

Cranfield University

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Aeroacoustic Simulation of
Rotorcraft Propulsion Systems

School of Aerospace, Transport and Manufacturing

PhD Thesis

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Supervisor: Prof Vassilios Pachidis

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Abstract

Rotorcraft constitute air vehicles with unique capabilities, including vertical take-off and landing, hover and forward/backward/lateral flight. The efficiency of rotorcraft operations is expected to improve rapidly, due to the incorporation of novel technologies into current designs. Moreover, enhanced or even new capabilities are anticipated after the introduction of advanced fast rotorcraft configurations into the future fleet.

The forecast growth in rotorcraft operations is essentially associated with an expected increase in adverse environmental impact. With respect to the forthcoming rotorcraft aviation advancements, regulatory and advisory bodies, as well as communities, have focused their attention on reducing pollutant emissions and acoustic impact of rotorcraft activity. Consequently, robust and computationally efficient noise modelling approaches are deemed as prerequisites towards quantifying the acoustic impact of present and future rotorcraft activity. Ultimately, these approaches need to cater for unique operational conditions encompassed by modern rotorcraft across designated flight procedures. Additionally, individual variations of key design variables need to be resolved, in the context of design or operational optimisation, targeted at noise mitigation.

This work elaborates on the development and application of a robust and computationally efficient methodology for the aeroacoustic simulation of rotorcraft propulsion systems. A series of fundamental modelling methods is developed for the prediction of helicopter rotor noise at fully-integrated operational level. An extensive validation is carried out against existing experimental data with respect to prediction of challenging aeroacoustic phenomena arising from complex aerodynamic interactions. The robustness of the deployed method is confirmed through a cost-effective uncertainty analysis method focused on aerodynamic sources of uncertainty. A set of generalised modelling guidelines is devised for the case of not available input parameters to calibrate the aerodynamic models.

The aspect of multi-disciplinary optimisation of rotorcraft at aircraft level in terms of maximising the potential benefits of novel technologies is also tackled within this work. A holistic schedule of optimal active rotor morphing control is derived, offering simultaneous mitigation of pollutant emissions and acoustic impact across a wide range of the helicopter flight envelope. Finally, the developed noise prediction method is incorporated into an operational-level optimisation algorithm, demonstrating the potential of active rotor morphing with respect to reduction of ground-noise impact.

The contribution to knowledge arising from the successful completion of this work comprises both the development of methodologies for helicopter aeroacoustic analysis and the derivation of guidelines and best practices for morphing rotor control. Specifically, a generic operational-level simulation approach is developed which effectively advances the state-of-the-art in mission noise prediction. New insight is provided with respect to the impact of wake aerodynamic modelling uncertainty on the robustness of noise predictions. Moreover, the aeroacoustic aspects of a novel morphing rotor concept are explored and quantifications with respect to the trade-off between environmental and noise disciplines are offered. Finally, a generalised set of optimal rotor control guidelines is derived towards achieving the challenging environmental goals set for a sustainable future rotorcraft aviation.

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Table of Contents

Abstract	i
Acknowledgements	iii
List of Tables	viii
List of Figures	x
Nomenclature	xviii
Abbreviations	i
1 Introduction	1
1.1 General Context.....	1
1.2 Scope of Work	3
1.3 Objectives	5
1.4 Overall Contribution	6
1.5 Thesis Structure.....	7
2 Literature Review	9
2.1 Introduction	9
2.2 Helicopter Rotor Noise Prediction.....	10
2.2.1 Classification of helicopter noise	10
2.2.2 Helicopter rotor aerodynamic modelling	13
2.2.3 Helicopter rotor aeroacoustic modelling	17
2.3 Uncertainty in Rotorcraft Aeroacoustic Simulation	22
2.4 Acoustic Impact of Helicopter Trajectories	26
2.5 Morphing Rotor Helicopter Concepts	28
2.5.1 Variable rotor speed	29
2.5.2 Active blade twist	31
2.5.3 Combined variable rotor speed and active blade twist	32
2.6 Discussion	34
3 Methodology	36
3.1 Introduction	36
3.2 Mathematical Modelling of Helicopter Rotor Source Noise	37
3.2.1 Rotor aerodynamic modelling.....	37
3.2.2 Near-field noise prediction method.....	41

3.3 Uncertainty Quantification in Aeroacoustic Simulation.....	46
3.4 Helicopter Rotor Noise Prediction at Mission Level	51
3.4.1 Mission flight dynamics model.....	51
3.4.2 Far-field noise propagation model.....	53
3.4.3 Integrated numerical procedure	56
3.5 Multi-Disciplinary Optimisation of Morphing Helicopter Rotors	58
3.5.1 Design space exploration and system approximation	58
3.5.2 Optimisation approach	61
3.6 Discussion	62
4 Results and Discussion; Helicopter Rotor Source Noise Prediction	64
4.1 Introduction	64
4.2 Unsteady Airloads.....	65
4.3 Blade-Vortex Interactions.....	73
4.4 Acoustic Pressure	75
4.5 Ground Noise.....	81
4.6 Computational Grid Convergence.....	86
4.7 Discussion	90
5 Results and Discussion; Uncertainty Quantification in Aeroacoustic Simulation	92
5.1 Introduction	92
5.2 Polynomial Chaos Expansion	93
5.3 Uncertainty Quantification.....	100
5.4 Sensitivity Analysis	103
5.5 Non-Deterministic Aeroacoustic Predictions	107
5.6 Fundamental Modelling Guidelines.....	111
5.7 Discussion	114
6 Results and Discussion; Helicopter Noise Prediction at Mission Level	116
6.1 Introduction	116
6.2 Adaptive Grid Sensitivity Analysis.....	117
6.3 Mission Scenarios.....	122
6.4 Impact of Fuel Burn on Source Noise Generation	125
6.5 Time-Domain Analysis	127
6.6 Cumulative Metrics of Mission Acoustic Impact	131
6.7 Assessment of Computational Performance	132
6.8 Discussion	134

7 Results and Discussion; Multi-Disciplinary Optimisation of Morphing Helicopter Rotors	136
7.1 Introduction	136
7.2 Cross Validation.....	136
7.3 Design Space Behaviour	139
7.4 Trade-Off between Source Noise and NO _x Emissions.....	146
7.5 Optimised Rotor Control Schedules.....	151
7.6 Discussion	158
 8 Results and Discussion; Aeroacoustics of Optimised Morphing Helicopter Rotors	 160
8.1 Introduction	160
8.2 Trajectory Definition	161
8.3 Blade-Vortex Interactions.....	166
8.4 Time-Domain Analysis	173
8.5 Ground Noise Impact	175
8.6 Discussion	177
 9 Summary and Conclusions	 178
9.1 Context Overview.....	178
9.2 Achievement of Set Objectives	178
9.2.1 Helicopter rotor source noise prediction	179
9.2.2 Uncertainty quantification in rotorcraft aeroacoustics	180
9.2.3 Helicopter noise prediction at mission level.....	181
9.2.4 Optimisation of novel morphing rotor helicopter rotors	182
9.2.5 Aeroacoustic analysis of optimised morphing helicopter rotors	183
9.3 Contribution to Knowledge	184
9.4 Future Work	186
 References	 189
 Appendix A: Author's Publications	 202

List of Tables

4.1	HART II rotor geometrical characteristics	65
4.2	HART II wind tunnel ambient and operational conditions.	66
4.3	Summary of grid convergence investigated cases.	86
4.4	Grid Convergence Indices for p'_{amp} and $OASPL$.	89
5.1	Distributions of uncertain input variables when tip-vortex experimental data are available (Gaussian distribution with $COV = 10\%$).	93
5.2	Distributions of uncertain input variables when tip-vortex experimental data are not available (Uniform distribution with $COV = 30\%$).	94
5.3	Statistics of leave-one-out Cross Validation of the derived PCE when tip-vortex experimental data are available (Gaussian distribution with $COV = 10\%$).	97
5.4	Statistics of leave-one-out Cross Validation of the derived PCE when tip-vortex experimental data are not available (Uniform distribution with $COV = 30\%$).	98
5.5	Uncertainty quantification when tip-vortex experimental data are available (Gaussian distribution with $COV = 10\%$).	99
5.6	Uncertainty quantification when tip-vortex experimental data are not available (Uniform distribution with $COV = 30\%$).	100
5.7	Sensitivity of output objectives to uncertain inputs when tip-vortex experimental data are available (Gaussian distribution with $COV = 10\%$).	101

