires an aerodynamic model capable of accurately predicting loads to determine the interaction between the structure and aerodynamics. Aeroacoustic predictions also require accurate aerodynamic input to capture phenomena such as blade-vortex interaction (BVI). Investigation of methods for active and passive control for vibration and noise reduction also requires accurate aerodynamic models to accurately predict the aeroelastic response.

Grid-based computational fluids dynamics (CFD) codes have been used with relative success to model rotorcraft aerodynamics [40] [41] [42]. While CFD has the potential to eventually provide very detailed rotorcraft aerodynamic calculations, its use is currently limited for several reasons. CFD rotorcraft aerodynamic analysis is too computationally

expensive to closely couple with analyses for other studies, such as aeroelasticity. Due to the presence of multiple independently moving bodies and a large flow domain, generating a suitable grid is a difficult task. In addition, CFD often has difficulty in capturing essential wake effects due to numerical dissipation. An alternative approach is to use a panel method or lifting-line model, coupled with a vortex wake model for aerodynamic modelling [36] [37]. Although they provide a lower resolution description of the flowfield than CFD, these methods provide an accurate prediction of the aerodynamic loading on the rotor blades in a variety of flight situations. More importantly, the prediction is obtained with computational costs that are orders of magnitude less than CFD [43].

The GENERAL Unsteady Vortex Particle (GENUVP) code was developed at the National Technical University of Athens (NTUA) by Voutsinas et al. [44] [45]. GENUVP is a panel method code with a vortex particle wake model for calculating the flowfield around multi-component configurations. The initial use of GENUVP focused on modelling of wind-turbines. Voutsinas and Triantos then extended the application of GENUVP to modelling of rotorcraft aerodynamics and aeroacoustics, while also adding features to reduce its computational cost [46] [47]. Through collaboration with NTUA, GENUVP is used as the aerodynamic component of the SMARTROTOR code that is developed within this work. This chapter provides an overview of the theoretical basis of GENUVP and a brief description of the GENUVP code structure. Results for aerodynamic test cases run with GENUVP are also presented.

2.1 GENUVP Theory

Voutsinas et al. give details of GENUVP's aerodynamic formulation in references [44] -[48]. For comprehensiveness, the main aspects of the formulation are described in this section.

The basis of GENUVP is the Helmholtz decomposition, which is also called the vorticity transport theorem. Using the Helmholtz decomposition, the flowfield around a rotorcraft can be decomposed into an irrotational part due to the presence of multiple bodies, and a rotational part due to the wakes emitted by lifting bodies. Let $\vec{u}(\vec{x}, t)$, $\vec{x} \in D$, $t \geq 0$ denote the velocity around a rotorcraft as a function of position $\vec{x}$ and time t , where D is the flowfield domain. Using the Helmholtz decomposition, the flowfield around the rotorcraft can then be decomposed as follows:

$$\vec{u}(\vec{x}, t) = \vec{u}_{ext}(\vec{x}, t) + \vec{u}_{solid}(\vec{x}, t) + \vec{u}_{near-wake}(\vec{x}, t) + \vec{u}_{far-wake}(\vec{x}, t) \quad (2.1).$$

In (2.1), $\vec{u}_{ext}$ is the external velocity field, which is typically a specified quantity. $\vec{u}_{solid}$ is the velocity field due to the influence of solid bodies such as the rotor blades, the fuselage, and stabilizers. $\vec{u}_{near-wake}$ and $\vec{u}_{far-wake}$ are the velocities due to the influence, in D , of the near-wake and far-wake, respectively. By using a boundary element or panel method, $\vec{u}_{solid}$ and $\vec{u}_{near-wake}$ are given in terms of singularity distributions over the surface of the solid bodies and near wake. The Biot-Savart law provides a means for obtaining $\vec{u}_{far-wake}$ as follows:

$$\vec{u}_{far-wake}(\vec{x}, t) = \int_{D_{\infty}(t)} \frac{\vec{\omega}(\vec{x}_o, t) \times (\vec{x} - \vec{x}_o)}{4\pi |\vec{x} - \vec{x}_o|^3} dD \quad (2.2).$$

Equation (2.2) can be evaluated by using vortex methods that describe the vorticity in the flowfield. Thus, equation (2.1) may be evaluated by a combination of a panel method and a vortex method, with appropriate coupling conditions between the two methods.

2.1.1 Panel method - $\bar{u}_{solid}$ & $\bar{u}_{near-wake}$

Following Hess [49] [50], a panel method was implemented in GENUVP for calculating the induced velocity in the flowfield due to solid bodies. Consider a body, described by the surface S , in an incompressible and inviscid potential flow. Since the flow is inviscid and incompressible, the continuity equation reduces to the Laplace equation:

$$\nabla^2 \phi = 0 \quad (2.3)$$

where ϕ is the scalar velocity potential. The solution for the potential is subject to two conditions: the condition of non-penetration of solid boundaries (2.4) and the condition of regularity (2.5).

$$grad\phi \cdot \bar{n}|_S = \frac{\partial \phi}{\partial n}|_S = (\bar{u}_{ext} \cdot \bar{n} - F)|_S \quad (2.4)$$

$$|grad\phi|_{\infty} \rightarrow 0 \quad (2.5)$$

In equation (2.4), $\bar{n}$ is the local unit outward normal vector on the surface S , and F is the local normal velocity due to motion of the body and far-wake induced velocity.

From Green's identity, general solutions to (2.3) may be constructed in terms of source and dipole distributions over the surfaces of the bodies in the flowfield. The potential due to a non-lifting body (e.g. a fuselage) is found by defining a continuous source distribution over the surface of the body. If σ is the local source intensity

distribution on the body, then the potential induced at a point $\bar{P}$, due to the presence of the body in the flow is given by:

$$\phi(\bar{P}) = \int_S \sigma \left(\frac{1}{r} \right) dS \quad (2.6).$$

The potential due to a lifting body (such as a rotor blade or stabilizer) is found by defining either a dipole distribution, or a source and dipole distribution, depending on whether the lifting body is modelled as thin or thick, respectively. The scope of the current work only includes thin lifting surfaces for modelling the rotor blades. A dipole distribution is necessary on lifting bodies to create circulation around the body, resulting in a net lift force. In addition, dipole sheets are trailed downstream of the lifting body, forming wake surfaces, to meet the condition of conservation of circulation as per Kelvin's theorem. If μ is the local dipole intensity distribution on the body or wake surface, then the potential induced at a point $\bar{P}$ due to the dipole distribution on the lifting body or wake surface is given by:

$$\phi(\bar{P}) = \int_S \mu \vec{n} \cdot \nabla \left(\frac{1}{r} \right) dS \quad (2.7)$$

The potential, due to the presence of solid bodies, at a point $\bar{P}$ in the flowfield is then given by:

$$\phi(\bar{P}) = -\frac{1}{4\pi} \int_{Lifting} \mu \vec{n} \cdot \nabla \left(\frac{1}{r} \right) dS - \frac{1}{4\pi} \int_{Non\ lifting} \sigma \left(\frac{1}{r} \right) dS - \frac{1}{4\pi} \int_{Wake} \mu \vec{n} \cdot \nabla \left(\frac{1}{r} \right) dS \quad (2.8)$$

It should be noted that special treatment of (2.8) is required if $\bar{P}$ is on one of the surfaces. Since equations (2.6) and (2.7) satisfy the regularity condition of (2.5), the remaining

condition to be satisfied is the non-penetration boundary condition (2.4). Combining (2.4) and (2.8) to meet the non-penetration boundary condition:

$$\left(-\frac{1}{4\pi} \int_{Lifting} \mu \nabla \left[\frac{\partial}{\partial n} \left(\frac{1}{r} \right) \right] dS - \frac{1}{4\pi} \int_{Non\ lifting} \sigma \nabla \left(\frac{1}{r} \right) dS - \frac{1}{4\pi} \int_{Wake} \mu \nabla \left[\frac{\partial}{\partial n} \left(\frac{1}{r} \right) \right] dS \right) \cdot \vec{n} = \vec{u}_{ext} \cdot \vec{n} - F \quad (2.9)$$

where equation (2.9) is evaluated over all body surfaces. Equation (2.9) defines the governing equation for the panel method used in GENUVP.

By discretizing the surfaces of the bodies and wakes into panel elements equation (2.9) can be approximated by a system of linear equations. Figure 2.1 shows an example discretization for lifting and non-lifting bodies. In GENUVP, only a single strip of wake elements – referred to as the near wake – is retained at any given timestep. Prior to the next timestep, the old near wake is transformed, and becomes a part of the far wake, as described in section 2.1.3.

Figure 2.1 An example of the discretization of a non-lifting body (representative fuselage) and lifting bodies (main rotor blades).

Each panel has a value of source σ or dipole μ intensity (depending on the surface type: σ - non-lifting, μ - lifting or wake) that is constant over the area of the panel. The source and dipole intensities can then be moved out of the integrals in equation (2.9). The remaining integrals only depend on the configuration geometry and the discretization. These integrals are evaluated at the control points of the elements, resulting in constant matrices of influence coefficients:

$$\left[C^{Lifting}_{ij} \right] \{ \mu_j \} + \left[C^{Near-wake}_{ik} \right] \{ \mu_k \} + \left[C^{Non-lifting}_{il} \right] \{ \sigma_l \} = \{ \vec{u}_{ext} \cdot \vec{n}_i - F_i \} \quad (2.10)$$

$$i = 1, (N_{Lifting} + N_{Non-lifting}) \quad j = 1, N_{Lifting} \quad k = 1, N_{Near-wake} \quad l = 1, N_{Non-lifting}$$

Equation (2.10) is the linear system approximation of equation (2.9), where $N_{Lifting}$, $N_{Non-lifting}$, and $N_{Near-wake}$ are the total number of panels on the lifting, non-lifting, and near-wake surfaces, respectively.

To obtain a unique solution to this system, certain physical conditions must be applied to the form of the near wake [46]. Setting the dipole intensity of the wake strip elements to equal the value of the adjacent emitting elements, on the trailing and tip edges of the lifting bodies, enforces a zero pressure jump Kutta condition. The near wake geometry is then determined from the flow velocity at the emission edges.

Once equation (2.10) is solved for the dipole and source distributions, the scalar velocity potential at any point in the flowfield can be determined from a discretized version of equation (2.8). From the potential, the velocity field can be calculated from equation (2.1), where:

$$\vec{u}_{panel} = \vec{u}_{solid} + \vec{u}_{near-wake} = grad(\phi) \quad (2.11).$$

Pressure distributions on solid bodies can be determined from the unsteady Bernoulli equation:

$$C_p = 1 - \frac{u^2}{u_{ref}^2} - \frac{2}{u_{ref}^2} \frac{\partial \phi}{\partial t} \quad (2.12).$$

From the pressure distribution the potential loads on lifting bodies can be calculated. It should be noted that in GENUVP, the potential load distribution on thin lifting bodies is corrected to include the leading edge suction force [51].

2.1.2 Vortex Particle Methods - $\bar{u}_{far-wake}$

Leishman [43] provides an excellent overview of the use of vortex methods for rotorcraft wake modelling. With vortex methods, the wake structure emitted by all lifting bodies is tracked through the flowfield domain. The wake structure can be described using different methods such as vortex segment elements, constant strength vorticity contours, or vortex blobs. Then, through the use of the Biot-Savart law (2.2) the wake induced velocity can be determined using any of the afore-mentioned wake structure descriptions.

Vortex wake models can be broadly classified as fixed or free wake models. Fixed-wake models pre-specify the location and shape of the wake produced by the rotorcraft based on analytical relations or experimental observations. Free-wake models allow the wake shape to develop during the simulation by numeric integration of vorticity transport equations. Comparing these two models, free-wake models are more computationally expensive due to the discretization needed to track the wake structure, and the resulting large number of calculations to capture the self-interactive nature of wake deformation.

The primary advantage of free-wake models over fixed-wake models is greater accuracy, due to the difficulty of developing analytical or experimentally based wake models for fixed-wake analyses in all but simple situations. In addition, through the use of suitable approximations the cost of free wake analysis can be kept manageable (e.g. the “fast vortex” method used by the CHARM code [37]), as will be discussed in section 2.1.4.

Following Rehbach [52], a vortex blob approach was implemented in GENUVP for modelling the far-wake. Beale and Madja [53] prove that vortex blobs are an automatically adaptive, stable, convergent, and arbitrarily high-order accuracy vortex method. With this approach, the wake is represented by a cloud of vortex particles, each with vector quantities of intensity, velocity, and position. Referring to equation (2.2), $D_\omega(t)$ is decomposed into volume elements, $D_{\omega,j}(t), j \in J(t)$, each of which has a vortex particle assigned. Let $\bar{\Omega}_j(t)$ and $\bar{Z}_j(t)$ denote, respectively, the vorticity and position of the j -th vortex particle, with vorticity defined as:

$$\bar{\Omega}_j(t) = \int_{D_{\omega,j}} \bar{\omega}(\bar{x}, t) dD \quad (2.13)$$

such that,

$$\bar{\omega}(\bar{x}, t) = \sum_j \bar{\Omega}_j(t) \delta(\bar{x} - \bar{Z}_j(t)) \quad (2.14)$$

$$\bar{\Omega}_j(t) \times \bar{Z}_j(t) = \int_{D_{\omega,j}} \bar{\omega}(\bar{x}, t) \times \bar{x} dD \quad (2.15).$$

The Biot-Savart law for far-wake induced velocity using the vortex particle model is then expressed as:

$$\bar{u}_{far-wake}(\bar{x}, t) = \sum_j \frac{\bar{\Omega}_j(t) \times (\bar{x} - \bar{Z}_j(t))}{4\pi |\bar{x} - \bar{Z}_j(t)|^3} \quad (2.16).$$

Equation (2.16) is highly singular. Therefore a smoothed approximation, given by Beale and Majda [54], is used:

$$\begin{aligned} \bar{u}_{far-wake}(\bar{x}, t) &= \sum_j \frac{\bar{\Omega}_j(t) \times \bar{R}_j}{4\pi R_j^3} f_\varepsilon(R_j) \\ \bar{R}_j &= \bar{x} - \bar{Z}_j(t) \\ f_\varepsilon(R_j) &= 1 - e^{-(R_j/\varepsilon)^3} \end{aligned} \quad (2.17)$$

where ε is the cut-off length for the vortex particles, hence the term vortex blobs. The convection of the vortex blobs is carried out in a Lagrangian sense, defined by the following equations:

$$\frac{d\bar{Z}_j}{dt} = \bar{u}(\bar{Z}_j, t) \quad (2.18)$$

$$\frac{d\bar{\Omega}_j}{dt} = (\bar{\Omega}_j \cdot \nabla) \bar{u}(\bar{Z}_j, t) = \bar{D} \cdot \bar{\Omega}_j \quad (2.19)$$

where $\bar{D}$ is the deformation tensor.

2.1.3 Near – Far Wake Coupling Conditions

After the panel method calculations of a given time step, the near-wake strip elements are transformed into vortex particles and become part of the far-wake. This is done by integrating the vorticity of each near-wake dipole element to form a vortex particle. The new vortex particles become part of the far wake, which evolves prior to the next time step. Refer to Figure 2.2 for a schematic of the vortex blob formation process.

Figure 2.2 The formation of vortex particles from a trailing edge wake strip.

2.1.4 Approximations to Reduce Computational Cost

For code effectiveness, computational cost must be minimized. Two schemes were introduced into GENUVP to reduce the code's computational cost – subgrid and particle-mesh approximations [46] [48]. Subgrid approximations are used for all simulations within the current work. Although particle-mesh approximations are not used within the current work, they are described in the current section for comprehensiveness.