5.8	Sensitivity of output objectives to uncertain inputs when tip-vortex experimental data are not available (Uniform distribution with $COV = 30\%$).	102
6.1	Bo 105 main rotor geometrical characteristics.	117
6.2	Cases examined in the observer grid sensitivity analysis.	119
6.3	Rotor airloads model computational performance.	133
6.4	Rotor aeroacoustics model computational performance.	133
7.1	Design space bounds and values of design and operational parameters to be directly simulated for the DSE process.	137
7.2	Statistical properties of LOOCV for the objectives of interest.	139

List of Figures

3.1	Free-wake rotor airloads model overview.	38
3.2	Tip vortex roll-up model overview	40
3.3	Concepts of permeable and impermeable acoustic data surface.	42
3.4	Uncertainty quantification method overview.	50
3.5	Rotorcraft flight dynamics model overview.	52
3.6	Rotor aeroacoustics model overview.	54
3.7	Integrated mission simulation numerical procedure.	57
3.8	Overview of Design Space Exploration approach.	60
4.1	Comparison of calculated normal force coefficient with HART II experimental data; (a) Baseline case, (b) Minimum noise case, (c) Minimum vibration case.	68
4.2	Comparison of calculated time-derivative of normal force coefficient with HART II experimental data; (a) Baseline case, (b) Minimum noise case, (c) Minimum vibration case.	70
4.3	Distribution of the time-derivative of normal force coefficient across the rotor disc for the HART II baseline case and comparison with potential BVI locations calculated with the spherical detection method.	72
4.4	Loading noise generation across the rotor disc for the HART II baseline case; (a) advancing side observer, (b) retreating side observer.	74
4.4	Comparison of predicted acoustic pressure history with experimental measurements for the HART II baseline case at a single observer located 2.215 m below the rotor disc, 0.0 m upstream and 1.81 m at the advancing side of the rotor; (a) acoustic pressure, (b) spectral analysis.	76
4.6	Comparison of predicted acoustic pressure history with experimental measurements for the HART II baseline case at a single observer located 2.215 m below the rotor disc, 0.92 m downstream and 2.00 m at the retreating side of the rotor; (a) acoustic pressure, (b) spectral analysis.	78

4.7	Sound pressure level relative error between predicted and measured acoustic pressure in the spectral domain; (a) advancing side observer, (b) retreating side observer.	79
4.8	Comparison of ground noise 2.215 m below the rotor disc of the baseline case of HART II rotor with measured data; (a) wind tunnel, (b) simulation.	80
4.9	Distribution of Overall Sound Pressure Levels average relative deviation between predicted and measured ground noise for the baseline case of the HART II experiment.	81
4.10	Comparison of ground noise 2.215 m below the rotor disc of the minimum noise case of HART II with measured data; (a) wind tunnel, (b) simulation.	82
4.11	Distribution of Overall Sound Pressure Levels average relative deviation between predicted and measured ground noise for the minimum noise case of the HART II experiment.	83
4.12	Comparison of ground noise 2.215 m below the rotor disc of the minimum vibration case with measured data; (a) wind tunnel, (b) simulation.	84
4.13	Distribution of Overall Sound Pressure Levels average relative deviation between predicted and measured ground noise for the minimum vibration case of the HART II experiment.	85
4.14	Distribution of actual fractional error as function of the discretisation parameters: (a) evaluation for p'_{amp} and (b) evaluation for $OASPL$.	88
5.1	Comparison of probability density functions for Gaussian and uniform distributions of r_{c0}^{wake} uncertain input variable; (a) $COV=10\%$ for both distributions, (b) $COV=10\%$ for Gaussian and $COV=30\%$ for uniform.	95
5.2	Leave-one-out Cross Validation for the case of available tip-vortex experimental data, applied to the developed Polynomial Chaos Expansion, referring to the advancing side observer of HART II experiment; (a) Amplitude of acoustic	96

	pressure pulse, (b) OASPL.	
5.3	Non-deterministic predictions of acoustic pressure at the advancing side observer of the HART II experiment: mean acoustic pressure and uncertainty bars; (a) Gaussian input with $COV = 10\%$, (b) uniform input with $COV = 30\%$.	104
5.4	Non-deterministic predictions of spectral content at the advancing side observer of the HART II experiment: mean SPL and uncertainty bars; (a) Gaussian input with $COV = 10\%$, (b) uniform input with $COV = 30\%$.	105
5.5	Non-deterministic predictions of OASPL below the HART II rotor for Gaussian input with $COV = 10\%$: (a) mean OASPL, (b) standard deviation.	106
5.6	Non-deterministic predictions of OASPL below the HART II rotor for uniform input with $COV = 30\%$: (a) mean OASPL, (b) standard deviation.	107
5.7	Comparison of probability histograms representing the amplitude of acoustic pressure pulse for Gaussian and uniform input uncertainty; (a) $COV = 10\%$, (b) $COV = 30\%$.	108
5.8	Comparison of probability histograms representing OASPL for Gaussian and uniform input uncertainty; (a) $COV = 10\%$, (b) $COV = 30\%$.	110
5.9	Impact of input uncertainty on the prediction of acoustic pressure at different levels of COV ; (a) Gaussian input, (b) Uniform input.	111
5.10	Impact of input uncertainty on the prediction of OASPL at different levels of COV ; (a) Gaussian input, (b) Uniform input.	113
6.1	Adaptive observer grid sensitivity analysis in level-flight for $h/R = 100$, $\mu = 0.16$ and $C_T = 0.0044$; (a) Ground noise footprint, (b) Free-stream and rotation directions.	118
6.2	Adaptive observer grid sensitivity analysis in level-flight for $h/R = 100$, $\mu = 0.16$ and $C_T = 0.0044$; (a) OASPL absolute X-gradient, (b) OASPL absolute Y-gradient.	119
6.3	Summary of the limits of the high-resolution regions, exported from the observer grid sensitivity analysis; (a) X-	120

	direction, (b) Y-direction.	
6.4	PAT scenario geographical definition in WGS 84.	121
6.5	SAR scenario geographical definition in WGS 84.	122
6.6	PAT scenario time variations of deployed operational altitude advance ratio, flightpath angle and thrust coefficient.	123
6.7	SAR scenario time variations of deployed operational altitude advance ratio, flightpath angle and thrust coefficient.	124
6.8	Loading noise acoustic spheres for the SAR scenario; (a) beginning of cruise 1: $t = 455$ s, $\mu = 0.206$, $\gamma = 0$ deg., $C_T = 0.0054$, (b) ending of the cruise 3: $t = 5965$ s, $\mu = 0.206$, $\gamma = 0^\circ$, $C_T = 0.0049$, (c) OASPL difference between (a) and (b).	125
6.9	Selected trajectory waypoint of the PAT scenario; (a) acoustic sphere, (b) corresponding ground noise footprint at $t = 1180$ s, $\mu = 0.230$, $\gamma = 0$ deg., $C_T = 0.0060$.	126
6.10	Selected trajectory waypoint of the SAR scenario; (a) acoustic sphere, (b) corresponding ground noise footprint at $t = 4990$ s, $\mu = 0.185$, $\gamma = -7.13$ deg., $C_T = 0.0047$.	128
6.11	Acoustic histories for selected isolated observers A and B for each mission; (a) PAT, (b) SAR.	129
6.12	Sound exposure levels; (a) PAT, (b) SAR.	132
7.1	Leave-one-out Cross Validation applied to the developed RSM; (a) $EAOASPL$, (b) NO_x^{rate} .	138
7.2	Design Space Exploration: impact of all-up-mass (AUM) on variable rotor speed and active blade twist noise and NO_x emissions at $h = 250$ m, $V_x = 30$ m / s, $V_z = -4$ m / s; (a) $AUM = 1600$ kg, (b) $AUM = 2400$ kg.	139
7.3	Correlation between rotor power requirement and NO_x for $V_z = 0$ m / s and $V_z = -5$ m / s at $AUM = 1600$ kg.	
7.4	Design Space Exploration: impact of density altitude (h) on combined variable rotor speed and active blade twist noise and NO_x emissions at $AUM = 2000$ m, $V_x = 30$ m / s, $V_z = -4$ m / s; (a) $h = 50$ m, (b) $h = 450$ m.	141

7.5	Design space behaviour: impact of flight speed (V_x) on combined variable rotor speed and active blade twist noise and NO_x emissions at $h = 250 \text{ m}$, $AUM = 2000 \text{ kg}$, $V_z = -4 \text{ m/s}$; (a) $V_x = 20 \text{ m/s}$, (b) $V_x = 40 \text{ m/s}$.	142
7.6	Design space behaviour: impact of climb rate (V_z) on combined variable rotor speed and active blade twist noise and NO_x emissions at $h = 250 \text{ m}$, $V_x = 30 \text{ m/s}$, $AUM = 2000 \text{ kg}$; (a) $V_z = 0 \text{ m/s}$, (b) $V_z = -5 \text{ m/s}$.	143
7.7	Impact of all-up mass (AUM) on Pareto fronts of optimised combined variable rotor speed and active blade twist helicopter rotor at $h = 250 \text{ m}$, $V_x = 30 \text{ m/s}$, $V_z = -4 \text{ m/s}$.	144
7.8	Impact of density altitude (h) on the Pareto fronts of optimised combined variable rotor speed and active blade twist helicopter rotor at $AUM = 2000 \text{ kg}$, $V_x = 30 \text{ m/s}$, $V_z = -4 \text{ m/s}$.	145
7.9	Impact of flight speed (V_x) on the Pareto fronts of optimised combined variable rotor speed and active blade twist helicopter rotor at $h = 250 \text{ m}$, $AUM = 2000 \text{ kg}$, $V_z = -4 \text{ m/s}$.	146
7.10	Impact of climb rate (V_z) on the Pareto fronts of optimised combined variable rotor speed and active blade twist helicopter rotor at $h = 250 \text{ m}$, $V_x = 30 \text{ m/s}$, $AUM = 2000 \text{ kg}$.	147
7.11	Optimised rotor control schedules for $AUM = 1600 \text{ kg}$: Optimal rotor speed for (a) Noise, (b) trade-off, (c) NO_x ; Optimal blade twist angle for (d) Noise, (e) trade-off, (f) NO_x .	149
7.12	Acoustic and environmental benefits for $AUM = 1600 \text{ kg}$: Noise benefits through optimising for (a) Noise, (b) trade-off, (c) NO_x ; NO_x benefits through optimising for (d) Noise, (e) trade-off, (f) NO_x .	150
7.13	Optimised rotor control schedules for $AUM = 2000 \text{ kg}$: Optimal rotor speed for (a) Noise, (b) trade-off, (c) NO_x ; Optimal blade twist angle for (d) Noise, (e) trade-off, (f) NO_x .	152

7.14	Acoustic and environmental benefits for $AUM = 2000\text{ kg}$: Noise benefits through optimising for (a) Noise, (b) trade-off, (c) NO_x ; NO_x benefits through optimising for (d) Noise, (e) trade-off, (f) NO_x .	153
7.15	Optimised rotor control schedules for $AUM = 2400\text{ kg}$: Optimal rotor speed for (a) Noise, (b) trade-off, (c) NO_x ; Optimal blade twist angle for (d) Noise, (e) trade-off, (f) NO_x .	155
7.16	Acoustic and environmental benefits for $AUM = 2400\text{ kg}$: Noise benefits through optimising for (a) Noise, (b) trade-off, (c) NO_x ; NO_x benefits through optimising for (d) Noise, (e) trade-off, (f) NO_x .	156
8.1	Approach path definition, selected control points and microphone locations.	161
8.2	Trajectory profile; (a) altitude, (b) flight speed, (c) flight path angle as functions of mission time.	162
8.3	Optimised rotor control profiles; (a) rotor speed, (b) blade twist angle as functions of mission time for different all-up- mass (AUM) options.	163
8.4	Evolution of main rotor wake across the approach trajectory for $AUM = 1600\text{ kg}$; (a) baseline rotor control at $t = 50\text{ s}$ with $V_x = 40\text{ m/s}$ $V_z = 0\text{ m/s}$, (b) optimised rotor control at $t = 50\text{ s}$ with $V_x = 40\text{ m/s}$ $V_z = 0\text{ m/s}$, (c) baseline rotor control at $t = 135\text{ s}$ with $V_x = 40\text{ m/s}$ $V_z = -4.2\text{ m/s}$, (d) optimised rotor control at $t = 135\text{ s}$ with $V_x = 40\text{ m/s}$ $V_z = -4.2\text{ m/s}$, (e) baseline rotor control at $t = 205\text{ s}$ with $V_x = 20\text{ m/s}$ $V_z = 0\text{ m/s}$, (f) optimised rotor control at $t = 205\text{ s}$ with $V_x = 20\text{ m/s}$ $V_z = 0\text{ m/s}$.	164

- 8.5 Evolution of potential BVI locations on the main rotor disc for $AUM = 1600 \text{ kg}$; (a) baseline rotor control at $t = 50 \text{ s}$ with $V_x = 40 \text{ m/s}$ $V_z = 0 \text{ m/s}$, (b) optimised rotor control at $t = 50 \text{ s}$ with $V_x = 40 \text{ m/s}$ $V_z = 0 \text{ m/s}$, (c) baseline rotor control at $t = 135 \text{ s}$ with $V_x = 40 \text{ m/s}$ $V_z = -4.2 \text{ m/s}$, (d) optimised rotor control at $t = 135 \text{ s}$ with $V_x = 40 \text{ m/s}$ $V_z = -4.2 \text{ m/s}$, (e) baseline rotor control at $t = 205 \text{ s}$ with $V_x = 20 \text{ m/s}$ $V_z = 0 \text{ m/s}$, (f) optimised rotor control at $t = 205 \text{ s}$ with $V_x = 20 \text{ m/s}$ $V_z = 0 \text{ m/s}$. 165
- 8.6 Evolution of time-derivative of normal force coefficient for $AUM = 1600 \text{ kg}$; (a) baseline rotor control at $t = 50 \text{ s}$ with $V_x = 40 \text{ m/s}$ $V_z = 0 \text{ m/s}$, (b) optimised rotor control at $t = 50 \text{ s}$ with $V_x = 40 \text{ m/s}$ $V_z = 0 \text{ m/s}$, (c) baseline rotor control at $t = 135 \text{ s}$ with $V_x = 40 \text{ m/s}$ $V_z = -4.2 \text{ m/s}$, (d) optimised rotor control at $t = 135 \text{ s}$ with $V_x = 40 \text{ m/s}$ $V_z = -4.2 \text{ m/s}$, (e) baseline rotor control at $t = 205 \text{ s}$ with $V_x = 20 \text{ m/s}$ $V_z = 0 \text{ m/s}$, (f) optimised rotor control at $t = 205 \text{ s}$ with $V_x = 20 \text{ m/s}$ $V_z = 0 \text{ m/s}$. 167
- 8.7 Evolution of main rotor noise hemispheres for $AUM = 1600 \text{ kg}$; (a) baseline rotor control at $t = 50 \text{ s}$ with $V_x = 40 \text{ m/s}$ $V_z = 0 \text{ m/s}$, (b) optimised rotor control at $t = 50 \text{ s}$ with $V_x = 40 \text{ m/s}$ $V_z = 0 \text{ m/s}$, (c) baseline rotor control at $t = 135 \text{ s}$ with $V_x = 40 \text{ m/s}$ $V_z = -4.2 \text{ m/s}$, (d) optimised rotor control at $t = 135 \text{ s}$ with $V_x = 40 \text{ m/s}$ $V_z = -4.2 \text{ m/s}$, (e) baseline rotor control at $t = 205 \text{ s}$ with $V_x = 20 \text{ m/s}$ $V_z = 0 \text{ m/s}$, (f) optimised rotor control at $t = 205 \text{ s}$ with $V_x = 20 \text{ m/s}$ $V_z = 0 \text{ m/s}$. 169
- 8.8 Comparison of baseline and optimised noise at microphone C, around $t = 135 \text{ s}$; (a) acoustic pressure pulse, (b) spectral analysis. 171
- 8.9 Comparison of baseline and optimised rotor control in terms of ground noise at three selected microphones; (a) microphone A, (b) microphone B, (c) microphone C. 172