Subgrid approximations were implemented to help reduce the cost of the panel method calculations. As pointed out by Hess [50], exact integral evaluations are necessary only when the distance between the evaluation control point and the inducing panel is small. In other words, at a distance the shape of the inducing panel has little effect on the induced velocity at the evaluation control point. In fact, when the distance of the evaluation point from the inducing panel is larger than 4 times the maximum diagonal of

the panel, the integral evaluation can be reduced to a point calculation. Reversing this result, one can expect that the error is also small if distant panels are grouped into larger ones, over which the integrals are evaluated [55]. A similar grouping strategy was extended to dipole distributions [48], and implemented by introducing a sequence of panelling at different levels of refinement, as shown in Figure 2.3. Calculations start at the lowest level (coarse panelling). Depending on the distance between the panel centre and the evaluation point, the calculations will either proceed with the integral evaluation over the large panel or pass to the next and more refined level of panelling as shown in Figure 2.3. Consider a panel of surface S , which contains n panels at the highest discretization level in which the unknown singularity X is defined. Let I_1 denote the value of the integral evaluated over S for unitary singularity strength. Then, the collective contribution of the n panels is approximated by:

$$I = \sum_{i=1}^n I_1 \frac{S_i}{S} X_i \quad (2.20)$$

where S_i denotes the surface area of the i -th high-level panel. Depending on the number of unknowns, the final system is either solved directly or iteratively, in which case the influence coefficient matrix need not be stored. Thus, panelling of the order of several thousands can be used on ordinary workstations.

(a) Different levels of paneling

(b) The activation of the different levels

Figure 2.3 A diagram of the grid refinement levels used for subgrid approximation.

Conventional vortex methods involve direct evaluation of the velocity and the deformation at every blob position based on the Biot-Savart law. This means that for N vortex blobs a complete time step requires N^2 point-to-point calculations. As the simulation proceeds N increases continuously in time, as does the computational cost, becoming prohibitive at large times. One way of managing cost is to use Particle-Mesh (PM) techniques [56], which reduces the cost from N^2 to $N \cdot \log(N)$ calculations per timestep.

For a large number of blobs, $\bar{u}$ and its spatial derivatives defining the deformation tensor $\bar{D}$, are evaluated at the nodes of a Cartesian grid containing $D_\omega(t)$. Then, local

interpolation is used to determine $\bar{u}$ and $\bar{D}$ at the exact positions of the blobs. To this end, the vector potential $\bar{A}$ of $\bar{u}_{far-wake}$ is introduced:

$$\nabla \times \bar{u}_{far-wake} = \bar{A} \quad (2.21)$$

and the corresponding Poisson equation:

$$\nabla^2 \bar{A} = -\bar{\omega}, \quad \text{in } D_\omega(t) \quad (2.22)$$

is solved by means of a Fourier method.

More specifically, at every time step the PM calculation procedure involves a projection step, a solution step, and an interpolation step. In the projection step, vorticity is evaluated over a Cartesian mesh that includes all vortex blobs by projecting the intensity of the vortex blobs located within a cell of the mesh onto the vertices of the cell. In the solution step, equation (2.22) is discretized using standard central differences. Three heptadiagonal linear systems are obtained. The values of $\bar{A}$ at the boundary nodes of the grid are obtained by point-to-point Biot-Savart calculations. A Fourier method is then used to solve the linear systems for the nodal values of the velocity potential $\bar{A}$. Once the nodal values of the vector potential are obtained, standard central differences are used to evaluate the velocity and the deformation at the nodes of the grid. Finally, in the interpolation step, the velocity and the deformation of vorticity of each vortex blob are calculated by interpolation from the nodal values of the nearby grid nodes.

Accuracy in PM methods is restricted by the grid cell size. In order to reduce the error of PM schemes, local corrections proposed by Anderson [57] are introduced. Experience has shown that even corrected PM schemes are not sufficiently accurate and, therefore, they should not be applied to areas of major importance. In GENUVP, a mixed scheme

was followed which excludes the regions close to solid boundaries from PM calculations. In these regions, the Biot-Savart law is used. For example, in the case of a helicopter in forward-flight, the PM region would start downstream of the tail rotor and extends to infinity.

Through a combination of subgrid and PM approximations, the computational cost in GENUVP can be kept manageable. Thus, GENUVP has the capability to model complete rotorcraft configurations, including the main and tail rotors, fuselage, and stabilizers. An example of a complete rotorcraft configuration modelled in GENUVP is shown in Figure 2.4.

Figure 2.4 A visualization of a complete rotorcraft configuration modelled with GENUVP [48].

2.2 GENUVP Code

The version of the GENUVP code used in the current work is based on the configuration of the code reported in reference [46]. GENUVP was written in standard

Fortran 77. It should be noted that certain portions of the code were written to take advantage of parallel processors if available. A few modifications were made to the code for the purposes of the current work. These modifications can be summarized as follows:

- Deactivation of the previously implemented blade structural model (based on classical beam theory).
- Provisioning for the inclusion of a new structural model by the development of new aeroelastic interfaces (refer to section 4.3).
- Modification of the body movement velocity calculations to allow modelling of articulated rotors.
- Addition of subroutines for outputting aerodynamic and aeroacoustic results in standard format for use with popularly used graphical software (e.g. TECPLOT™, MATLAB™, and MATHCAD™).

A top-level block diagram of GENUVP is shown in Figure 2.5. The GENUVP code has three major blocks: initialization, potential calculations, and vorticial calculations. The initialization block consists of reading input files that define the problem to be analyzed. Typical inputs include ambient conditions, geometry of the blades and other bodies, pitch control input (collective and cyclic), rigid body motion for articulated rotors (flap and lead-lag), and the desired panelling density. Within the potential block, the non-penetration boundary condition and the Kutta condition are satisfied. Then, output data for the current timestep can be obtained, including pressure and load distributions, the surrounding flowfield, and the acoustic pressure at observer positions (refer to Chapter 3). In the vorticial block, new wake particles are created from the near wake

panel strips, and the particle wake is convected downstream into position prior to the next set of potential calculations. The process continues until a periodic solution is obtained.

Further details of GENUVP's program structure are provided in Appendix A.

Figure 2.5 A top-level block diagram of GENUVP.

2.3 Aerodynamic Component Modelling

GENUVP was extensively tested and validated by Voutsinas et. al at NTUA [44] - [48]. However, numerous test cases were run for the purposes of testing the code modifications made within the current work, and achieving a familiarization with

GENUVP necessary for developing SMARTROTOR and performing the analysis presented in Chapter 5. In this section, results from selected test cases are presented. The approach taken for the test cases is to start with a very simple model and progressively work up to a complex model representative of that required for Chapter 5. This section describes the test case models in such order, from simplest to most complex. Validation of the test cases includes comparisons with experimental and computational data.

2.3.1 2D Airfoil Steady Lift-slope Curve

The objective of this test case is to compare results from GENUVP with simple 2D airfoil lift-slope data. This test case models a large-aspect ratio rectangular wing operating at a fixed airspeed and various angles of attack. Since GENUVP is a 3D code by nature, a high-aspect ratio wing geometry is chosen to minimize 3D effects and enable comparison with 2D data. A list of the test case parameters is given below in Table 2.1:

Span	25m
Chord	1m
Panel density	24 (spanwise) x 4 (chordwise)
Airfoil	NACA 23012
Airspeed	15 m/s
Angle of Attack Range	-10° to 10°

Table 2.1 A list of modelling parameters for the 2D airfoil steady lift slope curve test case.

Figure 2.6 A visualization of the simulation and wake for the steady lift slope curve comparison (note that the Y-axis scale is adjusted for clarity).

Figure 2.6 shows a visualization of the rectangular wing and the particle wake produced during the simulation (the near-wake is not shown). In the left half of the diagram, some roll-up of the particle wake at the tips is visible due to the capture of wing-tip vortex effects. In the right half of the diagram, the wake produced is approximately steady as one would expect. Far downstream, an initial wake rollup is visible due to the impulsive modelling of the wing velocity.

Figure 2.7 A steady lift-slope curve comparison between GENUVP and experimental results.

The method for comparison of results with experimental data is to construct a lift slope curve by running cases over the angle of attack range in 2° increments. The experimental comparison data is from Abbott and von Doenhoff [58]. As shown in Figure 2.7, GENUVP results compare well with the experimental 2D lift slope data. Some difference can be seen at the extreme ends of the comparison range, which can be attributed to the viscous effects not captured by GENUVP.

2.3.2 2D Airfoil Unsteady Lift-slope Curve

To evaluate the unsteady aerodynamic properties of GENUVP, a 2D oscillating airfoil test case was modelled and compared with experimental data. Piziali [59] performed an experiment to measure the loading over a semi-span wing undergoing pitching motions representative of a helicopter rotor blade. This experiment provides a source of experimental data for benchmarking unsteady aerodynamic results.

Similar to section 2.3.1, a high-aspect ratio wing is modelled, to minimize 3D effects, for comparison to 2D data. A summary of the test case parameters is given below in Table 2.2:

Span	30.48 m
Chord	0.3048 m
Panel density	24 (spanwise) x 4 (chordwise)
Airfoil type	NACA 0015
Reduced frequencies	0.038, 0.188
Inflow Mach number	0.293
Pitch oscillation range	4 +/- 2 degrees

Table 2.2 A list of modelling parameters for the 2D airfoil unsteady lift slope curve test case.

The wing is oscillated about the quarter-chord axis, and the potential loading normal to the inflow is used to calculate the lift coefficient. With a resolution of forty samples per cycle, the lift coefficient becomes periodic after the first cycle. Results are presented

for the unsteady curve of the fourth cycle. Figure 2.8 shows a comparison of lift coefficient versus angle of attack at a reduced frequency of 0.038; Figure 2.9 shows a comparison at a reduced frequency of 0.188. The experimental data is at the mid-span of the wing.

Figure 2.8 Unsteady lift slope curve comparison at a reduced frequency of 0.038.

Figure 2.9 Unsteady lift slope curve comparison at a reduced frequency of 0.188.

Although the characteristic unsteady effect is captured by GENUVP in both cases, it is under predicted at the lower reduced frequency and over predicted at the higher frequency. The GENUVP results show a correct trend of increased hysteresis with higher reduced frequency. The change in loading due to wing pitching velocity is important as it provides aerodynamic damping for aeroelastic modelling, as discussed in section 4.2.

2.3.3 Subgrid Approximation Study

To evaluate the effect of subgrid approximations, a model of the BO105 rotor is used [4]. The purpose of this test case is to evaluate the effect of subgrid approximations on computational cost and results. The test case modelling parameters are summarized in Table 2.3. The rotor is modelled in forward flight at an advance ratio of 0.15.

Number of blades, b	4
Rotor type	Hingeless
Airfoil type	NACA 23012 mod.
Radius, R	2m
Pretwist	-10° (linear $0R$ to R)
Nominal rotor speed, Ω	1034 RPM (108.3 rad/s)

Table 2.3 Modelling parameters for the BO105 rotor subgrid approximation study.

For this test case only the lifting surfaces of the four blades are modelled. A schematic diagram of the subgrid scaling levels for one of the blades is shown in Figure 2.10. The baseline grid of the lifting surface of the blade consists of 16 spanwise panels

by 8 chordwise panels. The subgrid scale level indicates the maximum number of distant panels that are grouped together for approximation. For example, with a level 1 approximation a maximum of 4 panels can be grouped together as a single panel, provided that they are far enough from the evaluation control point. With a level 3 approximation a maximum of 64 panels can be grouped together as a single panel, in addition to possible level 1 and 2 groupings.

Figure 2.10 A schematic diagram of the discretization and subgrid scaling levels for the subgrid approximation study (top view of the rotor blade).

First, the loading results with different subgrid levels are compared to evaluate the effect of subgrid approximations on numerical results. Figure 2.11 gives a comparison of normal force coefficient results at three radial stations on one of the blades. The subgrid approximation scheme implemented in GENUVP has negligible effect on loading results.

SUBGRID LEVEL NORMAL FORCE COEFFICIENT COMPARISON

Figure 2.11 A comparison of normal force coefficient ($C_N M^2$) with different levels of subgrid approximation.

As shown in Figure 2.12, subgrid approximation reduces the overall computational cost. A decrease in computational cost is evident between no subgrid, Level 1, and Level 2. Level 3 produced a negligible decrease in computational cost relative to Level 2 (not visible with the plotting scale). This is likely due to fact that the problem geometry limits the increase in speed that can be obtained using subgrid approximations. If a complete helicopter configuration, such as that shown in Figure 2.4, were modelled, one would expect the increase in computational cost due to subgrid approximations to be much more pronounced due the additional meshed surface area, dense panelling, and increased distances between panels. It should also be noted that the majority of the computational

cost is related to particle wake calculations. Hence, greater computational cost reduction is expected with PM techniques.

Figure 2.12 A comparison of CPU clock time for different subgrid scale levels.

2.3.4 ATR Model Scale Rotor Rigid Blade Modelling

Details of the ATR rotor are given in reference [30]. The test case modelling parameters are summarized in Table 2.4 below:

Number of blades, b	4
Rotor type	Fully Articulated
Airfoil type	NACA 0012
Radius, R	1.397
Pretwist	-10° (linear $0R$ to R)
Nominal rotor speed, Ω	687.5 RPM (72 rad/s)

Table 2.4 Modelling parameters for the ATR model rotor.

It should also be noted that the following results model the ATR operating in a heavy-gas environment (heavy gas was used during the experiment for aeroelastic scaling reasons). Also, since the ATR experiment was primarily aeroelastic in nature, experimental aerodynamic results are not available for comparison. Consequently, the comparisons in this section are with results from other rotorcraft aerodynamic codes.

Figure 2.13 A visualization of the wake produced by the ATR model rotor in hover.

Figure 2.13 is a visualization of the final wake geometry for the ATR in hover after 8 complete revolutions. Several realistic physical features can be seen in the wake shape. Contraction of the wake below the rotor, due to increased velocity, is visible. A descending helical pattern of the blade tip vortices is also visible, which compares favourably with that seen in practice (Figure 2.2 of reference [43] for example). Some upward movement of the wake particles at the hub is visible due to impulsive starting of the rotor and lack of hub or fuselage modelling.

Figure 2.14 A comparison of blade loading results with and without tip mesh refinement.

Figure 2.14 shows a comparison of blade spanwise loading for cases with and without tip refinement. This case is a good example on the importance of mesh refinement to capture the desired trends. Both results have 16 spanwise panels: one case with evenly spaced panels and the other with gradual refinement at the tip. Although both results

produce a similar integrated load on the blade, the result with tip refinement captures the expected loading peak due to wake contraction, and drop due to 3D effects.

Figure 2.15 shows a comparison of the final solution for blade spanwise loading. The comparison results are from a CAMRADII simulation, run as part of the analysis in reference [39]. The codes predict a similar spanwise load distribution, which increases with radial distance. Both codes predict a peak in the spanwise loading between $0.9 r/R$ and $1.0 r/R$. The codes differ in the magnitude of the predicted peak. This difference can be attributed to the difference in the wake geometry between the two models – specifically, the difference in the amount of wake contraction. It should be noted that the amount of wake contraction in CAMRAD II is based on a specified parameter calculated from measured wake geometry and airloads [36] – neither of which were available for the ATR experiment. Therefore, it is reasonable to assume that the GENUVP prediction is more accurate.

Figure 2.15 A comparison of blade spanwise loading for the ATR in hover.

Figure 2.16 shows a visualization of the wake produced by the ATR in forward flight at an advance ratio of 0.14. The formation and expansion of tip vortices at the edges of the rotor disc are visible.

Figure 2.16 Visualization of the wake produced by the ATR rotor in forward flight.

Figure 2.17 shows a comparison of the blade loading time history and frequency content of the time history for different timestep sizes. These results are indicative of the

testcases that were run to determine suitable timestep sizes and simulation lengths for the ATR model rotor. For each timestep size, a periodic solution is achieved after 2 complete revolutions of the rotor. With 36 timesteps per revolution, the baseline trend is captured, but the details of the loading history are not accurately predicted. By running repeated cases at different timestep sizes it was determined that the loading time history converged at approximately 60 timesteps per revolution. At timestep sizes of 60 per revolution and smaller, the difference in loading time history is negligible. Using this information, further test cases were typically run starting with 36 timesteps per revolution to quickly check the model, 60 timesteps per revolution for initial verification and comparison, and 90 timesteps per revolution for final results.