8.10	Noise and pollutant emissions benefits for three different AUM, arising from optimised rotor control as function of mission time; (a) Noise benefits, (b) NO _x benefits.	173
8.11	Sound Exposure Levels (SEL) of the approach trajectory for $AUM = 1600 \text{ kg}$; (a) baseline, (b) optimised, (c) difference (b)-(a).	174
8.12	Sound Exposure Levels (SEL) of the approach trajectory for $AUM = 2000 \text{ kg}$; (a) baseline, (b) optimised, (c) difference (b)-(a).	175
8.13	Sound Exposure Levels (SEL) of the approach trajectory for $AUM = 2400 \text{ kg}$; (a) baseline, (b) optimised, (c) difference (b)-(a).	176

Nomenclature

Symbol	Description	Unit
<i>Roman symbols</i>		
a	Lower limit of uniform distribution	
AUM	All-up-mass	kg
A_{cs}	Blade element cross sectional area	m ²
b	Upper limit of uniform distribution	
c_0	Speed of sound of quiescent medium	m/s
c_b	Blade chord	m
C_L	Rotor disc rolling moment coefficient = $L / (\rho \pi \Omega^2 R^5)$	
C_M	Rotor disc pitching coefficient = $M / (\rho \pi \Omega^2 R^5)$	
$C_n M^2$	Normal force coefficient = $F_n / (0.5 \rho c_0^2 c_b)$	
C_T	Rotor disc thrust coefficient = $T / (\rho \pi \Omega^2 R^4)$	
COV	Coefficient of variation = μ / σ	
d	Perpendicular distance between a point and a vortex filament	m
D	Differential of a physical property	
dy	Blade differential span length	m
dS	Blade elementary surface	m ²
E	Mean value of statistical sample	
$EAOSPL$	Energetically averaged overall sound pressure level	dB
$\hat{f} = 0$	Acoustic data line function	
F_n	Sectional normal force	N/m
h	Flight altitude	m
H	Heaviside function	
H_0	Hermite orthogonal polynomial	
$\vec{l}_{12}$	Vortex filament vector	m
$\vec{L}$	Blade element section loading vector	N/m

L	Rotor rolling moment	Nm
$\vec{M}$	Relative Mach number vector	
$ \vec{M} $	Magnitude of relative Mach number vector	
M	Rotor pitching moment	Nm
Mo	Mode of distribution	
n	Number of uncertain input parameters	
$\hat{n}$	Surface unit outward vector	
$N_{Pearson}$	Pearson's correlation index	
NO_x	NO _x emissions	kg
NO_x^{rate}	NO _x rate	kg/s
$OASPL$	Overall sound pressure level	dB
p	Order of polynomial expansion	
p'	Acoustic pressure	Pa
p'_m	Monopole acoustic pressure	Pa
p'_d	Dipole acoustic pressure	Pa
P_{ij}	Compressible stress tensor	Pa
$\vec{r}$	Position vector of a point on a vortex filament	m
$\hat{r}$	Unit radiation vector	
r	Radiation distance	m
r_c	Tip-vortex viscous core radius	m
r_{c0}	Initial tip-vortex viscous core radius	m
r_{refine}	Computational grid refinement ratio	
$\vec{r}_1$	Position vector of a point relative to the beginning of a vortex filament	m
$\vec{r}_2$	Position vector of a point relative to the end of a vortex filament	m
R	Rotor radius	m
$R(n,m)$	Shorthand notation function	m ⁻ⁿ
S_i	Sobol sensitivity index	
SEL	Sound exposure levels	dB
t	Observer time	s

T	Rotor thrust	N
T_0	Reference duration of acoustic event	s
T_{ij}	Lighthill stress tensor	Pa
$\vec{u}$	Fluid velocity vector	m/s
$\vec{v}$	Acoustic source surface velocity vector	m/s
V	Variance of statistical sample	
$\vec{V}$	Flow velocity vector	m/s
$\vec{V}_{ind}$	Induced velocity vector	m/s
V_x	Horizontal component of flight velocity	m/s
V_z	Vertical component of flight velocity (rate of climb)	m/s
W	Shorthand notation term	m/s
$\vec{x}$	Observer position vector, with components x_i	m
y	Response of system at specific collocation point	
$\vec{y}$	Source position vector	m
Y	Non-deterministic response of system	

Greek symbols

α	Oseen constant for the vortex core radius	
α_1	Squire parameter for the eddy viscosity factor	
a_{eff}	Effective rotor shaft angle	rad
a_i	Coefficient of orthogonal polynomial basis function	
α_s	Wind tunnel rotor shaft angle	rad
β	Acoustic sphere elevation angle	rad
γ	Flight path angle	rad
γ_1	Skewness of distribution	
γ_2	Excess kurtosis of distribution	
Γ	Circulation	m ² /s
δ	Eddy viscosity parameter	
δ_{ij}	Kronecker delta	

$\delta(f)$	Dirac delta function	
ζ	Distance along trailed tip-vortex	rad
θ	Blade pitch angle	rad
θ_{1c}	Rotor lateral cyclic pitch angle	rad
θ_{1s}	Rotor longitudinal cyclic pitch angle	rad
$\theta_{.75}$	Rotor collective pitch angle applied at 0.75R	rad
θ_3	Amplitude of higher harmonic rotor control	rad
θ_{el}	Blade elastic twist deformation	rad
θ_{tw}	Rotor blade linear twist angle	rad
μ	Mean value of distribution	
μ	Advance ratio	
ν	Kinematic viscosity	m ² /s
ξ	Value within known distribution	
ρ	Density of acoustic medium	kg/m ³
σ	Standard deviation of distribution	
τ	Source time	s
φ_i	Polynomial basis function	
ψ	Blade azimuthal position	rad
ψ_3	Phase angle of higher harmonic control	rad
$\vec{\omega}$	Vorticity of vortex element	1/s
Ω	Rotor rotational speed	rad/s

Subscripts

$(\cdot)_0$	Physical property of the quiescent medium
$(\cdot)_{amp}$	Amplitude

Superscripts

$(\cdot)^{adv}$	Quantity referring to rotor advancing side
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$(\cdot)^{exact}$	Quantity referring to the exact value for an infinitely fine computational grid
$(\cdot)^{inflow}$	Vortex core quantity used to evaluate induced velocity for inflow calculations
$(\cdot)^{Opt}$	Optimised quantity
$(\cdot)^{ret}$	Quantity referring to rotor retreating side
$(\cdot)^{wake}$	Vortex core quantity used to evaluate induced velocity for wake calculations