Figure 2.17 A comparison of loading results with different timestep sizes for the ATR in forward flight.

Figure 2.18 shows a comparison of the total lift loading for a single blade over a period of two complete revolutions of the rotor. The comparison is versus results for a finite-state induced flow model [60], run as part of the analysis in reference [39]. For this result, both codes compare favourably in terms of overall trends. Differences can mainly be attributed to the difference in wake modelling, which allows GENUVP to capture higher frequency loading variation, that the finite-state induced flow model is not able to capture [60].

Figure 2.18 A comparison of total lift loading on a single blade in forward flight.

Chapter 3 Acoustic Component

The study of rotorcraft aeroacoustics is the study of rotorcraft aerodynamically generated noise. In the field of rotorcraft aeroacoustics, two main methods are available for prediction in the time domain: the Ffowcs Williams-Hawkings (FW-H) equation and the Kirchhoff formulation. Brentner provides an excellent comparison of these methods [61]. The FW-H equation is the most developed and widely used method for predicting rotorcraft aeroacoustics, with several codes emerging from the pioneering work of Farassat [62] and Brentner [63]. In the 1990's Farassat and others investigated the Kirchhoff formulation for rotorcraft aeroacoustic prediction [64]. Initially it appeared as though the Kirchhoff formulation would be superior to the FW-H equation because it could capture the non-linear component of an emitted acoustic signal more efficiently than the FW-H equation. However, the Kirchhoff formulation was found to be impractical for rotorcraft aeroacoustics because erroneous results were obtained either if the Kirchhoff surface was not carefully placed, or if wakes passed through the Kirchhoff surface [65].

The present chapter presents the theory behind a new FW-H based aeroacoustic component that is added to GENUVP as part of the current work. Details of the

implementation of the theory into a numeric code are presented, along with results from validation test cases for a rotor in hover and forward flight.

3.1 Acoustic Component Theory

Ffowcs-Williams and Hawkings derived the FW-H equation [16] based on the Lighthill acoustic analogy. The FW-H equation treats the problem of sound generated by a body in arbitrary motion in a fluid as a problem of mass and momentum conservation, with a mathematical surface discontinuity corresponding to the body surface. Outside of the surface the fluid flow is the same as the physical exterior flow; the flow on the interior of the surface is arbitrary (although, typically, assumed to be at rest). Mass and momentum sources are used to create the surface discontinuity, and ultimately act as sound generators [16]. The acoustic analogy is then applied to obtain the governing equation for the problem. That is, the fluid mass and momentum conservation equations are written, including the surface discontinuity, and combined to obtain a wave equation.

Let $\bar{x}$ and $\bar{y}$ be the observer and source position vectors, respectively, and $f(\bar{y}, t) = 0$ describe the motion of the surface of a body ($f > 0$ outside the body). The FW-H equation is then written as:

$$\left(\frac{1}{c^2} \frac{\partial^2}{\partial t^2} - \nabla^2 \right) p' = \frac{\partial}{\partial t} [\rho_o v_n |\nabla f| \delta(f)] - \frac{\partial}{\partial x_i} [l_i |\nabla f| \delta(f)] + \frac{\partial^2}{\partial x_i \partial x_j} [T_{ij} H(f)] \quad (3.1).$$

Equation (3.1) gives the sound generated by the body moving through a fluid, where p' , is the acoustic pressure measured at the observer position $\bar{x}$, c and ρ are the speed of sound and the density of the undisturbed medium, respectively, $v_n = v_i n_i$ is the local

normal velocity on the body surface (n_i is the body local outward normal), l_i is the local force on the fluid per unit area, and T_{ij} is the Lighthill stress tensor. $\delta(f)$ and $H(f)$ are the Dirac delta and Heaviside functions, respectively. The three terms on the right-hand side of equation (3.1) are the thickness, loading, and quadrupole noise sources, respectively.

Figure 3.1 An illustration of the physical meaning of the sources in the FW-H equation [61].

Referring to Figure 3.1, the thickness noise source accounts for noise due to the displacement of the fluid by the finite thickness of the body. The loading noise source accounts for noise due to loading and change of loading on the body. Noise due to compressibility effects is included in the quadrupole noise source. The loading and thickness noise are surface sources; the quadrupole noise is a volume source.

For the current work, the quadrupole noise source is not considered, as it is a volume source that requires a volume-grid based flow solver. However, as stated by Brentner [65], neglecting the quadrupole term is a practical approximation as the thickness and loading noise terms account for most of the noise when the flow is not transonic. Thus, the formulation with only thickness and loading sources is valuable for a great deal of rotor noise prediction applications, including BVI.

Voutsinas and Triantos [47] previously introduced a solution of the FW-H equation into GENUVP for thickness and loading noise given by Farassat and Succi [66]. This solution discretizes the body into elements, each with an associated volume and loading. Each element is a source of thickness and loading noise. Voutsinas and Triantos were successful in implementing this solution into GENUVP and validating their results against the HELINOISE and HART experiments [47].

In the current work a new acoustic formulation is added to GENUVP, based on Farassat's 1A solution of the FW-H equation [67]. The 1A formulation is a solution of the FW-H equation for thickness and loading noise by integration over the body surface. The 1A formulation is a well-validated method that is used extensively in rotorcraft aeroacoustics [68] [69]. Currently, most advanced rotorcraft aeroacoustic codes are based on the 1A formulation, but enhancements for approximating the quadrupole term are added [70]. The derivation of the 1A formulation can be found in [67]. Therefore, only the final solution is repeated in the current work:

$$4\pi p'_L(\bar{x}, t) = \frac{1}{c} \int_{f=0} \left[\frac{\dot{l}_i \hat{r}_i}{r(1-M_r)^2} \right]_{ret} dS + \int_{f=0} \left[\frac{l_r - l_i M_i}{r^2(1-M_r)^2} \right]_{ret} dS + \frac{1}{c} \int_{f=0} \left[\frac{l_r (r \dot{M}_i \hat{r}_i + c M_r - c M^2)}{r^2(1-M_r)^3} \right] dS \quad (3.2)$$

$$4\pi p'_T(\bar{x}, t) = \int_{f=0} \left[\frac{\rho_o v_n (r \dot{M}_i \hat{r}_i + c M_r - c M^2)}{r^2(1-M_r)^3} \right]_{ret} dS \quad (3.3)$$

$$p'(\bar{x}, t) = p'_L(\bar{x}, t) + p'_T(\bar{x}, t) \quad (3.4)$$

where $\bar{r} = \bar{x} - \bar{y}$, $M_i = v_i/c$, $M_r = M_i r_i / r$, and $l_r = l_i r_i / r$. Equations (3.2) – (3.4) give the loading (p'_L), thickness (p'_T), and total (p'), acoustic pressure at $\bar{x}$, respectively. The $[]_{ret}$ subscripts indicate that the integrals are evaluated in the retarded or source time

frame τ – the time that the acoustic signal is emitted. The acoustic signal history is received at the observer position in the observer time frame t . The integrals in (3.2) and (3.3) are approximated by discretizing the body surface into elements and calculating the contribution of each element.

3.2 Acoustic Component Code

The new acoustic component is coded as an add-on to the GENUVP aerodynamic code, to obtain aeroacoustic results. The acoustic component uses the same input data as the aerodynamic case, with the observer locations (i.e. the microphone positions) as the only additional input. The acoustic code uses the same discretization as the panel method, treating each panel as an acoustic source. After the potential calculations are completed, the aerodynamic results are used to calculate the emitted acoustic signals. At the end of the simulation, the predicted acoustic pressure history at each specified microphone location is obtained.

The loading noise contribution of each panel is calculated as a function of the potential loading, velocity, and distance to the microphone. Viscous corrections are not included. Special consideration is needed for the thickness noise term since thin lifting bodies are used in the current work and the thickness noise term is dependant on v_n , the local normal velocity on the *actual* blade surface. Therefore, for each panel projected upper and lower surface panels are determined from the known airfoil geometry. The normal velocity is then calculated for the upper and lower surface panels using the velocity of the mean surface panel and the normal vectors of the projected panels; refer to

Figure 3.2. The thickness noise contribution is then evaluated as the sum of the contribution of the upper and lower projected panels.

Figure 3.2 An illustration of the method used to calculate the normal velocity on the actual blade surface.

Once the acoustic pressure contribution of each panel is calculated, the total acoustic pressure history at each microphone can be determined by summation, taking into account time sequencing due to the difference in travel time for each acoustic emission. Referring to Figure 3.3, consider an acoustic signal $p'(\vec{x}, \tau_i)^k$ emitted by the k -th panel at time τ_i in the source time frame, which corresponds to the time frame in which the aerodynamic potential calculations are computed. An approximation for the relation between source time (i.e. the time the signal is emitted) and observer time (i.e. the time the signal is received by the observer) is as follows:

$$t = \tau + \frac{|\vec{r}|}{c + \vec{u}_{\infty} \cdot \hat{r}} \quad (3.5).$$

Equation (3.5) accounts for travel time from the acoustic source to the observer position. The total acoustic pressure history, $p'(\vec{x}, t_j)$ at time t_j , is then updated by the contribution of the k -th panel at τ_i . The observer time frame is discretized into the same evenly spaced intervals as the source time frame based on the aerodynamic time discretization. Typically, the conversion from source to observer time does not produce a result that coincides exactly with an endpoint of the interval of the time discretization. Linear interpolation is used to distribute the contribution of the signal between the endpoints that bound the interval that the signal arrives in.

Figure 3.3 A diagram of the time-shifting scheme used to determine the acoustic pressure history in the observer time scale.

3.3 Acoustic Component Modelling and Validation

For the purpose of initial validation of the new acoustic component added to GENUVP, a $1/4$ -scale model UH1H rotor is modelled. Details of the UH1H rotor are

given in reference [71]. Some of the key modeling parameters of the rotor are summarized in Table 3.1. The UH1H rotor was tested in standard atmospheric conditions.

Number of blades, b	2
Rotor type	Teetering
Airfoil type	NACA 0012
Radius, R	1.829m
Pretwist	-10.9° (linear 0.08 R to R)
Nominal rotor speed, Ω	1296 RPM (135.7 rad/s)

Table 3.1 Key modelling parameters for the 1/4-scale UH1H rotor.

To accurately capture the acoustic signal, very fine discretization is required in both space and time. By running repeated test cases it was found that the timestep size to accurately capture the acoustic signal is 90 timesteps per revolution. In comparison, a resolution of 60 timesteps per revolution is sufficient for aerodynamic results; refer to section 2.3.4. Similarly, it was found that a very fine chordwise discretization, with refinement at the leading edge, was required for accurate results. Figure 3.4 shows a comparison of the blade tip panelling used for aerodynamic results with the ATR, versus that used for acoustic results with the UH1H rotor.

Figure 3.4 A comparison of the panelling required for the UH1H acoustic results versus that for the ATR aerodynamic results.

Figures 3.5-3.7 show comparisons of loading, thickness, and total acoustic pressure, respectively, measured at a microphone situated 3m from the hub, in the plane of the rotor. Comparison results are from the original version of WOPWOP [63], which uses a slightly modified Farassat 1A acoustic formulation. The two codes agree well in all three comparisons. The slight time shift visible in the results is caused by the difference, between the two codes, of the starting position of the rotor.

Figure 3.5 A comparison of loading acoustic pressure for the UH1H rotor in hover.

Figure 3.6 A comparison of thickness acoustic pressure for the UH1H rotor in hover.

Figure 3.7 A comparison of the total acoustic pressure for the UH1H rotor in hover.

Figure 3.8 shows a visualization of the wake produced by the UH1H rotor in forward flight ($\mu = 0.208$, 8.85° forward disc tilt) and the locations of two microphones where acoustic pressure is compared. Microphone 4 is situated 4m ahead of the rotor hub on the advancing side; microphone 5 is situated 4m ahead of the rotor hub on the retreating side.

Figure 3.8 A visualization of the wake produced by the UH1H rotor in forward flight and the locations of the microphones.

Figures 3.9 and 3.10 show a comparison of the total acoustic pressures at the two microphone locations. Comparison is given with both WOPWOP [63] and experimental

data [71] on the advancing side, and with experimental data on the retreating side. In both cases, the overall trend is captured by GENUVP. The higher frequency component present in the experimental results is not captured by either code. While some of this high frequency component may be due to experimental scatter, it is suspected that some of the oscillation may be due to blade flexibility effects. Ffowcs-Williams and Hawkings state that the only restriction placed upon a surface in the FW-H equation is that it is smooth; otherwise it is allowed to move in an arbitrary fashion and change its shape or orientation [16]. This suggests that, in principle, the effect of elastic motion of the body surface on the emitted acoustic signal can be captured by the FW-H equation. This can be explored further using SMARTROTOR.

Sound pressure measurements are often evaluated on a decibel scale called the sound pressure level (SPL), which is calculated as follows:

$$SPL = 20 \cdot \log \left(\frac{p'}{p_{ref}} \right) \quad (3.6)$$

where $p_{ref} = 20 \mu Pa$, the threshold of audible sound.

Figure 3.11 shows a comparison of sound pressure level versus frequency for the microphone on the advancing side of the rotor. The peaks in SPL at harmonics of the blade passage frequency are accurately captured by GENUVP. The overall trend of the frequency spectrum is captured by GENUVP.

Figure 3.9 A comparison of total acoustic pressure (advancing side microphone) for the UH1H in forward flight.

Figure 3.10 A comparison of total acoustic pressure (retreating side microphone) for the UH1H in forward flight.

Figure 3.11 A comparison of frequency content of the total acoustic pressure prediction of the advancing side microphone - UH1H in forward flight.

Chapter 4 Aeroelastic Coupling

In the current chapter, the method of aeroelastic coupling of the aerodynamic and structural components that form SMARTROTOR is presented. The theory behind the structural component is reviewed, and a brief description of the implementation of this theory into a code is given. The aeroelastic coupling method, based on extension of the classic 2D aeroelastic section problem, is described. An overview of the implementation of this method, to form a closely-coupled aeroelastic code, is presented. Initial aeroelastic validation results, for a fixed and a rotating blade, are given.

4.1 Structural Component

4.1.1 Rotor Blade Structural Modeling

Rotor blades are typically modelled using a rotating beam approximation¹ with consideration of flapping, lagging, extension and torsional degrees of elastic freedom, and rigid body flapping and lead-lagging degrees of freedom for articulated rotors.

¹ Note that the terms beam and blade are used interchangeable in this section to describe the rotor blade structure.

Examples of simple equations of motion for rotor blades, considering these degrees of freedom, can be found in references [12] and [13]. Although such equations provide a starting point for analysis of rotor blade structural dynamics, they are based on linear beam deformation theory. Helicopter rotor blades typically undergo large deformations that make equations of motion based on linear beam deformation unsuitable. An additional consideration is inertial and elastic coupling between the degrees of freedom of the blade due to the mass distribution and material properties of the blade structure.

Hodges [15] used a mixed variational approach to develop a geometrically exact formulation for the dynamics of rotating beams. Hodges' formulation decomposes the analysis of the three-dimensional blade structure into a linear analysis of the cross-section geometry and material properties, and a non-linear one-dimensional analysis along the beam reference line. A geometrically exact description of the beam reference line makes Hodges' formulation suitable for capturing the large deformations that rotor blades typically experience. In addition, Hodges's formulation uses full constitutive relations for elastic and inertial properties, allowing the capture of coupling between the beam degrees of freedom.

4.1.2 Structural Component Theory

The structural component used in SMARTROTOR is based on equations of motion, for the dynamics of rotating beams, developed by Shang [72]. Shang extended the work of Hodges [15] by writing the mixed variational equations of motion in a global rotating frame, and applying the finite element method for spatial discretization. Portions of

Shang's derivation are presented in the current work to provide a comprehensive description of the theory behind the current version of SMARTROTOR.