Abbreviations

ABT	Active Blade Twist
ACARE	Advisory Council for Aviation Research in Europe
ADS	Acoustic Data Surface
ATS	Air-Transport System
BL	Baseline
BPF	Blade Passage Frequency
BVI	Blade-Vortex Interaction
CFD	Computational Fluid Dynamics
CSD	Computational Structural Dynamics
DNW	Duits-Nederlandse Windtunnel
EASA	European Union Aviation Safety Agency
FFT	Fast Fourier Transformation
FPT	Free Power Turbine
FW-H	Ffowcs Williams-Hawkings
GG	Gas Generator
GCI	Grid Convergence Index
GRC	Green RotorCraft
HECTOR	HEliCopter Omni-disciplinary Research platform
HHC	Higher Harmonic Control
HSI	High-Speed Impulsive
IBC	Individual Blade Control

ICAO	International Civil Aviation Organization
ICARUS	Integral Computational AeRoacoustics Unified Software
ITD	Integrated Technology Demonstrator
JTI	Joint Technology Initiative
LES	Large-Eddy Simulation
LOOCV	Leave-One-Out Cross-Validation
MAE	Mean Absolute Error
MC	Monte Carlo
MN	Minimum Noise
MTOW	Maximum Take-Off Weight
MV	Minimum Vibration
NOTAR	No Tail-Rotor device
NRMSE	Normalised Root Mean Square Error
NSGA II	Non-dominated Sorting Genetic Algorithm II
PAT	Passenger Air-Transport
PCE	Polynomial Chaos Expansion
PDF	Probability Density Function
RAM	Random Access Memory
RSM	Response Surface Model
SAR	Search and Rescue
SC	Stochastic Collocation
SFC	Specific Fuel Consumption
SRIA	Strategic Research and Innovation Agenda

TE	Technology Evaluator
TEL	Twin-Engine Light
UC	Uncertainty Quantification
VRS	Variable Rotor Speed
WGS 84	World Geodetic System of 1984

1 Introduction

1.1 General Context

Rotorcraft has always been a versatile means of transportation, fulfilling a unique role in aerospace operations. Rotorcraft are equipped with one or more rotors to provide thrust throughout the flight envelope. A helicopter is a type of rotorcraft with single main-rotor and single tail-rotor or no tail-rotor (NOTAR) device. Civil helicopters have been predominantly used for passenger transportation, fire suppression, rescue, aerial work, law enforcement and medical evacuation. Present civil rotorcraft activities amount to approximately 1.5 million flight hours per year within the European airspace, representing an annual consumption of 400,000 tons of aviation fuel (Ref. [1]).

Rotorcraft traffic is expected to develop sharply in the near future, due to (a) enhancement of flight efficiency and operability of the current helicopter fleet through improvements achieved through application of novel technologies, and (b) extension of flight envelope through introduction of advanced rotorcraft configurations, like tiltrotor aircraft and compound rotorcraft (Ref. [2]). The aspiring applications offer improved, or even new, capabilities to passengers, including intercity transportation, city-to-airport shuttle operations and island-mainland connection. The forecast growth in rotorcraft operations is essentially leading to a significant increase in the associated environmental impact, if no change is made on technology levels or operational procedures.

With respect to the forthcoming rotorcraft aviation advancements, regulatory and advisory bodies have focused their present and future attention on reducing pollutant emissions and acoustic impact. Notably for noise, the International Civil Aviation Organization (ICAO) requirements for new helicopter certification set noise margins at the same level of importance as safety, reliability and manufacturability (Ref. [3]). The Advisory Council for Aviation Research and innovation in Europe (ACARE) has developed a Strategic Research and Innovation Agenda (SRIA) to meet the challenging goals set by FlightPath 2050. These include a 75% reduction in CO₂ emissions per passenger kilometre, a

90% reduction in NO_x emissions and a 15 dB reduction in Effective Perceived Noise (EPNdB) per operation (departure or approach) (Refs. [4], [5]). The targeted noise reduction refers to departure or approach operational profiles and is quantified through the cumulative time-integrated EPNdB metric for the noise levels of entire profile. The aforementioned reductions are relative to the capabilities of typical new aircraft in 2000.

Noise-related issues have led to operational restrictions and limitations to airport construction or expansion. ICAO has proposed the so called “Balanced Approach” (Ref. [6]) to aircraft noise management in the vicinity of airports. This promotes the achievement of maximum acoustic benefits in the most cost-efficient way, through combination of the following measures: (a) reduction of noise at source, (b) land-use planning and management, (c) noise abatement operational procedures and (d) operational restrictions.

To meet the aforementioned objectives, public acceptance, as well as noise certification capability has emerged as dominant matter for the manufacturers. Consequently, the quantification of acoustic impact of rotorcraft activity, prior to certification, has become of utmost priority for the aerospace sector. Clean Sky Joint Technology Initiative (JTI) is the largest European research programme, developing innovative technology aimed at reducing gaseous and noise emissions produced by current and future aircraft (Ref. [7]). The Clean Sky Technology Evaluator (TE) is an integral part of the Clean Sky programme, responsible for assessing the environmental and acoustic impact of innovative rotorcraft technologies, and determining the extent of Clean Sky’s contribution to ACARE’s environmental objectives.

Robust and computationally efficient noise modelling approaches are deemed as prerequisites towards quantifying the acoustic impact of present and future rotorcraft activity. These approaches need to cater for unique operational conditions encompassed by modern rotorcraft across designated flight procedures, as well as individual variations of key design variables in the context of design optimisation targeted at noise mitigation. Therefore, an advanced level of simulation fidelity is required, ultimately based on first-

principles. However, the associated computational overhead should be maintained within acceptable thresholds, to enable the capability of incorporating advanced aeroacoustic simulation in the context of multi-disciplinary optimisation and holistic noise assessment of present and future rotorcraft configurations.

1.2 Scope of Work

Aeroacoustic simulation of rotorcraft comprises multiple disciplines including unsteady aerodynamics, aeroelasticity, flight dynamics and acoustics. Various modelling approaches have been proposed, ranging from simple empirical correlations to highly-complex numerical methods (Refs. [8]–[11]). The identification of appropriate modelling methods depends on the scope and is essentially determined from the trade-off between predictive accuracy and computational efficiency. Each of the modelling approaches is fundamentally associated with inherent sources of uncertainty, which effectively propagates into aeroacoustics, affecting the robustness of the outcomes. Therefore, quantification of the impact of modelling uncertainty on noise predictions is a prerequisite before implementing the comprehensive models into simulation. Comprehensive simulation needs to resolve the strong variations in flight speed, flight path angle, altitude and gross weight, encountered by helicopters across “real-world” operations. These variations are naturally reflected on the associated acoustic ground footprint. Hence, noise prediction should be ultimately treated at mission-level, rather than at individual flight states.

Computationally efficient, robust and validated aeroacoustic models can be directly employed for the assessment of noise impact of existing or conceptual helicopter configurations. Novel helicopter technologies aim to improve operability and mitigate adverse performance, environmental and aeroacoustic effects, actively or passively (Refs. [12], [13]). Active rotor morphing technologies have the potential to effectively reduce noise at source through time-varied tailoring of rotor geometrical characteristics to the unique operational requirements which arise across helicopter missions.

Combined variable rotor speed (VRS) and active blade twist (ABT) constitutes a newly proposed, promising active rotor morphing concept, which can contribute towards the first measure proposed by ICAO for airport noise management. ABT refers to the collective variation of blade twist angle with respect to the individual operating state of the helicopter. The potential improvements arising from the implementation of combined VRS and ABT rotor technologies on present or future helicopters can be extended to various aspects of rotorcraft operation, including performance and pollutant emissions, along with noise. Ultimately, the multi-disciplinary optimisation of helicopters employing VRS and ABT technologies has the potential to maximise the anticipated benefits and contribute towards fulfilling the targets set by ACARE.