In Shang's formulation, three coordinate frames are used in the equations of motion: the global a frame, local beam undeformed b frame, and the local beam deformed B frame. A schematic diagram of the coordinate frames is shown in Figure 4.1. The a frame is a global rotating reference frame which follows the beam around the rotor azimuth. The b frame describes the local undeformed orientation of the beam reference line as a function of the spanwise curvilinear coordinate x_l . The b frame accounts for the initial curvature and pretwist of the beam and can be related to the a frame through direction cosine matrices. The B frame describes the local deformed orientation of the beam reference line as a function of x_l . The B frame includes both rigid and elastic rotations. The use of these three reference frames allows a compact geometrically exact beam formulation, while also allowing the use of simple constitutive relations using the deformed reference frame B [72].

Figure 4.1 A schematic diagram of the coordinate frames used by the structural component.

Conversion of measures between the reference frames is accomplished by using transformation matrices, an excellent review of which is given by Kane [73]. It should be noted that Hodges and Shang use transformation matrices that are the transpose of those in used by Kane. The following notation is used for transformation matrices:

$$Y_a = C^{ab}Y_b, \quad Y_B = C^{Ba}Y_a, \quad Y_b = C^{Bb}Y_b \quad (4.1)$$

where Y_a , Y_b , and Y_B , are arbitrary vectors of the same measure in the a , b , and B frames respectively, denoted by the subscripts. C^{ab} , C^{Ba} , and C^{Bb} are the transformation matrices for conversion from b to a , a to B , and b to B , respectively, as denoted by the superscripts.

The beam equations of motion are derived from Hamilton's principle, which is written as [15]:

$$\int_{t_1}^{t_2} \int_0^l [\delta(K - U) + \overline{\delta W}] dx_1 dt = \overline{\delta A} \quad (4.2)$$

where K and U are the potential and strain energy densities per unit length, respectively. $\overline{\delta W}$ is the virtual work of applied loads per unit length and $\overline{\delta A}$ is the virtual action at the ends of the spatial and temporal domains. The kinetic and potential strain energy variations in equation (4.2) can be related to section stress resultants and section linear and angular momentum as follows:

$$\begin{aligned} F_B &= \left(\frac{\partial U}{\partial \gamma} \right)^T, \quad M_B = \left(\frac{\partial U}{\partial \kappa} \right)^T \\ P_B &= \left(\frac{\partial K}{\partial V_B} \right)^T, \quad H_B = \left(\frac{\partial K}{\partial \Omega_B} \right)^T \end{aligned} \quad (4.3)$$

where F_B and M_B are vectors of the internal force and moment, γ and κ are the generalized force and moment strain vectors, P_B and H_B are vectors of the section linear

and angular momentum, and V_B and Ω_B are the linear and angular velocity vectors, respectively. To form a mixed formulation, Lagrange multipliers are used to satisfy the following geometrically exact kinematic equations for the generalized strains and velocities [72]:

$$\begin{aligned}
 \gamma^* &= C^{Ba}(C^{ab}e_1 + u_a') - e_1 \\
 \kappa^* &= C^{ba} \left(\frac{\Delta - \frac{\tilde{\theta}}{2}}{1 + \frac{\theta^T \theta}{4}} \right) \dot{\theta}, \\
 V_B^* &= C^{Ba}(v_a + \dot{u}_a + \tilde{\omega}_a u_a) \\
 \Omega_B^* &= C^{ba} \left(\frac{\Delta - \frac{\tilde{\theta}}{2}}{1 + \frac{\theta^T \theta}{4}} \right) \dot{\theta} + C^{Ba} \omega_a
 \end{aligned} \tag{4.4}$$

where e_1 is the unit vector $[1 \ 0 \ 0]^T$, u_a is the displacement vector, and v_a and ω_a are the initial linear and angular velocity vectors, respectively. θ is the vector that measures rotation, and it is expressed in terms of Rodrigues parameters as follows:

$$\theta_i = 2E_i \tan\left(\frac{\alpha}{2}\right) \tag{4.5}.$$

θ defines a rotation of magnitude α about the unit vector $E_i = [E_1 \ E_2 \ E_3]^T$. The prime and dot symbols denote differentiation with respect to the curvilinear coordinate x_1 and time, respectively. $\tilde{\theta}$ converts the θ vector to its dual matrix (also known as the cross product operator).

Using equation (4.3), and applying Lagrange multipliers to enforce (4.4), equation (4.2) is written as:

$$\begin{aligned}
& \int_{t_1}^{t_2} \int_0^l [\delta V_B^{*T} P_B + \delta \Omega_B^{*T} H_B - \delta \gamma^{*T} F_B - \delta \kappa^{*T} M_B \\
& + \delta F_B^T (\gamma - \gamma^*) + \delta M_B^T (\kappa - \kappa^*) - \delta P_B^T (V_B - V_B^*) - \delta H_B^T (\Omega_B - \Omega_B^*)] dx_1 dt \\
& + \int_{t_1}^{t_2} \int_0^l \overline{\delta W} dx_1 dt = \partial \bar{A}
\end{aligned} \tag{4.6}$$

Equation (4.6) can be written in the a frame according to the following principles: 1) all variational terms are measured in the a frame, 2) displacements and rotations are measured in the a frame, and 3) strains, velocities, forces and momenta are measured in the B frame [72]. Quantities measured in the B frame are transformed to the a frame by using the concept of “virtual rotation” [15]. A rotation matrix C , containing a measure of the unknown rigid and elastic rotation of the deformed beam, is defined as:

$$C = C^{ab} C^{Ba} \tag{4.7}$$

and expressed in terms of the previously defined Rodrigues parameters as follows:

$$C = \frac{\left(1 - \frac{\theta^T \theta}{4}\right) \Delta - \tilde{\theta} + \frac{\theta \theta^T}{2}}{1 + \frac{\theta^T \theta}{4}} \tag{4.8}$$

It should be noted that the Rodrigues parameters are related to the rotation sequence given in equation (4.7), and, thus, do not correspond to a physical rotation sequence. By multiplication of the appropriate transformation matrices, an alternate set of Rodrigues parameters can be obtained that define the physical rotation of interest.

The variational formulation, based on exact intrinsic equations for dynamics of moving beams, is then written in the a frame as [72]:

$$\begin{aligned}
& \int_{t_1}^{t_2} \delta \Pi_a dt = 0 \\
& \delta \Pi_a = \int_0^l \left[\delta u_a'^T C^T C^{ab} F_B + \delta u_a^T \left[\left(C^T C^{ab} P_B \right)' + \tilde{\omega}_a C^T C^{ab} P_B \right] \right. \\
& \quad + \overline{\delta \psi}_a'^T C^T C^{ab} M_B - \overline{\delta \psi}_a^T C^T C^{ab} (\tilde{e}_1 + \tilde{\gamma}) F_B \\
& \quad \left. + \overline{\delta \psi}_a^T \left[\left(C^T C^{ab} H_B \right)' + \tilde{\omega}_a C^T C^{ab} H_B + C^T C^{ab} \tilde{V}_B P_B \right] \right. \\
& \quad - \overline{\delta F}_a^T \left[C^T C^{ab} (e_1 + \gamma) - C^{ab} e_1 \right] - \overline{\delta F}_a'^T u_a - \overline{\delta M}_a^T \left(\Delta + \frac{\tilde{\theta}}{2} + \frac{\theta \theta^T}{4} \right) C^{ab} \kappa - \overline{\delta M}_a'^T \theta \\
& \quad + \overline{\delta P}_a^T \left(C^T C^{ab} V_B - v_a - \tilde{\omega}_a u_a \right) - \overline{\delta P}_a'^T \dot{u}_a + \overline{\delta H}_a^T \left(\Delta - \frac{\tilde{\theta}}{2} + \frac{\theta \theta^T}{4} \right) \left(C^T C^{ab} \Omega_B - \omega_a \right) \\
& \quad \left. - \overline{\delta H}_a'^T \dot{\theta} - \delta u_a^T f_a - \overline{\delta \psi}_a^T m_a \right] dx_1 - \left(\delta u_a^T \hat{F}_a + \overline{\delta \psi}_a^T \hat{M}_a - \overline{\delta F}_a^T \hat{u}_a - \overline{\delta M}_a^T \hat{\theta} \right) \Big|_0^l
\end{aligned} \tag{4.9}$$

where f_a and m_a are vector quantities of the applied aerodynamic forces and moments per unit length, respectively. The $(\hat{\cdot})$ terms in (4.9) denote boundary values, and account for all possible beam boundary conditions.

By applying the finite element method, equation (4.9) can be discretized [72]. The beam is discretized into N elements, of length Δl_i , with the index i denoting the i -th element. Shang uses simple shape functions for discretization [15]. The following discretization and interpolation is applied to each element:

$$\begin{aligned}
x_1 &= x_{1i} + \xi \Delta l_i, \quad dx_1 = \Delta l_i d\xi, \quad (\cdot)' = \frac{1}{\Delta l_i} \frac{d}{d\xi} (\cdot) \\
\delta u_a &= \delta u_i (1 - \xi) + \delta u_{i+1} \xi & u_a &= u_i \\
\overline{\delta \psi}_a &= \overline{\delta \psi}_i (1 - \xi) + \overline{\delta \psi}_{i+1} \xi & \theta &= \theta_i \\
\overline{\delta F}_a &= \overline{\delta F}_i (1 - \xi) + \overline{\delta F}_{i+1} \xi & F_B &= F_i \\
\overline{\delta M}_a &= \overline{\delta M}_i (1 - \xi) + \overline{\delta M}_{i+1} \xi & M_B &= M_i \\
\overline{\delta P}_a &= \overline{\delta P}_i & P_B &= P_i \\
\overline{\delta H}_a &= \overline{\delta H}_i & H_B &= H_i
\end{aligned} \tag{4.10}$$

where u_i , θ_i , F_i , and M_i are constant nodal vector quantities, and P_i and M_i are constant element vector quantities. ξ is the element local curvilinear coordinate, varying over the element from 0 to 1.

Equation (4.9) is then written in discretized form as:

$$\int_{t_1}^{t_2} \sum_{i=1}^N \delta \Pi_i dt = 0$$

$$\sum_{i=1}^N \delta \Pi_i = \sum_{i=1}^N \left[\delta u_i^T f_{u_i} + \overline{\delta \psi}_i^T f_{\psi_i} + \overline{\delta F}_i^T f_{F_i} + \overline{\delta M}_i^T f_{M_i} + \overline{\delta P}_i^T f_{P_i} + \overline{\delta H}_i^T f_{H_i} \right. \\ \left. + \delta u_{i+1}^T f_{u_{i+1}} + \overline{\delta \psi}_{i+1}^T f_{\psi_{i+1}} + \overline{\delta F}_{i+1}^T f_{F_{i+1}} + \overline{\delta M}_{i+1}^T f_{M_{i+1}} \right] \\ - \left[\delta u_{N+1}^T \hat{F}_{N+1} + \overline{\delta \psi}_{N+1}^T \hat{M}_{N+1} - \overline{\delta F}_{N+1}^T \hat{u}_{N+1} - \overline{\delta M}_{N+1}^T \hat{\theta}_{N+1} \right. \\ \left. - \delta u_1^T \hat{F}_1 - \overline{\delta \psi}_1^T \hat{M}_1 + \overline{\delta F}_1^T \hat{u}_1 + \overline{\delta M}_1^T \hat{\theta}_1 \right] \quad (4.11)$$

where f_{u_i} , f_{ψ_i} , ..., $f_{M_{i+1}}$ are explicitly integrated element functions, expressed as:

$$\begin{aligned}
f_{u_i} &= -C^T C^{ab} F_i + \frac{\Delta l_i}{2} \tilde{\omega}_a C^T C^{ab} P_i + \frac{\Delta l_i}{2} (C^T C^{ab} P_i)^\top - \bar{f}_i \\
f_{u_{i+1}} &= C^T C^{ab} F_i + \frac{\Delta l_i}{2} \tilde{\omega}_a C^T C^{ab} P_i + \frac{\Delta l_i}{2} (C^T C^{ab} P_i)^\top - \bar{f}_{i+1} \\
f_{\psi_i} &= -C^T C^{ab} M_i - \frac{\Delta l_i}{2} C^T C^{ab} (\tilde{e}_1 + \tilde{\gamma}_i) F_i \\
&\quad + \frac{\Delta l_i}{2} (\tilde{\omega}_a C^T C^{ab} H_i + C^T C^{ab} \tilde{V}_i P_i) + \frac{\Delta l_i}{2} (C^T C^{ab} H_i)^\top - \bar{m}_i \\
f_{\psi_{i+1}} &= C^T C^{ab} M_i - \frac{\Delta l_i}{2} C^T C^{ab} (\tilde{e}_1 + \tilde{\gamma}_i) F_i \\
&\quad + \frac{\Delta l_i}{2} (\tilde{\omega}_a C^T C^{ab} H_i + C^T C^{ab} \tilde{V}_i P_i) + \frac{\Delta l_i}{2} (C^T C^{ab} H_i)^\top - \bar{m}_{i+1} \\
f_{F_i} &= u_i - \frac{\Delta l_i}{2} [C^T C^{ab} (e_1 + \gamma_i) - C^{ab} e_1] \\
f_{F_{i+1}} &= -u_i - \frac{\Delta l_i}{2} [C^T C^{ab} (e_1 + \gamma_i) - C^{ab} e_1] \\
f_{M_i} &= \theta_i - \frac{\Delta l_i}{2} \left(\Delta + \frac{\tilde{\theta}_i}{2} + \frac{\theta_i \theta_i^T}{4} \right) C^{ab} \kappa_i \\
f_{M_{i+1}} &= -\theta_i - \frac{\Delta l_i}{2} \left(\Delta + \frac{\tilde{\theta}_i}{2} + \frac{\theta_i \theta_i^T}{4} \right) C^{ab} \kappa_i \\
f_{P_i} &= C^T C^{ab} V_i - v_a - \tilde{\omega}_a u_i - \dot{u}_i \\
f_{H_i} &= \Omega_i - C^{ba} C \omega_a - C^{ba} \left(\frac{\Delta - \frac{\tilde{\theta}_i}{2}}{1 + \frac{\theta_i^T \theta_i}{4}} \right) \dot{\theta}_i
\end{aligned} \tag{4.12}.$$

In equation (4.12), the distributed aerodynamic loads and moments are converted to effective discrete nodal loads as follows:

$$\begin{aligned}
\bar{f}_i &= \int_{\Delta l_i} (1 - \xi) f_a dx_1, \quad \bar{f}_{i+1} = \int_{\Delta l_i} \xi f_a dx_1 \\
\bar{m}_i &= \int_{\Delta l_i} (1 - \xi) m_a dx_1, \quad \bar{m}_{i+1} = \int_{\Delta l_i} \xi m_a dx_1
\end{aligned} \tag{4.13}.$$

Since the forces, generalized strains, momenta, and velocities are B (deformed) frame measures, simple constitutive relations are used to substitute the unknown generalized strains and velocities in terms of the unknown internal loads and momenta, respectively:

$$\begin{Bmatrix} \gamma_i \\ \kappa_i \end{Bmatrix} = [S]_i^{-1} \begin{Bmatrix} F_i \\ M_i \end{Bmatrix}, \quad \begin{Bmatrix} V_i \\ \Omega_i \end{Bmatrix} = \begin{bmatrix} m\Delta & -m\tilde{\xi} \\ m\tilde{\xi} & I \end{bmatrix}_i^{-1} \begin{Bmatrix} P_i \\ H_i \end{Bmatrix} \quad (4.14)$$

where $[S]_i$ is the 6×6 cross-section stiffness matrix, m is the linear mass density, and I is the cross-section moment of inertia matrix. $m\tilde{\xi}$ is the sub-matrix of the first mass moments of inertia, which accounts for coupling between linear and angular momenta due to the offset between the beam reference and centre of mass axes.