In this context, the overall scope of this work can be expressed by two fundamental requirements:

- Generic and robust methodology for modelling helicopter aeroacoustics, able to resolve individual design variations and applicable on three-dimensional mission simulation.
- Multi-disciplinary optimisation of combined variable rotor speed and active blade twist concepts targeting the simultaneous mitigation of environmental and acoustic impact at both aircraft and operational level.

The main ambition of this work is not only to develop and validate a computationally efficient noise prediction strategy, but also to utilise the developed methodology in uncertainty quantification, mission simulation and multi-disciplinary optimisation applications. This ambition targets at: (a) demonstrating the applicability and robustness of the developed approach within demanding engineering applications, and (b) contributing towards shedding light on technologies to achieve the challenging environmental targets set for a sustainable rotorcraft aviation.

1.3 Objectives

A comprehensive set of individual objectives is defined to address the fundamental requirements formulated in Section 1.2. Specifically:

1. *Development and validation of an analytical procedure for the evaluation of helicopter rotor aeroacoustics resolving rotorcraft flight dynamics, blade design, three-dimensional kinematics of elastic rotor blades and unsteady rotor aerodynamic loading.*
2. *Implementation of a computationally-efficient uncertainty quantification method to assess the robustness of the developed noise prediction approach and to provide fundamental modelling guidelines for non-deterministic aeroacoustic simulation in the absence of reliable experimental data for aerodynamic correlations.*
3. *Introduction of a novel, mission simulation approach based on the integration of existing flight dynamics and aerodynamics models with the developed aeroacoustics approach, capable of simulating realistic three-dimensional helicopter operations.*
4. *Incorporation of the developed noise prediction method into an elaborate multi-disciplinary optimisation approach, undertaken at aircraft level, targeting the concurrent reduction of pollutant and noise emissions of novel morphing-rotor helicopter configurations.*
5. *Assessment of the aerodynamics and aeroacoustics of optimised variable rotor speed and active blade twist concepts at operational level, with respect to realistic trajectories in the vicinity of airports.*

It is noted that significant weight has been placed with regards to the 1st objective, which indispensably constitutes the core of all activities undertaken in the other four objectives.

1.4 Overall Contribution

The main contribution arising from the successful completion of this work is the development of a computational framework for the aeroacoustic simulation of helicopter rotors. This framework effectively comprises a series of mathematical formulations, tackling the noise prediction problem in a computationally efficient manner based on first-principles. The framework has been named “*ICARUS*” (Integral Computational AeRoacoustics Unified Software) after the homonymous mythological hero, son of the master craftsman Daedalus, the creator of the Labyrinth. ICARUS, developed in FORTRAN 90/95 and comprising more than 20,000 lines of code, has been delivered in a fully productionised layout to the Centre for Propulsion Engineering at Cranfield University. ICARUS is currently utilised in a number of MSc and PhD research projects across the Centre. Parts of the outcomes obtained in this work have found place in Research Proposals related to rotorcraft research, submitted by Cranfield University.

Furthermore, the author has actively participated in the Technology Evaluator activities of the Clean Sky Green Rotorcraft (GRC) Integrated Technology Demonstrator (ITD) European Research Programme. Specifically, contribution has been made towards assessing the impact of novel helicopter technologies on fuel burn, pollutant and noise emissions at mission, operational and air-transport system (ATS) level.

Further to computational development, scientific contribution has been made towards various subjects associated with rotorcraft research. These specifically comprise:

- A generic methodology, comprising a series of fundamental mathematical formulations for the prediction of helicopter rotor noise in the time-domain.
- A computationally efficient strategy for uncertainty quantification in helicopter aeroacoustic simulation, accompanied by a set of fundamental modelling guidelines for the non-deterministic aeroacoustic analysis of helicopter rotors in the absence of sufficient aerodynamic input data.

- A novel approach to helicopter mission simulation, which alleviates the need of pre-stored acoustic data and enables the prediction of source noise on-the-fly.
- A generalised control schedule for combined variable rotor speed and active blade twist morphing, offering simultaneous reduction of helicopter pollutant and acoustic emissions.
- Insight into the aerodynamics and aeroacoustics of optimised variable rotor speed and active blade twist concepts, with respect to realistic trajectories in the vicinity of airports.

The contribution to knowledge achieved through the scientific outcomes obtained during the period of this PhD research, has been disseminated in twelve publications, specifically: eight international conference papers, one published high-impact journal article and three more articles to be submitted to high-impact scientific journals.

1.5 Thesis Structure

This thesis has been structured in a Monograph format, comprising the following parts: Introduction, Literature Review, Methodology, Results and Discussion, and Summary and Conclusions. The Results and Discussion part has been divided into five chapters, referring to the individual objectives defined in Section 1.3. The remaining parts constitute individual chapters, resulting to a total of nine chapters. Each of the Chapters 2-8 begins with a dedicated introductory section and ends with a section focused on discussion of obtained outcomes. The contents of each of the nine chapters are summarised as follows:

The 1st chapter constitutes an introduction to the context of this work, describing the overall scope, individual objectives and contribution arising from the successful completion of undertaken research.

The 2nd chapter provides a critical literature review on the specific fields related to this study. Emphasis is given to computational methods for helicopter rotor noise prediction, whilst significant effort has also been placed on uncertainty quantification in aeroacoustics, ground noise impact of helicopter trajectories and aeroacoustics of morphing-rotor helicopter rotor concepts.

The 3rd chapter offers a detailed description of the methods developed or employed to achieve the individual objectives set in the context of this work.

The 4th chapter presents a thorough validation study of the complete noise prediction method, providing comparisons with experimental data referring to rotor aerodynamic loading, acoustic pressure and ground noise impact.

The 5th chapter contains a systematic analysis examining the impact of uncertainty in aerodynamic modelling on the aeroacoustic analysis of helicopter rotors.

The 6th chapter focuses on large-scale mission-level simulation of rotorcraft noise, based on a novel method which alleviates the need of pre-stored acoustic data.

The 7th chapter discusses a generalised multi-disciplinary optimisation of novel VRS and ABT helicopter concepts, targeting the concurrent reduction of pollutant and noise emissions.

The 8th chapter performs an in-depth analysis of the aeroacoustics of optimised VRS and ABT helicopter concepts operating in descent trajectories in the vicinity of airports.

Finally, the 9th chapter discusses the major findings obtained from this research work, whilst performs an evaluation of the achievements and contribution to knowledge. A set of suggestions for future research and expansion of current capabilities is also provided.

Appendix A includes a list of all journal articles and conference papers produced within the period of this work.

2 Literature Review

2.1 Introduction

The literature survey undertaken for the purposes of this work is focused on modelling of helicopter noise generation and propagation mechanisms. A classification of distinct aerodynamic noise components is attempted. A critical review of state-of-the-art modelling methods is provided to be used for the identification and development of appropriate aerodynamic and aeroacoustic methodology. This should principally fit the purpose of helicopter mission simulation, uncertainty analysis and multi-disciplinary optimisation.

A thorough review on approaches for uncertainty quantification in rotorcraft simulation is carried out with emphasis on non-deterministic methods for helicopter rotor noise prediction. The necessity of uncertainty analysis towards improving noise predictive accuracy and robustness is discussed. Furthermore, the proposed methods in open literature regarding noise prediction across helicopter trajectories are investigated. The approaches followed to predict source noise and far-field propagation are identified and the demonstrated flight paths are assessed.

Finally, a comparison between morphing rotor technologies is performed and their potential use towards active helicopter noise reduction is explored. The advantages and limitations of each technology are evaluated, aiming to achieve the best compromise between performance, environmental impact, acoustic impact and applicability.

The state-of-the-art regarding all the aforementioned topics is discussed and the gaps in existing literature are identified. This chapter aims to pave the way for development of a novel methodology capable of addressing the objectives set to achieve the overall scope of this work, as well as, generate findings that can substantially contribute to the scientific knowledge in the broad field of rotorcraft aeroacoustics.