The beam equations of motion are obtained by assembling the element functions and combining functions relating to the same node. Let X be a vector containing the unknown discrete measures of displacement, rotation, internal loads, momenta, and the unknown boundary values. The form of X depends on the type of rotor being considered. For a hingeless rotor $\hat{u}_1 = \hat{\theta}_1 = \hat{F}_{N+1} = \hat{M}_{N+1} = 0$. The unknown boundary quantities are the root forces and moments and the tip displacements and rotations. Then, X takes the following form:

$$X = \left[\hat{F}_1^T \quad \hat{M}_1^T \quad u_1^T \quad \theta_1^T \quad F_1^T \quad M_1^T \quad P_1^T \quad H_1^T \quad \dots u_N^T \quad \theta_N^T \quad F_N^T \quad M_N^T \quad P_N^T \quad H_N^T \quad \hat{u}_{N+1}^T \quad \hat{\theta}_{N+1}^T \right]^T \quad (4.15).$$

Equation (4.11) is then written as:

$$\int_{t_1}^{t_2} \delta X^T [F_S(X, \dot{X})] dt = 0 \quad (4.16)$$

where F_S is the matrix operator containing the complete non-linear equations of motion. F_S also depends on the type of rotor considered. For a hingeless rotor it is written as:

$$F_S = \begin{bmatrix} f_{u_1}^{(1)} + \hat{F}_1 \\ f_{\psi_1}^{(1)} + \hat{M}_1 \\ f_{F_1}^{(1)} \\ f_{M_1}^{(1)} \\ f_{P_1}^{(1)} \\ f_{H_1}^{(1)} \\ f_{u_2}^{(1)} + f_{u_2}^{(2)} \\ f_{\psi_2}^{(1)} + f_{\psi_2}^{(2)} \\ f_{F_2}^{(1)} + f_{F_2}^{(2)} \\ f_{M_2}^{(1)} + f_{M_2}^{(2)} \\ f_{P_2}^{(2)} \\ f_{H_2}^{(2)} \\ f_{u_3}^{(2)} + f_{u_3}^{(3)} \\ f_{\psi_3}^{(2)} + f_{\psi_3}^{(3)} \\ f_{F_3}^{(2)} + f_{F_3}^{(3)} \\ f_{M_3}^{(2)} + f_{M_3}^{(3)} \\ \vdots \\ f_{u_{N+1}}^{(N)} \\ f_{\psi_{N+1}}^{(N)} \\ f_{F_{N+1}}^{(N)} + \hat{u}_{N+1} \\ f_{M_{N+1}}^{(N)} + \hat{\theta}_{N+1} \end{bmatrix} \equiv \{0\} \quad (4.17)$$

where the superscripts indicate the element number, and the subscripts indicate the node number.

Articulated rotors can be modelled using the same formulation. However, for an articulated rotor, the root rotation vector $\hat{\theta}_1$ is unknown, as are the second and third elements of the root moment vector, $\hat{M}_1$. Equation (4.15) is, accordingly, modified to include the additional unknown rotations, and $\hat{\theta}_1$ is added to the fourth row of F_S in equation (4.17).

Cesnik and Shin extended Shang's formulation to the modelling of blades with integral twist actuation [74]. The effect of embedded piezoelectric composite layers can be included by modifying the constitutive equation (4.14):

$$\begin{Bmatrix} \gamma_i \\ \kappa_i \end{Bmatrix} = [S]^{-1} \begin{Bmatrix} F_i + F_i^{(a)} \\ M_i + M_i^{(a)} \end{Bmatrix} \quad (4.18)$$

where $F_i^{(a)}$ and $M_i^{(a)}$ are the section internal force and moment vectors due to the activation of the piezoelectric fibres. Substitution of equation (4.18) into the first equation of (4.14) adapts the beam equations of motion to include the additional cross section strain that occurs when the piezoelectric fibres are activated. $F_i^{(a)}$ and $M_i^{(a)}$ are functions of the cross section geometry, material properties, and the electric field (voltage) applied to the piezoelectric fibres. Expressions for $F_i^{(a)}$ and $M_i^{(a)}$ as a function of applied voltage may be obtained through suitable analysis of the blade cross section [74] [75].

4.1.3 Structural Component Code

Equation (4.17) is a non-linear system of equations giving a geometrically exact description of the dynamics of a rotating beam. To apply a numeric method to solve these equations two main considerations are necessary; namely, a method for computing the derivatives in equation (4.12) and a scheme for solution of the non-linear equations of motion. When selecting numeric methods for differentiation and non-linear system solution the accuracy, stability, and computational cost of the methods must be considered [13]. Accuracy is of course important to obtain a solution with minimal

numerical error. Stability is an especially important consideration to avoid introduction of numerical oscillations and instabilities into the solution. It should be noted that the selection of an accurate method does not necessarily guarantee stability, and vice versa. Finally, computational cost must be considered.

Cheng [76] developed a structural code for SMARTROTOR based on Shang's formulation, including Cesnik and Shin's modification to account for the effects of integral twist actuation. The structural code solves the non-linear equations of motion (4.17) in a time-marching fashion. An Euler second-order backward finite difference scheme is applied to numerically approximate the derivatives in equation (4.12). Equation (4.17) is solved iteratively for the X vector of unknowns by using the Newton-Raphson method [77], since equation (4.12) is already in a format that facilitates the exact analytical determination of the Jacobian matrix.

A top-level block diagram of the structural code is shown in Figure 4.2. The structural code contains four main subroutine blocks: ATR_INITIALIZE, FORCE_NODE, STRUCTURAL_COMPONENT, and OUTPUT. In ATR_INITIALIZE, a data file that defines the configuration to be modelled is read, and code variables are initialized. Distributed aerodynamic loads are converted into effective nodal loads in FORCE_NODE, following equation (4.13). Subroutine STRUCTURAL_COMPONENT solves the non-linear system of equations using the iterative Newton-Raphson solver. The structural code performs two analyses: 1) an initial steady-state analysis, and 2) a dynamic analysis. Both analyses use ATR_INITIALIZE for initialization.

In the dynamic analysis, `FORCE_NODE` and `STRUCTURAL_COMPONENT` are called once per each rotor blade, and per each timestep. At the end of the dynamic analysis, `OUTPUT` is called to write structural results to file, including tip deformations and rotations, root forces and moments, and momenta and deformations along the beam length.

The steady-state analysis determines the initial equilibrium state of the beam. The steady-state analysis is similar to the dynamic analysis, and based on the same four main subroutine blocks, modified to remove time-dependant terms. The output of the static analysis provides the initial condition input for the dynamic analysis.

Figure 4.2 A top level block diagram of the structural component code.

Initial testing of the structural component was accomplished in reference [76] during the initial development of SMARTROTOR. Several comparisons were reported with DYMORE [33], to validate the structural component. However, as discussed in Chapter 6, further testing of the structural code should be accomplished concurrently with further development of SMARTROTOR. And importantly, testing is necessary to definitively establish the stability characteristics of the numerical methods used by the structural component.

A few modifications were necessary to the code reported by Cheng in reference [76] for use with the current version of SMARTROTOR. These modifications include changes to the method of application of root boundary conditions, the method of application of pitch control, and the conversion of the Rodrigues parameters from equation (4.5) to physical orientation angles.

4.2 Aeroelastic Coupling

The theory behind the method of aeroelastic coupling of SMARTROTOR's aerodynamic and structural components is based on the classic aeroelastic problem of a two-dimensional airfoil section with elastic degrees of freedom. A schematic diagram of this problem is depicted in Figure 4.3. An airfoil section is shown with degrees of freedom in the plunge (vertical), lag (horizontal), and pitch (rotational) degrees of freedom. These degrees of freedom are elastically restrained, represented in Figure 4.3 by linear and torsional springs at the beam reference point. Due to the inflow velocity u_∞ , aerodynamic lift L , drag D , and a pitching moment M are produced at the aerodynamic

reference point. The aerodynamic loads cause elastic deformation about the beam reference point, changing the position and orientation of the airfoil section. The elastic pitch, along with the rates of elastic plunge, lag, and pitch alter the aerodynamic loads, and formulate a closed-loop aeroelastic feedback system. The eccentricity $\bar{x}_a$ between the aerodynamic and beam reference points affects the moment transferred to the beam reference point. The eccentricity $\bar{x}_{CG}$ between the centre of mass and the beam reference point produces coupling between the pitch and flap degrees of freedom. Following equation (4.14), provisions are included to consider the mass and beam reference point eccentricity.

Figure 4.3 A schematic diagram of the 2D basis for aeroelastic coupling of the aerodynamic and structural components.

By extending the two-dimensional aeroelastic system to three dimensions, a framework is developed for coupling the aerodynamic and structural components, as shown schematically in Figure 4.4. The structural and aerodynamic components retain separate representations of the blade. The structural component uses a one-dimensional beam element discretization of the beam reference axis to model the blade structure, typically from the hub attachment point to the blade tip. The aerodynamic component models the three-dimensional lifting surface of the blade as a two-dimensional surface of panel elements.

Each section of the three dimensional lifting surface is similar to the problem shown in Figure 4.3. The effective section loading and elasticity are provided by the aerodynamic and structural components, respectively. The aerodynamic component solves for the potential loading L at every spanwise station. Using L , an effective section angle of attack is calculated. Next, using airfoil lookup tables the section drag D is obtained to correct for viscous effects, which cannot be captured by the potential loading. The pitching moment coefficient is also obtained from the airfoil lookup tables to avoid the need for a very fine discretization near the airfoil leading edge, to decrease the computational cost. The airfoil tables are typically indexed by both angle of attack and section Mach number, allowing for compressibility corrections. The spanwise loading distribution is applied by the structural component to the beam reference axis to solve for the aeroelastic response of the beam (following equation (4.13)). The elastic deformation of the blade is then used to deform the aerodynamic mesh of the lifting surface; an exaggerated torsional deformation about the beam reference axis is shown in Figure 4.4.

Although the structural component has six degrees of freedom at each spanwise node (three displacements and three rotations), only the plunge (i.e. flap), lag, extension, and pitch degrees of freedom are used for the aeroelastic coupling.

The rate of elastic deformation of the blade provides an additional body velocity component that is accounted for in the non-penetration condition of the aerodynamic formulation – the term F in equations (2.4), (2.9), and (2.10). In the case of an articulated blade, the rotations at the hub attachment node create an additional rigid body velocity component that is applied to the aerodynamic non-penetration boundary condition in similar fashion as the elastic velocity. The elastic and rigid velocities alter the system aerodynamic loads, which adds damping to the aeroelastic system.

Figure 4.4 Schematic diagram of the aeroelastic coupling method for SMARTROTOR.

4.3 SMARTROTOR Code Structure

SMARTROTOR can be classified as a closely-coupled aeroelastic code. Closely coupled aeroelastic codes solve separate aerodynamic and structural systems and

exchange aeroelastic coupling data at every timestep. Coupling of the aerodynamic and structural components in SMARTROTOR is achieved by creating interface subroutines to communicate aeroelastic data between the codes. A main program is then used to alternately call the aerodynamic and structural components, and call the interface subroutines for exchange of data.

Two main interfaces were required for SMARTROTOR. The first to communicate loading data from the aerodynamic solution to the structural component, and the second to communicate rigid and elastic deformations from the structural solution to the aerodynamic component. The first interface (aerodynamic to structural) uses two subroutines: THWEAOA computes a spanwise loading distribution with respect to the beam reference axis and FORCE_NODE converts the loading distribution into equivalent discrete nodal forces. The second interface (structural to aerodynamic) uses four subroutines: TRAN extracts rigid body rotations and elastic deformations from the structural solution, BLADE_DEFORM applies the elastic deformations to the aerodynamic mesh about the beam reference axis, and ELAST_BC and ARTICULATE calculate the elastic deformation and rigid body rotation rates, respectively, for the application of the non-penetration boundary condition. The global rotating a frame is used as the reference frame for exchange of data for each blade. Both codes use base metric units for calculations and exchange angular data in radians.

Figure 4.5 shows an overview of the scheme for the close coupling of the components. Inputs such as the rotor geometry, flight conditions, and active control are specified using input files. Initialization subroutines use the input data to define the problem in each

component. The main program calls the aerodynamic and structural components alternately, once per timestep. Aeroelastic data is communicated between the components at each timestep. The simulation is run until the results become periodic. At the end of the simulation, results such as the flowfield around the rotor, aerodynamic loading, structural deformation, the acoustic field, and hub vibration are obtained. All output results include the effects of aeroelastic coupling.

Figure 4.5 An overview block diagram of the coupling of components to form SMARTROTOR.

The aerodynamic and structural meshes are spanwise coincident over the length of the blade lifting surface. That is, for every structural beam element there is a corresponding aerodynamic strip of panel elements over the same span length. This approach was taken to avoid the need for data interpolation at different spanwise stations during exchange of

aeroelastic data. Both components use the same timestep size during the simulation, as opposed to having an integer ratio of timestep sizes. In future investigation, it may be possible to optimize the code in terms of computational cost and accuracy by investigating the present mesh and timestep constraints.

4.4 Initial Aeroelastic Validation of SMARTROTOR

Test cases are run to validate the aeroelastic coupling scheme used in SMARTROTOR. The approach taken is to validate the aeroelastic coupling scheme against simple approximate solutions to confirm that the SMARTROTOR results are consistent with basic aeroelastic theory. It should be noted that the approximate solutions used for comparison are far from exact analytical solutions, but rather show basic trends that should appear in the SMARTROTOR results. SMARTROTOR is intended to capture effects that cannot be predicted by these numerical approximations. Similar to the approach taken by Sikorsky [19], the best approach for validating a code such as SMARTROTOR is to perform numerous analyses such as that given in Chapter 5. Although simple, the validation presented in this section is a respectable benchmark to confirm the ability of SMARTROTOR to capture some of the basic physics required to analyze the more complex problem presented in Chapter 5.

A test case is modelled that considers the aeroelastic deformation of a straight slender cantilevered non-rotating blade. The SMARTROTOR model is based on the geometry and material properties of the lifting surface of the BO105 model scale rotor blades used in the HELINOISE experiment [4]. The material properties of the model lifting surface are assumed to be constant over the span. Pretwist of the lifting surface is not considered.

The blade is modelled at a constant angle of attack. The aeroelastic deformation is evaluated by running simulations at different airspeeds. SMARTROTOR results for vertical tip displacement and tip twist are compared with approximate solutions.

Dowell [78] provides an approximate solution for the torsional aeroelastic twist, based on the classic linear beam torsion theory. A lift distribution, considering aeroelastic twist and finite wing effects, is next used to determine the approximate blade vertical tip deflection, given by the classic linear beam theory.

Figure 4.6 A comparison of tip deflection results for a non-rotating cantilevered blade.

Figure 4.6 shows a comparison of SMARTROTOR results with approximate solutions for tip twist and tip deflections. In both comparisons, SMARTROTOR results follow the trend provided by the approximate solution. The tip twist magnitude increases quadratically with airspeed, and it is negative because the beam reference axis is located in front of the aerodynamic centre. The tip deflection also increases quadratically as a function of airspeed. The difference seen at the higher airspeeds is mainly due to the limitations in the linear beam theory, preventing the capture of the large deformations expected at higher airspeeds. In addition, the tip deflection predictions show some additional difference due to the estimated finite spanwise loading distribution assumption used in the approximate solution.

A similar test case, which considers the aeroelastic deformation of a straight slender cantilevered blade with geometric stiffening, is investigated. The aerodynamic model is identical to the previous test case, with a constant airspeed of 50m/s. In this case, the aerodynamic model simulates a non-rotating blade. The structural model is similar to the previous model, with the exception of the inclusion of a specified rotational velocity to produce geometric stiffening of the blade. Simulations are run with different rotation speeds to identify the effect of geometric stiffening on the blade aeroelastic deformations. In this case, equation 9.75 of Bramwell [12] is solved to obtain the aeroelastic twist of the blade, considering the “propeller moment” due the blade rotation. Similar to the previous test case, the twist distribution from the approximate solution is used to obtain a spanwise lift distribution and determine the tip deflection. To approximate the effect of geometric stiffening, the non-rotating blade stiffness is increased by a factor given by the square of

the ratio of the non-rotating and rotating first-mode natural frequencies, obtained from reference [4].

Figure 4.7 A comparison of tip deformation results for a rotating cantilevered blade.

Figure 4.7 provides a comparison of SMARTROTOR results with the results from the approximate solutions. The predictions differ at the lower rotational speeds, but present similar magnitude and trend over the majority of the comparison range. The differences visible at the lower rotational speeds are due to the large deformations, which are not well predicted by the linear beam theory.

Chapter 5 BO105 Model Scale Rotor Analysis

In the early 1990's, several members of the European rotorcraft research community began an effort to collaboratively research rotorcraft aerodynamics and aeroacoustics. One of the programs that resulted from this collaboration is the HELINOISE program [4]. The goal of the HELINOISE program was “to further the understanding – both theoretically and experimentally – of the physics of helicopter rotor noise”. Although the program also included theoretical and computational research, the major component of HELINOISE was a wind-tunnel experiment with a geometrically and dynamically scaled helicopter rotor. The HELINOISE experiment resulted in a large database of experimental results for helicopter aeroacoustics, aerodynamics, blade structural dynamics, and performance.

This chapter presents validation results from SMARTROTOR using the HELINOISE experiment as a basis. The results reflect the initial validation of SMARTROTOR for use with hingeless rotor configurations. The results presented in this chapter provide a demonstration of the potential of SMARTROTOR for future rotorcraft computational studies.

5.1 HELINOISE Overview

For use with the HELINOISE program, a scaled model of the main rotor of an ECD (formerly MBB) BO105 four-bladed hingeless helicopter was constructed. A summary of rotor data for the BO105 model is presented in Table 5.1. The test rotor was scaled forty-percent geometrically, in addition to Mach number and dynamic scaling. An acoustically lined scale model of the fuselage enclosed the rotor test rig. One of the test rotor blades was extensively instrumented with pressure transducers to measure aerodynamic loads, and with strain gauges to measure the blade flapping, lead-lagging, and torsional deformation.

Scaling factor	2.455
Number of blades	4
Rotor type	Hingeless
Airfoil type	NACA 23012 modified
Rotor radius	2 m
Blade chord	0.121m
Pretwist (outboard of $r = 0.44\text{m}$)	$-4^\circ/\text{m}$
Nominal rotor speed, Ω	1050 RPM (110 rad/s)

Table 5.1 A summary of data for the BO105 model rotor.

The HELINOISE test was conducted in the German-Dutch Wind Tunnel (DNW). The DNW is a relatively large subsonic, atmospheric, anechoic wind tunnel used extensively for aeroacoustic testing. A transversing microphone array was used during the

experiment to measure the rotor acoustic signature on a plane below the rotor. In addition, a laser light sheet technique was used for flow visualization, primarily to capture BVI events.

The HELINOISE test plan was chosen to simulate tests previously conducted with a full-scale helicopter. The executed test plan included hover, level flight, climb and descent conditions, over a range of advance ratios. Particular attention was paid to descent flight conditions to capture BVI event test data.

5.2 SMARTROTOR BO105 Model Input

Although effort was made to model the HELINOISE test set-up as closely as possible, certain approximations were necessary due to lack of availability of detailed blade structural data and for simplification purposes. Only the lifting surfaces of the blades are modelled with both the structural and aerodynamic components, with a cantilevered root boundary condition for the blade flapping and lead-lagging motion. This simplification means that the SMARTROTOR model does not include the effects of flexibility of the root cutout portion of the blade. However, since the root cutout is much stiffer than the lifting surface of the blade, this simplification is reasonable. The fuselage is not considered in the present model, which would have some affect on the aerodynamic results due to changes in the inflow, and on the acoustic results due to acoustic shielding of microphones. Finally, due to lack of availability of detailed blade construction information, some of the structural input data are estimated by comparison with similar aeroelastically scaled model rotors (reference [30] for example). Table 5.2 gives a

summary of the blade section inertial and elastic properties used for structural input. An asterisk denotes properties that are estimated using indirect data.

m	0.948 kg/m
I_{11}	$6.23 \times 10^{-4} \text{ kg-m}^2/\text{m}$
I_{22}^*	$8.89 \times 10^{-6} \text{ kg-m}^2/\text{m}$
I_{33}^*	$6.10 \times 10^{-4} \text{ kg-m}^2/\text{m}$
EA^*	$1.64 \times 10^6 \text{ N}$
GJ	$2.50 \times 10^2 \text{ N-m}^2$
EI_{FLAP}	$1.90 \times 10^2 \text{ N-m}^2$
EI_{LAG}	$4.71 \times 10^3 \text{ N-m}^2$
<p><i>* - Parameter estimated by comparison with similar aeroelastically scaled model rotors.</i></p> <p><i>NOTE: The beam reference axis is at 21% chord.</i></p>	

Table 5.2 A summary of inertial and elastic input properties for the SMARTROTOR BO105 model.

The present BO105 analysis includes the modelling of four HELINOISE test cases, which are given in Table 5.3. Each test case is run for a total simulation length of approximately eighteen rotor revolutions. Over the first three revolutions, the loads transferred to the structural component are linearly increased to minimize structural transient oscillation due to the impulsive starting of the rotor. To allow this initial structural transient response to decay, ten revolutions are used from the time of full load application to the start of particle wake activation. The particle wake is activated for the

final five revolutions. The particle wake cannot be activated for the entire simulation due to prohibitive computation cost. The results typically become periodic prior to the final three revolutions.

HELINOISE TEST CASE NUMBER	DESCRIPTION
947	Near hover
344	Low-speed level flight (0.15 advance ratio)
508	Low-speed climb (0.15 advance ratio, 12° flight path angle)
1333	Low-speed descent (0.15 advance ratio, -6° flight path angle)

Table 5.3 A list of HELINOISE test cases modelled with SMARTROTOR.

5.3 Analysis Results

Results from the SMARTROTOR analysis of the HELINOISE experiment are presented in the current section. Indicative results and comparisons are presented in this section; additional results from the analysis are included in Appendix B.

5.3.1 Aerodynamic Loading Results

The thrust of the rotor is determined by adding the aerodynamic loads, in the direction along to the rotor axis, of all four blades. Figure 5.1 shows the predicted rotor thrust

history for case 1333. In comparison with the other cases, shown in Appendix B, case 1333 has the highest amplitude of oscillation of the rotor thrust, while case 947 has the lowest amplitude. This result is as expected since case 1333 is a descent case, which is notorious for high rotor vibration due to the blades descending into the wake shed by previous rotor blades. Case 947 is close to a hover flight condition; therefore the thrust is close to steady state, since the rotor wake descends below the disc, and the blade pitch input has little cyclic component. Similarly, case 508 has the lowest oscillation amplitude of the forward flight cases since the rotor is in a simulated climb, which causes the wake to pass below the rotor blades.

**Figure 5.1 A graph of the SMARTROTOR predicted rotor thrust history;
HELINOISE case 1333.**

Using the mean value of thrust over one revolution, the non-dimensional thrust coefficient is determined and compared with experimental results. The comparison is presented in Table 5.4. Similar to the experimental results, the SMARTROTOR results

predict an approximately constant thrust coefficient for each simulated flight condition. However, the SMARTROTOR results are lower than the experimental results by a consistent margin. The most probable cause of this margin is a difference between the simulated disc tilt angle and the actual disc tilt angle. As discussed in the experimental report [4], when a rotor is operated in an open wind tunnel, as it was in HELINOISE, the incident flow is deflected more than would have happened in a free air situation, simulated in SMARTROTOR.

CASE #	C_T – SMARTROTOR	C_T – EXPERIMENTAL	% DIFF.
1333	0.00380	0.00448	-15.2
344	0.00377	0.00446	-15.5
508	0.00369	0.00454	-18.7
947	0.00375	0.00436	-14.0

Table 5.4 A comparison of thrust coefficients between SMARTROTOR predictions and HELISNOISE experimental results.

The normal force coefficient is a non-dimensional measure of the loading on a radial section of the blade, normal to the section chord. The normal force coefficient can be used to detect blade-vortex interaction (BVI) encounters. Referring to Figure 1.1, BVI is expected in the first quadrant of the azimuthal cycle. Although not depicted in Figure 1.1, BVI also occurs in the last quadrant of the azimuthal cycle. Figure 5.2 shows the predicted normal force coefficient history at the outer portion of the blade, for one revolution, for HELINOISE case 1333; Figure 5.3 shows the experimental results for the same case. SMARTROTOR accurately predicts the overall trend and magnitude of the

normal force coefficient history. In addition, the BVI events on the advancing and retreating sides are captured. The magnitude of the predicted BVI on the retreating side is larger than the experimental measurement. This is a point for further investigation in SMARTROTOR. Additional normal force coefficient results are given in Appendix B. These results follow expected trends for BVI occurrence. Case 344 (level flight) predicts some BVI, but with lower magnitude than the descent case shown in Figure 5.2. Relatively little BVI is predicted in cases 508 (climb) and 947 (near hover).

**Figure 5.2 The normal force coefficient history predicted by SMARTROTOR;
HELINOISE case 1333.**

Figure 5.3 Measured normal force history; HELINOISE case 1333 [4].

5.3.2 Hub Vibration Results

The vibratory load transferred to the hub can be calculated by adding the structural forces at the root boundary of the rotor blades. This gives an approximate measure of the vibration that the rotor would transfer to the fuselage. This result is approximate only because the inertia of the hub is not considered. However, it is a useful measure for vibration control study purposes.

The results of the SMARTROTOR analysis are evaluated as a comparison between different flight conditions. Figure 5.4 shows a comparison of predicted vertical vibratory load amplitude at harmonics of the rotor frequency. Chapter 10 of Bramwell [12] indicates how constant “perceived” levels of vibration have the form of vibration amplitude decreasing asymptotically as a function of frequency. In other words, the same

amplitude of vibration would be perceived as worse at a higher frequency. Considering this, relative to the other cases case 1333 (descent) has the highest predicted vibratory load, with large oscillation amplitudes at $3p$, $4p$, and $8p$. Case 344 (level flight) is also predicted to have relatively high vibration, with large oscillation amplitudes at $4p$ and $8p$. These results are consistent with the aerodynamic results presented in section 5.3.1, and primarily reflect the structural response of the blades to the aerodynamic forcing due to interaction with the wake.

Figure 5.4 A comparison of the SMARTROTOR predicted vertical vibratory loads transferred to the hub for different HELINOISE test cases.

5.3.3 Aeroelastic Deformation Results

Figure 5.5 is a visualization of the undeformed and deformed aerodynamic mesh of the BO105 model for HELINOISE case 947. The deformation follows what would be expected for a hingeless rotor: zero deformation at the root and an upwards flap deformation increasing non-linearly in the outward radial direction.

Figure 5.5 A visualization of the undeformed (grey) and deformed (translucent black mesh) aerodynamic mesh of the BO105 rotor for HELINOISE case 947.

Tip deflection histories for HELINOISE case 1333 are presented in Figures 5.6, 5.7, and 5.8, with respect to the $a1$, $a2$, and $a3$ axis, respectively, of the global rotating

coordinate system. The tip torsional rotation history about the global rotating $a1$ axis is presented in Figure 5.9. Results for cases 344, 508, and 947 are given in Appendix B.

Reference [4] does not provide detailed data for aeroelastic deflection of the rotor during testing. Consequently, consideration of the current aeroelastic deflection results is limited to evaluation of magnitudes and trends visible in the results. Figure 5.6 shows a predicted mean extensional deformation of the blade, which is expected due to centrifugal loading. The oscillation shown is primarily due to the projection, along the $a1$ axis, of the flapping and lead-lagging motion of the blade.

Figure 5.7 shows a predicted 1/rev lead-lag motion of the blade tip. The mean value of the lead-lag motion is negative, indicating a mean lag position, which is expected due to the drag loads on the blade. Although experimental data is not available to verify the lead-lag result, it is suspected that the amplitude of the motion may be larger than would actually occur. The probable explanation for the increased amplitude is the lack of modelling of structural damping. In general, helicopter rotors are susceptible to excessive lead-lag motion, hence the use of lead-lag dampers or other means to reduce the amplitude of oscillation. Therefore, it is plausible that the SMARTROTOR prediction, while accurate for the specified input data, does not match the experiment. Further modelling of hingeless rotors with SMARTROTOR should investigate the lead-lag characteristics of the blade structural model to ensure a match with actual values.

Figure 5.8 shows the predicted flap deflection history of the blade tip. The mean value of deflection indicates an upward flap deformation, which is expected due to the thrust loading on the blade. Figure 5.9 shows the predicted torsional rotation history of the

blade tip. The mean value of the rotation is negative, indicating an expected nose down rotation since the beam reference axis (21% chord from the leading edge) is ahead of the aerodynamic centre (25% chord from the leading edge). A higher frequency response is visible, which is feasible since the blade rotating first torsional mode frequency is approximately 5/rev [4].

Figure 5.6 SMARTROTOR predicted blade tip extension history for HELINOISE case 1333.

Figure 5.7 SMARTROTOR predicted blade tip lead-lag history for HELINOISE case 1333.

Figure 5.8 SMARTROTOR predicted blade tip flap history for HELINOISE case 1333.

**Figure 5.9 SMARTROTOR predicted blade tip torsional rotation history for
HELINOISE case 1333.**

5.3.4 Aeroacoustic Results

During the HELINOISE experiment, using a transversing microphone array, acoustic signals were measured on a plane approximately 1.2 rotor radii below the rotor hub. A diagram of the measurement plane relative to the model rotor is shown in Figure 5.10. Collecting data on such a grid allows a comprehensive evaluation of the intensity and directivity of noise produced by the rotor. An acoustic grid that matches the points that were used in the HELINOISE experiment was specified for the SMARTROTOR BO105 model of test case 1333.

Figure 5.10 A diagram of the HELINOISE acoustic measurement plane relative to the model rotor [4].

Figure 5.11 shows the predicted acoustic signal with the transversing microphone array 2m downstream of the rotor hub. Microphone 1 is on the retreating side, with the microphone numbers increasing to microphone 11, which is on the advancing side. For comparison, the experimentally measured results are shown in Figure 5.12. Both results have a periodic signal that repeats four times per revolution, corresponding to the blade passage frequency. The predicted acoustic signals are similar to the measured acoustic signals in terms of waveform and magnitude. The similarity along the span of the microphone array indicates the capture, by SMARTROTOR, of the directivity of the rotor noise. A primary difference between the SMARTROTOR and measured results is that the impulsive pressure peaks, of the microphones on the retreating side, are not consistently predicted. This is likely a truncation error, which can be resolved by using a

smaller timestep size to capture the impulsive nature of the signal. Additional results, at other streamwise positions of the microphone array, are presented in Appendix B.

Figure 5.11 SMARTROTOR predicted acoustic signals at streamwise position +2.0m, HELINOISE case 1333.

Figure 5.12 Measured acoustic signals at streamwise position +2.0m, HELINOISE case 1333.

The HELINOISE experimental acoustic measurements were further reduced by computing a single acoustic measure – called a frequency summary level – for each microphone location [79]. Using a Fast Fourier Transform, an acoustic pressure time history, such as that shown in Figure 5.11, is converted into a narrowband frequency versus magnitude spectrum. The magnitude is expressed on a decibel scale, referred to as the sound pressure level, such as shown in Figure 3.11. This spectrum has distinct tones at harmonics of the blade passage frequency. Frequency summary levels are obtained by adding the sound pressure levels of harmonics within a selected frequency range. The low frequency summary level (LFSL) includes harmonics up to the 10th harmonic of the blade passage frequency, and is an indicative measure of thickness and impulsive noise [79]. The mid frequency summary level (MFSL) includes the 6th to the 40th harmonics of blade passage frequency, and is an indicative measure of BVI noise [79]. By computing frequency summary level data on a grid such as that used in the HELINOISE experiment, contour plots can be produced that give an indication of noise intensity and directivity.