2.2 Helicopter Rotor Noise Prediction

2.2.1 Classification of helicopter noise

Helicopter rotors operate in a highly complex and fundamentally unsteady aeromechanical environment. Hover is the most unstable flight state, requiring constant pilot control inputs to maintain rotorcraft position. In forward flight, the advancing side of the main rotor encounters high, even transonic, tangential velocity in the blade tip region, whilst the retreating side suffers from blade stall. Take-off and landing manoeuvres include rotorcraft acceleration and deceleration terms which lead to dynamic wake distortion and non-periodic aerodynamic forces on rotor blades. In descent, the main rotor typically operates into its own wake, giving rise to highly complex aerodynamic interaction phenomena.

Helicopter main rotors feature high aspect-ratio blades, which make them sensitive to unsteady aeroelastic effects, leading to oscillatory structural blade loads. Additionally, vibrations and dynamic loads are introduced by the engine and transmission system to the main rotor hub. The wake of the main rotor is dominated by strong tip-vortices which have substantial impact on blade aerodynamic loading. Finally, the presence of fuselage close to the main rotor causes time-varying aerodynamic interactions due to (i) impingement of the main rotor wake on the fuselage body and (ii) flowfield induced from the fuselage in the vicinity of the rotor, especially in high-speed flight. Interaction and counteraction between the above complex aeroelastic and aerodynamic effects result in low- and high-frequency blade motion, as well as unsteady structural and aerodynamic loads.

Considering noise as static pressure deviation from its ambient value, it can be concluded that all the aforementioned unsteady phenomena are potential sources of aerodynamic noise. Depending on the contextual aerodynamic source, helicopter noise can be of different nature and have distinct signal characteristics. Hence, a classification of the potential noise sources is deemed as prerequisite towards identifying the appropriate modelling techniques for each noise source.

Helicopter rotor noise can principally be classified as either discrete frequency (deterministic) or broadband (non-deterministic). Discrete frequency noise is divided into thickness noise, loading noise and high-speed impulsive noise. Loading noise comprises both low-frequency and high-frequency components, depending on the associated aerodynamic loading components. Thickness and low-frequency loading noise are usually referred together as “rotational noise”. Broadband noise consists of turbulence ingestion noise, blade-wake interaction noise and blade self-noise (Ref. [14]).

Thickness noise is generated from the fluid displacement caused by solid blade body motion in the quiescent acoustic medium. It is dependent upon blade geometry and kinematics, local fluid velocity and density. Thickness noise is primarily directed in the rotor tip-path plane. Loading noise is generated from the accelerating force acting on the fluid by the solid blade surface motion (Ref. [14]). Hence, it is dependent upon the unsteady surface static pressure distribution which generates the aerodynamic forces. The term “loading noise” usually refers to the low-harmonic content of loading-induced noise, excluding impulsive loading sources. Both thickness and low-harmonic loading noise are non-impulsive and propagate at frequencies close to the blade passage frequency (BPF) (Ref. [15]).

Blade-vortex-interaction (BVI) noise is a special subcase of loading noise which occurs at higher frequencies than low-harmonic noise. It is a result of the interaction between rotor blades and wake tip-vortices. The tip-vortices are the most noticeable part of helicopter rotor wake and are strongly influenced by the three-dimensional wake roll-up. BVI noise is of highly impulsive nature considering that it is generated from unsteady pressure fluctuations induced by the close proximity between tip-vortices and rotor blades (Ref. [16]). It propagates at intermediate frequencies of the helicopter noise spectrum, where human ear is most sensitive. It usually occurs during descending low-speed flight, where tip-vortices tend to lie closer to the main rotor (Ref. [17]). For these reasons, BVI noise is the most annoying noise source for communities,

therefore attracting great attention from research associations and regulatory bodies.

High-speed impulsive (HSI) noise is associated with high intensity transonic effects and shock waves in the chordwise blade direction. It only occurs during high-speed forward flight, when the supersonic flow region of the advancing rotor blade extends off the blade into the far field (Ref. [18]). HSI noise has in-plane directivity, mainly towards the streamwise direction. In civil rotorcraft aviation, it is usually avoided to operate at very high flight speeds which might result in transonic blade tip Mach numbers; hence emission of HSI noise is limited.

Broadband noise includes all the non-deterministic components of loading noise related to random pressure fluctuations generated on the blade surface (Ref. [19]). Turbulence ingestion occurs from shear flow within the wake or even naturally from the atmosphere. Interaction between wake turbulence and following blades generates mid-frequency broadband noise (Ref. [14]). Self-noise is usually of high-frequency and produced by boundary layer turbulence and trailing edge turbulence shedding (Ref. [20]).

The aforementioned noise generation mechanisms refer to the main rotor, which is the major noise source of conventional helicopters. Similar phenomena describe noise generated by the tail rotor, albeit shifted at higher frequencies, due to the higher rotational speed. Other noise sources, such as engine noise or transmission noise are of lower intensity and broadband nature. Nevertheless, high-frequency broadband noise of low intensity is easily attenuated with distance.

A computational framework for helicopter aeroacoustic simulation would ideally require efficient aerodynamic and aeroacoustic modelling of all the aforementioned noise components. Specifically, individual noise models complying with the above classification would be required for the assessment of operational acoustic footprint, as well as design optimisation focusing on the reduction of specific noise components.

2.2.2 Helicopter rotor aerodynamic modelling

In light of the aforementioned noise generation mechanisms, it becomes clear that steady as well as unsteady aerodynamic terms need to be captured for the accurate representation of time-varying blade airloads for aeroacoustic applications. For this reason, steady aerodynamic modelling methods will not be discussed in the context of this research. Two basic types of unsteady aerodynamic modelling approaches have been primarily used in helicopter applications: Computational fluid dynamics (CFD) methods and analytical methods.

CFD-based methods have provided insight into the roll-up of helicopter rotor tip-vortices (Ref. [21]). CFD methods have also been used for the prediction of flowfield around a helicopter rotor in low- and high-speed forward flight (Ref. [22]). Despite the progress made in CFD modelling of the rotor flowfield, most of current methods are facing limitations in the prediction of helicopter rotor wake. The main reason for this inability is the introduction of artificial diffusion of wake vorticity, caused by the numerical error in the approximations of Euler or Navier-Stokes equations. As a result, the coherence of tip-vortex is deteriorated due to numerical diffusion within the first wake revolution (Ref. [17]), although three or more revolutions are required for the accurate capture of BVIs across the rotor disc. Additionally, CFD-based models would require significantly fine temporal and spatial resolution to accurately capture the high-frequency noise-generating phenomena. That would result in extensively high computational times, rendering the use of such solvers prohibitive for mission simulation or optimisation applications.

Analytical unsteady aerodynamic models constitute a computationally-efficient approach for the prediction of time-varying forces acting on helicopter rotor blades. A semi-empirical model for the prediction of unsteady aerofoil behaviour was developed by ONERA (Ref. [23]) and modified by Peters (Ref. [24]). The airloads within the ONERA model were expressed as a combination of linear behaviour of attached flow and incremental deviations arising from dynamic stall. The drawback of this model is that it is based on many empirical

coefficients which are determined by parameter identification from experimental data. An improved analytical model for unsteady aerodynamics was developed by Leishman and Beddoes (Ref. [25]). The model was an implementation of the Indicial Response Method (Ref. [17]), able to cater for attached flow, trailing edge separation and vortex shedding, using only a limited number of empirical coefficients. The Leishman-Beddoes model emphasises the complete physical representation of unsteady aerodynamic aerofoil behaviour, accompanied with limited computational overheads; hence it has been used in a variety of helicopter comprehensive codes (Refs. [26]–[28]).