From SMARTROTOR predicted sound pressure level versus frequency data, similar to Figure 3.11, a LFSL value is calculated for each microphone as follows:

$$LFSL = 10 \cdot \log \left(\sum_i 10^{SPL_i/10} \right) \quad (5.1)$$

where SPL_i is the maximum sound pressure level within a 2 Hz band centred about the i -th harmonic of the blade passage frequency, and summation taken up to the tenth harmonic.

Figure 5.13 shows a LFSL contour plot for HELINOISE case 1333, produced using acoustic pressure result predictions from SMARTROTOR. A projection of the rotor

circumference is shown with a black circle. Figure 5.14 is a LFSL plot for the same case, produced with measured results. The SMARTROTOR predictions display similar directivity and intensity trends as the measured results. The SMARTROTOR results predict a similar intensity range over the extent of the comparison range, but are slightly higher. A region of high noise intensity is correctly predicted downstream of the hub, within the projected rotor area. An intensity gradient is visible just upstream of the rotor hub, slanted in the crossflow direction. The predicted gradient is more slanted than the measured gradient. Different from the measured results, peaks are predicted on the retreating side, downstream of the hub. These peaks are consistent with the over-prediction of BVI intensity on the retreating side, discussed in section 5.3.1. Figure 5.15 is a predicted frequency summary level plot for the 1st to the 5th blade passage harmonics. The over prediction of BVI on the retreating side is no longer visible. The intensity trends and overall intensity range compares more favourably with Figure 5.14. Although the predicted LFSL plot is respectable, it can be improved by optimizing the mesh discretization and simulation timestep size, in order to capture the acoustic signals with better resolution.

HELINOISE CASE 1333 LOW-FREQ. SUMMARY LEVEL

Figure 5.13 The SMARTROTOR predicted low-frequency summary level (LFSL) plot for HELINOISE case 1333; (up to the 10th blade passage harmonic).

Figure 5.14 The measured low-frequency summary level (LFSL) plot for
HELINOISE case 1333 [4] ; (up to the 10th blade passage harmonic).

HELINOISE CASE 1333 LOW-FREQ. SUMMARY LEVEL

Figure 5.15 The SMARTROTOR predicted frequency summary level plot (up to the 5th blade passage harmonic only) for HELINOISE case 1333.

Chapter 6 Summary, Conclusions, and Recommendations

6.1 Summary

Rotorcraft are subject to many noise and vibration problems. These noise and vibration problems are often the result of the unsteady aerodynamic environment that the rotor creates and the aeroelastic interaction between the rotor blades. In pursuit of means for noise and vibration alleviation, comprehensive rotorcraft aeromechanical codes are required. The current work developed a code called SMARTROTOR, a general active aeroelastic aeroacoustic rotorcraft analysis code.

GENUVP, a code developed by the National Technical University of Athens, was used as the aerodynamic component for SMARTROTOR. GENUVP is an aerodynamic code for prediction of the flow around multi-component configurations. Through the combination of an unsteady panel method and a vortex particle wake, GENUVP is able to provide predictions of the aerodynamic loads on helicopter blades, the theory of which was described in the current work. Subgrid and particle-mesh approximations were

implemented in GENUVP to reduce its computational cost. A variety of GENUVP validation test cases were presented in the current work.

A new acoustic component was developed in the current work for inclusion in SMARTROTOR. The acoustic component was based on the Farassat 1A solution, for thickness and loading noise, of the Ffowcs Williams-Hawkings equation. Farassat's 1A solution was implemented into a code, which provides acoustic predictions based on aerodynamic results from GENUVP. Validation test cases were presented for a model scale rotor in hover and forward flight.

A existing structural component based on a geometrically exact formulation for the dynamics of rotating beams, with inclusion of the effects of integral twist actuation, was added to GENUVP. Extending the classic two-dimensional aeroelastic section problem to three dimensions, a method was developed for coupling GENUVP with the structural component, to form SMARTROTOR. The coupling method was applied to obtain a closely-coupled aeroelastic code. Validation test cases were presented for fixed and rotating blades.

Further validation of SMARTROTOR was accomplished by comparison with the HELINOISE experiment BO105 model scale rotor. Aerodynamic loading, hub vibration, aerodynamic deflection, and acoustic predictions were presented.

6.2 Conclusions

By comparison with several test cases, ranging from simple two-dimensional steady and unsteady lift slope curves to a fully articulated rotor in forward flight, GENUVP was shown to be an accurate and computationally efficient means of obtaining rotorcraft

aerodynamic loading predictions. The results presented in Chapter 2 provided additional validation of GENUVP.

Initial validation of a new acoustic component, developed in the current work, was completed. The acoustic component is valid for predicting thickness and loading acoustic pressure histories, for rotors in hover and forward flight, with accurate capture of intensity and directivity. The acoustic component predictions were validated by comparison with computational and experiment data, in the time and frequency domain.

Initial validation of the aeroelastic coupling method that was used to combine GENUVP with the new structural component was completed. The resulting code, SMARTROTOR, is a closely-coupled rotorcraft aeroelastic code. Validation test cases, for fixed and rotating blades, showed that SMARTROTOR captures the basic aeroelastic deformation trends given by approximate aeroelastic solutions.

Finally, validation of SMARTROTOR against a set of rotorcraft experimental data – the HELINOISE BO105 test – was presented in Chapter 5. Four flight scenarios were successfully investigated. The SMARTROTOR aerodynamic predictions compared well with experimental results. The hub vibration and aeroelastic deformation results were respectable, and agreed with expectations for the simulated flight scenarios. The aeroacoustic predictions produced magnitudes and trends that were similar to the experimental results. A respectable capability for prediction of the acoustic signature of a rotorcraft was demonstrated. The complete set of results in Chapter 5 demonstrated the potential of the current version of SMARTROTOR for aerodynamic, structural dynamic, and aeroacoustic analyses of hingeless rotors.

6.3 Recommendations

Short-term continuation of the current work could focus on further validation of SMARTROTOR with HELINOISE experimental data. The HELINOISE experiment has a comprehensive set of data, which could be compared with results produced by SMARTROTOR. An example would be a comparison with the data measured by strain gauges. Such work would provide additional insight into modelling techniques with SMARTROTOR, and the capability of the code for hingeless rotorcraft predictions. In addition, parametric studies could be concurrently run for methods to optimize SMARTROTOR. Timestep ratios between the components, and non-spanwise coincident meshes are suggested areas for investigation.

Concurrently with further investigation of the BO105 rotor, the investigation could be expanded to an evaluation of the effect of active twist rotor blades on a helicopter's acoustic signature. The current version of SMARTROTOR is provisioned for modelling active twist rotors. A simulated active twist rotor, based on the BO105, could be investigated to provide predictions of the control authority of active twist rotor blades on rotor acoustics.

An important consideration for further development of SMARTROTOR is further testing and characterization of the structural component. Further independent testing of the structural component is recommended, along with optimization of the source code structure. Presently, the numerical stability characteristics of the structural code are not well defined. Proper definition of the stability of the structural code is essential for

effective modelling. In addition, implementing a numerical differentiation method with controllable numerical damping may be considered.

Next, a validation should be completed which investigates a fully articulated rotor. The current version of SMARTROTOR has provisions for modelling articulated rotors. The code changes necessary to model an articulated rotor are revisions to the structural component method of separating of rigid and elastic feedback data and the lead-lag damper model. A candidate rotor for comparison is the NASA/ARMY/MIT Active Twist Rotor (ATR) [30].

Further development of the version of GENUVP used by SMARTROTOR should first investigate the inclusion of the particle mesh approximations discussed in section 2.1.4. The inclusion of particle mesh approximations would greatly reduce the computation cost, and be a tremendous aid during development of models. Subsequent development could investigate the inclusion of embedded compressibility domains, which are currently being investigated by NTUA [47] [48].

References

- [1] American Helicopter Society (AHS), "Helicopter Fact Sheet", *AHS Homepage*, <http://www.vtol.org/helifact.html> (current June 2002).
- [2] A. Concilio, A. Sorrentino, and F. Cenedese, "Measurement of the Human Perception of Noise and Vibration Aimed to Define a General Comfort Index", *Proc. 25th European Rotorcraft Forum*, Rome, Italy, Sept. 14-16, 1999.
- [3] G.R. Barnes, "Human Factors of Helicopter Vibration 1: The Physiological Effects of Vibration", *Proc. Royal Aeronautical Society Conference on Helicopter Vibration and its Reduction*, London, England, Nov. 16, 1987, pp. 20-30.
- [4] W.R. Splettstoesser et. al., "The HELINOISE Aeroacoustic Rotor Test in the DNW – Test Documentation and Representative Results", DLR-Mitt. 93-09, DLR, Braunschweig, Germany, 1993.
- [5] Y. Yu et al., "Aerodynamic and Acoustic Test of a United Technologies Model Scale Rotor at DNW", *Proc. 46th AHS Annual Forum*, Washington, DC, May 1990, pp. 1233-1250.
- [6] C. Tung, W.G. Bousman, and S. Low, "A Comparison of Airload Data Between Model-Scale and Full-Scale Flight Test", *Proc. AHS 2nd Int'l. Aeromechanics Specialists Conference*, Bridgeport, CT, Oct. 11-13, 1995, pp. 1-19.

- [7] R. Kube et al., "Initial Results from the Higher Harmonic Control Aeroacoustic Rotor Test (HART) in the German-Dutch Wind Tunnel", *Proc. AGARD FDP Symp. on Aerodynamics and Aeroacoustics of Rotorcraft*, CP-552, Berlin, Germany, Oct. 10-13, 1994.
- [8] S.M. Swanson et al., "Effect of Individual Blade Control on Noise Radiation", *Proc. AGARD FDP Symp. on Aerodynamics and Aeroacoustics of Rotorcraft*, CP-552, Berlin, Germany, October 10-13, 1994.
- [9] W.G. Bousman, "The Response of Helicopter Rotors to Vibratory Airload", *Proc. AHS National Specialists Meeting on Rotorcraft Dynamics*, Arlington, Texas, Nov. 13-14, 1989, pp. 1-20.
- [10] C. Kitaplioglu and F.X. Caradonna, "A Study of Blade-Vortex Interaction Aeroacoustics Utilizing an Independently Generated Vortex", *Proc. AGARD FDP Symp. on Aerodynamics and Aeroacoustics of Rotorcraft*, CP-552, Berlin, Germany, October 10-13, 1994.
- [11] P.C. Tarttelin and Lt. Cdr. A.W. Martyn RN, "In-flight Research with Instrumented Main and Tail Rotor Blades using the DRA Bedford Aeromechanics Research Lynx Helicopter", *Proc. AGARD FDP Symp. on Aerodynamics and Aeroacoustics of Rotorcraft*, CP-552, Berlin, Germany, October 10-13, 1994.
- [12] A.R.S. Bramwell, *Helicopter Dynamics*, Edward Arnold (Publishers) Ltd., London, 1976.
- [13] R.L. Bielawa, *Rotary Wing Structural Dynamics and Aeroelasticity*, American Institute of Aeronautics and Astronautics, 1992.

- [14] D.H. Hodges and E.H. Dowell, "Non-linear Equations of Motion for the Elastic Bending and Torsion of Twisted Non-uniform Rotor Blades", TN D-7818, NASA Ames Research Centre, Moffett Field, CA, 1974.
- [15] D. H. Hodges, "A Mixed Variational Formulation based on Exact Intrinsic Equations for Dynamics of Moving Beams", *Int'l Journal of Solids and Structures*, vol. 26, no. 11, 1990, pp. 1253-1273.
- [16] J.E. Ffowcs-Williams and D.L. Hawkings, "Sound Generation by Turbulence and Surfaces in Arbitrary Motion", *Philosophical Transactions of the Royal Society*, A264, 1969.
- [17] F. Farassat, "The Evolution of Methods for Noise Prediction of High Speed Rotors and Propellers in the Time Domain", *Recent Advances in Aeroacoustics*, Springer-Verlag, New York, New York, 1986, pp. 129-147.
- [18] K.S. Brentner and F. Farassat, "An Analytical Comparison of the Acoustic Analogy and Kirchhoff Formulation for Moving Surfaces", *AIAA Journal*, Vol. 36, No. 8, Aug. 1998, pp. 1379-1386.
- [19] J.M. Wang, "Recent Rotor Dynamics Predictions at Sikorsky Aircraft", *Proc. 24th European Rotorcraft Forum*, Marseilles, France, September 15-17, 1998.
- [20] W.K. Wilkie, *Anisotropic Piezoelectric Twist Actuation of Helicopter Rotor Blades: Aeroelastic Analysis and Design Optimization*. PhD dissertation, Dept. of Aerospace Engineering Sciences, University of Colorado, Boulder, Colorado, 1997.

- [21] M. Ikuta and T. Kobayashi, "Vibration Reduction Designing for MH2000 Development", *Proc. AHS 57th Annual Forum*, Washington, DC, May 9-11, 2001, pp. 226-300.
- [22] T. Millott and W. Welsh, "Helicopter Active Noise and Vibration Reduction", *Proc. 25th European Rotorcraft Forum*, Rome, Italy, Sept. 14-16, 1999.
- [23] S.J. Elliott, C.C. Boucher, and T.J. Sutton, "Active Control of Rotorcraft Interior Noise", *Proc. of the Royal Aeronautical Society Conference on Innovation in Rotorcraft Technology*, London, England, June 24-25, 1997.
- [24] F. Nitzsche, "Tailoring the Dynamic Response of Helicopter Rotors Using Smart Materials", *Proc. AIAA/ASME Adaptive Structures Forum*, New Orleans, LA, April 12-13, 1995, AIAA 95-1083.
- [25] F. Nitzsche, R. Lammering, and E. Breitbach, "Can Smart Material Modify Blade Root Boundary Conditions to Attenuate Helicopter Vibration?", *Proc. 4th International Conference on Adaptive Structures*, Cologne, Germany, Nov. 2-4, 1993, pp. 139-150.
- [26] D. Zimcik et al., "'Smart Spring' Concept for Active Vibration Control in Helicopters", *Proc. American Helicopter Society 58th Annual Forum*, Montreal, Canada, June 11-13, 2002.
- [27] N.A. Koratkar, and I. Chopra, "Wind Tunnel Testing of a Mach-Scaled Rotor Model with Trailing-Edge Flaps", *Smart Materials and Structures*, vol. 10, no. 1, Feb. 2001, pp1-14.

- [28] F.K. Straub et al., “Development of a Piezoelectric Actuator for Trailing Edge Flap Control of Full Scale Rotor Blades”, *Smart Materials and Structures*, vol. 10, no. 1, Feb. 2001, pp. 25-34.
- [29] G. Depailler and P.P. Friedmann, “Reduction of Vibrations Due to Dynamic Stall in Helicopters using an Actively Controlled Flap”, *Proc. 43rd Structures, Structural Dynamics, and Materials Conference*, Denver, Colorado, April 22-25, 2002.
- [30] S. Shin, and C.E.S. Cesnik, “Helicopter Vibration Reduction in Forward Flight using Blade Integral Twist Control”, *Proc. 43rd Structures, Structural Dynamics, and Materials Conference*, Denver, Colorado, April 22-25, 2002.
- [31] I. Chopra, “Recent Progress on the Development of a Smart Rotor System”, *Proc. 26th European Rotorcraft Forum*, The Hague, Netherlands, Sept. 28-29, 2000.
- [32] P.C. Chen et al., “Blade-Vortex Interaction Noise Reduction with Active Twist Smart Rotor Technology”, *Smart Materials and Structures*, vol. 10, no. 1, Feb. 2001, pp. 77-85.
- [33] O.A. Bauchau, “DYMORE: User’s Manual and Theory Manual”, *DYMORE*, <http://www.ae.gatech.edu/~obauchau/dymore/manual/manual.html> (current June 2002).
- [34] O.A. Bauchau, C.L. Bottasso, and Y.G. Nikishkov, “Modeling Rotorcraft Dynamics with Finite Element Multibody Procedures”, *Mathematical and Computer Modeling*, vol. 33, 2001, pp. 1113-1137.

- [35] W. Johnson, "Rotorcraft Aeromechanics Applications of a Comprehensive Analysis", *Proc. Heli Japan 98: AHS International Meeting on Advanced Rotorcraft Technology and Disaster Relief*, Nagarafukumitsu, Gifu, Japan, April 21-23, 1998.
- [36] W. Johnson, "Rotorcraft Aerodynamics Models for a Comprehensive Analysis", *Proc. American Helicopter Society 54th Annual Forum*, Washington, DC, May 20-22, 1998.
- [37] D.A. Wachspress, T.R. Quackenbush, and A.H. Boschitsch, "Rotorcraft Interactional Aerodynamics Calculations with Fast Vortex / Fast Panel Methods", *Proc. American Helicopter Society 56th Annual Forum*, Virginia Beach, VA, May 2-4, 2000.
- [38] D.A. Wachspress and T.R. Quackenbush, "BVI Noise Prediction using a Comprehensive Rotorcraft Analysis", *Proc. American Helicopter Society 57th Annual Forum*, Washington, DC, May 9-11, 2001, pp. 1015-1032.
- [39] S. Shin, *Integral Twist Actuation of Helicopter Rotor Blades for Vibration Reduction*, PhD dissertation, Dept. of Aeronautics and Astronautics, Massachusetts Institute of Technology, Cambridge, MA, 2001.
- [40] R.C. Strawn and R.C. Djomehri, "Computational Modelling of Hovering Rotor and Wake Aerodynamics", *Proc. 57th AHS Annual Forum*, Washington, DC, May 9-11, 2001, pp. 1723-1734.

- [41] K. Pahlke and E. Chelli, "Calculation of Multibladed Rotors in Forward Flight using a 3D Navier-Stokes Method", *Proc. 26th European Rotorcraft Forum*, The Hague, Netherlands, Sept. 26-29, 2000.
- [42] A.J. Landgrebe, "New Directions in Rotorcraft Computational Aerodynamics Research in the U.S.", *Proc. AGARD FDP Symp. on Aerodynamics and Aeroacoustics of Rotorcraft*, CP-552, Berlin, Germany, Oct. 10-13, 1994.
- [43] J.G. Leishman, *Principles of Helicopter Aerodynamics*, Cambridge University Press, New York, NY, 2000.
- [44] S.G. Voutsinas, *A GENeralized Unsteady Vortex Particle Method for Solving the Unsteady Flow around Multi-component Configurations*, Fluids Section – Dept. of Mechanical Engineering, National Technical University of Athens, Greece, 1990.
- [45] S.G. Voutsinas, M.A. Belessis, and K.G. Rados, "Investigation of the Yawed Operation of Wind Turbines by means of a Vortex Particle Method", *Proc. FDP Symposium on Aerodynamics and Aeroacoustics of Rotorcraft (AGARD CP552)*, Berlin, Germany, 1995.
- [46] S.G. Voutsinas and D.G. Triantos, "High-resolution Aerodynamic Analysis of Full Helicopter Configurations", *Proc. 25th European Rotorcraft Forum*, Rome, Italy, Sept. 14-16, 1999.
- [47] S.G. Voutsinas and D.G. Triantos, "Aeroacoustics of Full Helicopter Configurations using Vortex Particle Flow Approximations", *Proc. CEAS Forum on Aeroacoustics of Rotors and Propellers*, Rome, Italy, June 9-11, 1999.

- [48] D.G. Triantos, *Aerodynamic and Aeroacoustic Analysis of Helicopter Rotors*, PhD dissertation, Fluids Section – Dept. of Mechanical Engineering, National Technical University of Athens, Greece, 2002.
- [49] J.L. Hess, *Calculation of Potential Flow about Arbitrary Three-Dimensional Lifting Bodies*, McDonnell Douglas tech. report MDCJ5679-01, 1972.
- [50] J.L. Hess and A.M.O. Smith, *Calculation of Non-Lifting Potential Flow about Arbitrary Three-Dimensional Bodies*, McDonnell Douglas tech. report ES40622, 1968. (Also *Progress in Aeronautical Sciences*, Vol. 8, 1967, pp. 1-138).
- [51] J. Katz and A. Plotkin, *Low-Speed Aerodynamics*, Cambridge University Press, New York, NY, 2001.
- [52] C. Rehbach, “Calcul d’écoulements autour dailes sans epaisseur avec nappes tourbillonnaires evolutives”, *Recherche Aerospatiale*, No. 2, 1973, pp. 53-61.
- [53] J.T. Beale and A. Majda, “Vortex Methods I: Convergence in Three Dimensions”, *Mathematics of Computation*, Vol. 39, No. 159, July 1982, pp. 1-27.
- [54] J.T. Beale and A. Majda, “High Order Accurate Vortex Methods with Explicit Velocity Kernels”, *Journal of Computational Physics*, Vol. 58, 1985, pp.188-208.
- [55] J.C. Vassberg, *A Fast Surface Panel Method Capable of Solving Million-element Problems*, AIAA paper 97-0168, 1997.
- [56] R.W. Hockney and J.W. Eastwood, *Computer Simulation Using Particles*, McGraw-Hill, New York, NY, 1981.

- [57] C. Anderson, "A Method of Local Corrections for Computing the Velocity due to a Distribution of Vortex Blobs", *Journal of Computational Physics*, Vol. 62, 1986, pp. 111-123.
- [58] I.H. Abbott and A.E. von Doenhoff, *Theory of Wing Sections Including a Summary of Airfoil Data*, Dover Publications, New York, NY, 1958.
- [59] R.A. Piziali, "2-D and 3-D Oscillating Wing Aerodynamics for a Range of Angles of Attack Including Stall", TM-4632, NASA Ames Research Centre, Moffett Field, CA, 1994.
- [60] D.A. Peters and C.J. He, "Finite State Induced Flow Models Part II: Three-Dimensional Rotor Disk", *Journal of Aircraft*, Vol. 32, No. 2, 1995, pp 323-333.
- [61] K.S. Brentner, "Challenges of Large Acoustic and Unsteady Fluid Dynamics Computations", *High Performance Computing Invited Lecture Series*, April 26, 1999, <http://www.psu.edu/dept/ihpca/spring597e99.html> (current June 2002).
- [62] F. Farassat, "Theory of Noise Generation from Moving Bodies with an Application to Helicopter Rotors", TR R-451, NASA Langley Research Centre, Hampton, VA, 1975.
- [63] K.S. Brentner, "Prediction of Helicopter Rotor Discrete Frequency Noise: A Computer Program Incorporating Realistic Blade Motions and Advanced Acoustic Formulation", TM-87721, NASA Langley Research Centre, Hampton, VA, 1986.
- [64] F. Farassat, "Generalized Functions and Kirchhoff Formulas", *2nd AIAA/CEAS Aeroacoustics Conf.*, AIAA96-1705, May 6-8, 1996, State College, PA.

- [65] K.S. Brentner, "Modeling Aerodynamically Generated Sound: Recent Advances in Rotor Noise Prediction", *Presented at 38th Aerospace Sciences Meeting and Exhibit*, AIAA 2000-0345, January 10-13, 2000, Reno, Nevada.
- [66] F. Farassat and G.P. Succi, "A Review of Propeller Discrete Frequency Noise Prediction Technology with Emphasis on two Current Methods for Time Domain Calculations", *Journal of Sound and Vibration*, Vol. 71, No. 3, 1980, pp. 399-419.
- [67] F. Farassat and G.P. Succi, "The Prediction of Helicopter Rotor Discrete Frequency Noise", *Vertica*, Vol. 7, No. 4, 1983, pp. 309-320.
- [68] K.J. Schultz et. al., "Aeroacoustic Calculation of Helicopter Rotors at DLR", *Proc. AGARD FDP Symp. on Aerodynamics and Aeroacoustics of Rotorcraft*, CP552, Berlin, Germany, Oct. 10-13, 1994.
- [69] P. Beaumier et. al., "Effect of Higher Harmonic Control on Helicopter Rotor Blade-Vortex Interaction Noise: Prediction and Initial Validation", *Proc. AGARD FDP Symp. on Aerodynamics and Aeroacoustics of Rotorcraft*, CP552, Berlin, Germany, Oct. 10-13, 1994.
- [70] K.S. Brentner, "An Efficient and Robust Method for Predicting Helicopter High-Speed Impulsive Noise", *Journal of Sound and Vibration*, Vol. 23, No. 1, 1997, pp. 87-100.
- [71] D.A. Conner and D.R. Hoad, "Reduction of High-Speed Impulsive Noise by Blade Planform Modification of a Model Helicopter Rotor", TM-84553, NASA Langley Research Centre, Hampton, VA, 1982.

- [72] X. Shang, *Aeroelastic Stability of Composite Hingeless Rotors with Finite-State Unsteady Aerodynamics*, PhD dissertation, School of Aerospace Engineering, Georgia Institute of Technology, Atlanta, Georgia, 1995.
- [73] T.R. Kane, P.W. Likins, and D.A. Levinson, *Spacecraft Dynamics*, McGraw-Hill, New York, NY, 1983.
- [74] C.E.S. Cesnik and S. Shin, "On the Modelling of Integrally Actuated Helicopter Blades", *International Journal of Solids and Structures*, Vol. 38, 2001, pp 1765-1789.
- [75] B. Bailey, F. Afagh, and F. Nitzsche, "Modelling Dynamic Equations of Motion of a Composite Hingeless Helicopter Rotor Blade", *Proc. 26th European Rotorcraft Forum*, The Hague, Netherlands, Sept. 26-29, 2000.
- [76] T. Cheng, *Structural Dynamics Modelling of Helicopter Blades for Computational Aeroelasticity*, Master's of Science dissertation, Dept. of Aeronautics and Astronautics, Massachusetts's Institute of Technology, Cambridge, Massachusetts, 2002.
- [77] W.H. Press, *Numerical Recipes in FORTRAN 77: The Art of Scientific Computing*, Cambridge University Press, New York, NY, 1992.
- [78] E.H. Dowell et. al., *A Modern Course in Aeroelasticity*, Kluwer Academic Publishers, Norwell, Massachusetts, 1989, pp15-17.
- [79] V. Kloppe, "Helicopter and Tilt-Rotor Aircraft Exterior Noise Research (HELINOISE)", *Advances in Acoustics Technology*, John Wiley and Sons, New York, NY, 1995.

Appendix A GENUVP Code

This appendix provides additional details of the GENUVP aerodynamic code – the aerodynamic component of the current work. The main topics covered in this appendix include input files and a detailed flowchart of the code. Suggestions for modelling certain parameters are also given where appropriate.

A.1 Input files

Four main input files are used for defining configurations in GENUVP. These files are as follows:

1. **dfile** ← Input of general data to define the desired configuration.
 - a.* Number of bodies
 - b.* Simulation length
 - c.* Timestep size
 - d.* Emission control parameters
 - e.* Potential calculation convergence tolerances
 - f.* Inflow parameters (e.g. rotorcraft horizontal and / or vertical velocity)
 - g.* Movement levels (maximum complexity of body movements)
 - h.* Wake deformation parameters

- i.* Atmospheric conditions (density, viscosity, speed of sound)
 - j.* *.geo file name
- 2. ***.geo** ← Primarily used to define body movements
 - a.* Body type (thin-lifting, thick-lifting, or non-lifting)
 - b.* Mesh density if not specified in ***.bld** file
 - c.* Subgrid approximation parameters
 - d.* Tip emission control (whether or not the body has tip emission)
 - e.* Body movement maximum level (i.e. complexity of the body motion)
 - f.* Details of each body movement level (rotation and/or translation)

NOTE: Care must be taken in the order in which body movements are specified. Generally movements should be specified in an order from local to global movement. The Table A.1 gives suggested orders for movement level specification for configurations that may be modelled with the code developed in this work.

<u><i>Fixed or Oscillating Wing</i></u>	<u><i>Hingeless Rotor</i></u>	<u><i>Fully Articulated Rotor</i></u>
Pitch (constant or varying)	Cyclic control	Cyclic control
--	Main rotor rotation	Lead-lag hinge distance (constant)
--	Disc tilt angle (constant)	Lead-lag motion
--	Disc roll angle (constant)	Flap hinge distance (constant)
--	--	Flap motion
--	--	Hinge distance from centre of rotation (constant)
--	--	Main rotor rotation
--	--	Disc tilt angle (constant)
--	--	Disc roll angle (constant)

Table A.1 Suggested order for movement specification for wings, hingeless rotors, and articulated rotors.

NOTE: Although it is possible to define a rotation and translation with a single movement level, for simplicity it is suggested that separate levels be used for each rotation and translation.

g. **.bld* file name

h. **.cld* file name

3. **.geo* ← Defines the body geometry.

a. Body size

- b.* Blade planform, curvature, and pretwist (as polynomial functions)
 - c.* Airfoil type (NACA 4 and 5 digit airfoils are built into code)
 - d.* Option to read in body geometry from separate file(s).
- 4. ***.cld** ← 2D Airfoil data to correct blade section loads for viscous and compressible effects.
 - a.* Tabulated lists of C_L , C_D , and C_M , indexed by angle of attack and Mach number.

A.2 GENUVP Flowchart

This section provides a flowchart of the GENUVP program structure. The scope of the flowchart only includes the aerodynamic component, with notes on interface points with the acoustic and structural components.

1. Initialize Program - p. 1/6

1. Initialize Program - p. 2/6

1. Initialize Program - p. 3/6

1. Initialize Program - p. 4/6

2-A. Potential Calculations - p1/5

2-A. Potential Calculations - p 2/5

2-A. Potential Calculations - p 3/5

2-A. Potential Calculations - p 4/5

2-A. Potential Calculations - p 5/5

2-B. Vortical Calculations - p 1/3

2-B. Vortical Calculations - p 2/3

2-B. Vortical Calculations - p 3/3

Appendix B Additional HELINOISE Analysis

Results

B.1 Thrust Histories

Figure B. 1 SMARTROTOR predicted rotor thrust history; HELINOISE case 344.

Figure B. 2 SMARTROTOR predicted rotor thrust history; HELINOISE case 508.

Figure B. 3 SMARTROTOR predicted rotor thrust history; HELINOISE case 947.

B.2 Normal Force Coefficient Histories

**Figure B. 4 Normal force coefficient history predicted by SMARTROTOR;
HELINOISE case 344.**

Figure B. 5 Normal force coefficient history predicted by SMARTROTOR;

HELINOISE case 508.

Figure B. 6 Normal force coefficient history predicted by SMARTROTOR;

HELINOISE case 947.

B.3 Aeroelastic Tip Deflections

Figure B. 7 SMARTROTOR predicted blade tip extension history; HELINOISE case 344.

Figure B. 8 SMARTROTOR predicted blade tip extension history; HELINOISE case 508.

Figure B. 9 SMARTROTOR predicted blade tip extension history; HELINOISE case 947.

Figure B. 10 SMARTROTOR predicted blade tip lead-lag history; HELINOISE case 344.

Figure B. 11 SMARTROTOR predicted blade tip lead-lag history; HELINOISE case 508.

Figure B. 12 SMARTROTOR predicted blade tip lead-lag history; HELINOISE case 947.

Figure B. 13 SMARTROTOR predicted blade tip flap history; HELINOISE case 344.

Figure B. 14 SMARTROTOR predicted blade tip flap history; HELINOISE case 508.

Figure B. 15 SMARTROTOR predicted blade tip flap history; HELINOISE case 947.

**Figure B. 16 SMARTROTOR predicted blade torsional rotation history;
HELINOISE case 344.**

**Figure B. 17 SMARTROTOR predicted blade torsional rotation history;
HELINOISE case 508.**

**Figure B. 18 SMARTROTOR predicted blade torsional rotation history;
HELINOISE case 947.**

B.4 Acoustic Pressure Histories

Figure B. 19 SMARTROTOR predicted acoustic signals at streamwise position 0.0m; HELINOISE case 1333.

**Figure B. 20 SMARTROTOR predicted acoustic signals at streamwise position
-2.0m; HELINOISE case 1333.**

**Figure B. 21 SMARTROTOR predicted acoustic signals at streamwise position
-4.0m; HELINOISE case 1333.**