Analytical unsteady aerodynamic models can be implemented in any helicopter comprehensive code using blade-element-type discretisation of the rotor blades. Nevertheless, they essentially need to be coupled with a wake model able to represent the behaviour of discrete flow structures, i.e. blade tip-vortices, in order to capture BVI events. Two main types of analytical vortex wake models have been identified in the open literature: (a) prescribed wake models and (b) free-wake models.

Prescribed wake models estimate the wake geometry utilising expressions derived from experimental data and/or momentum considerations. The induced velocity from the vortex filaments on the rotor is calculated by applying the Biot-Savart law to all vortex filaments. Due to their low computational overhead, prescribed wake models have found use in helicopter aerodynamic predictions for aeroacoustic purposes (Refs. [27], [29]). However, the lack of generality is a fundamental drawback of prescribed wake models, as their predictions are subject to the availability of data for wake geometry prescription. Additionally, self or mutual interactions between wake vortices are not accounted for, resulting in exclusion of unsteady behaviour within the wake.

Free-wake models solve for the vortex strengths and rotor wake geometry without prescriptions (Ref. [17]). The positions of individual vortex elements are part of the solution process and determined by the free-stream, blade unsteady loading and interactions with the other wake vortex filaments. In this manner, a physics-based solution, representative of each unique flight state can be

accrued for the aerodynamics of helicopter rotor blades. Free-wake models offer the best compromise between prescribed wake models and CFD in terms of computational time requirements. Nevertheless, free-wake models, as well as prescribed wake models, require a well-calibrated vortex core diffusion model to avoid singularities in the application of Biot-Savart law for induced velocity calculations. Vortex core models rely on semi-empirical parameters for the vortex core growth, typically based on experimental evidence or empiricism. This constitutes a potential source of uncertainty in aerodynamic modelling which essentially propagates in aeroacoustics calculations. Bhagwat and Leishman investigated the impact of vortex core model on rotor airloads prediction (Ref. [30]). Empirical guidelines were provided for the choice of viscous core growth parameters, targeting the best match between predicted and measured axial and swirl velocities for small-scale rotors in hover and forward flight.

Free-wake models have been extensively used for demanding BVI aerodynamic predictions with applications in aeroacoustics. Lim *et al.* (Ref. [31]) compared a single-trailer with a multiple-trailer free-wake model in terms of BVI airloads prediction capability. The rotor employed for the comparisons was the HART II rotor (Ref. [32]) which is a 40% scale model of the Bo-105 helicopter rotor. For the baseline (BL) case, it was observed that resolution of all trailed vortices improves the airloads prediction accuracy compared with tip-vortex roll-up wake models. On the other hand, for the minimum noise (MN) and minimum vibration (MV) cases of the experiment, no particular improvement was found from the use of multiple-trailer wake models.

A comprehensive model was implemented in Ref. [33] to investigate the potential of noise and vibration reduction using actively controlled flaps. The model incorporated a planar detection method for identifying potential BVIs. Specifically, a pre-specified fictitious plane, aligned on the blade with direction perpendicular to the rotor plane was assumed. When a tip-vortex filament intersected the fictitious plane within a specified distance from the blade, a BVI event was recorded. However, Gandhi and Tauszig (Ref. [34]) proved that the

plane detection method required extremely small wake azimuthal step size, and even so had difficulty in detecting the most parallel BVIs. Despite that, they introduced a spherical detection method, where BVI locations were detected via the direct calculation of miss-distance among blade control points and vortex segment mid-points. A potential BVI event was recorded when the miss-distance was lower than a pre-defined tolerance. They concluded that the spherical detection method was able to identify parallel BVIs with a less demanding wake azimuthal resolution.

A free-wake aerodynamic model was employed in Ref. [35] for the prediction of BVI airloads for the purposes of noise abatement flight procedures. The wake model was an implementation of three-dimensional panel method with full-span free-wake calculation. The same model was previously used for modelling main-tail rotor interaction aerodynamics (Ref. [36]). Nevertheless, it was found that the effect of tail rotor induction on the main rotor is insignificant. Chen *et al.* (Ref. [37]) employed a free-wake model to investigate the impact of transient manoeuvres on wake geometry, airloads and noise. Blade tip-vortices are the dominant structures in the wake flowfield; hence the approach allowed the free evolution of tip-vortex filaments, whilst near wake was treated with a prescribed geometry approach.

Gennaretti *et al.* (Refs. [38], [39]) developed a panel boundary integral method which used Rankine vortices instead of doublet distributions to eliminate numerical instabilities arising when wake panels impinge on the blade surface. The aerodynamic model provided input to aeroacoustic formulations towards the aeroacoustic characterisation of helicopters (Ref. [40]). The same comprehensive code was successfully utilised in the assessment of computational methodologies for BVI noise prediction. Satisfactory correlation was observed between calculated and measured data referring to the HART II experiment (Ref. [32]) in terms of trim performance, blade motion, airloads and noise. It was shown that the inclusion of fuselage-induced velocity perturbation on the main rotor improves airloads predictive accuracy. Additionally, a sensitivity analysis was carried out, which showed that the wake model is

sensitive to vortex core diffusion empirical coefficients. However, the analysis was a simplified preliminary investigation, based on the assumption of $\pm 50\%$ variation of wake diffusion parameters. A more detailed analysis of the effects of wake modelling uncertainty on the prediction of airloads and subsequently, noise, would shed light on the robustness of analytical comprehensive modelling of helicopter rotors.

The aforementioned analysis suggests that analytical aerodynamic formulations based on free-wake inflow models constitute the most suitable tool for the prediction of unsteady airloads including BVI phenomena. Coupling of these methods with appropriate aeroacoustic models will enable the efficient helicopter rotor noise prediction. Ultimately, such a combined approach would allow the incorporation of aeroacoustics into mission simulation and rotor design optimisation processes.

2.2.3 Helicopter rotor aeroacoustic modelling

The complex aeromechanical environment of helicopter rotors gives rise to multiple noise generation mechanisms. Helicopter rotor noise comprises distinct components, classified according to the fundamental sources of aerodynamic unsteadiness. Analytical models are capable of predicting the unsteady motion and airloads of elastic rotor blades, arising from the governing aeromechanical phenomena. In this context, analytical aeroacoustic formulations, able to be coupled with the aerodynamic models, are essential to close the problem of helicopter rotor noise prediction.

Lighthill was the first who recognised that, away from the region of agitated jet flow, the propagation of the perturbations of flow quantities is governed by the three-dimensional mechanical wave equation (Ref. [41]). In this context, the original continuity and Navier Stokes equations were re-arranged in order to form an inhomogeneous wave equation for the perturbation of the density of the acoustic medium. Mathematically, the Acoustic Analogy (Ref. [42]) determines equivalent acoustic sources in the quiescent medium, based on the knowledge of flow quantities in the region of noise generation.

Ffowcs Williams and Hawkings (Ref. [43]) extended the Acoustic Analogy to include arbitrary convective source motion. The moving source was represented as an acoustic data surface (ADS) with known flow quantities across it. In this way, an artificial discontinuity was introduced in the quiescent medium which was addressed through the use of generalised functions in the mathematical manipulation of mass and momentum conservation equations. The outcome was the pioneering Ffowcs Williams-Hawkings (FW-H) equation for aerodynamic noise propagation (Refs. [14], [43]). The equation describes the propagation of density perturbation in the form of three-dimensional wave as a summary of monopole, dipole and quadrupole source terms. The acoustic monopole terms are first-order temporal derivatives and correspond to thickness noise. The acoustic dipole terms are first-order spatial derivatives and represent loading noise. Finally, the acoustic quadrupoles are second-order spatial derivatives and describe HSI noise. The FW-H equation is fundamentally exact in nature which allows its general use in stationary or moving acoustic source problems.

The original derivation of the FW-H equation requires impermeable-surface-type of aerodynamic input across the solid blade surfaces for the calculation of thickness and loading noise, whilst knowledge of the flowfield within a volume surrounding the blade is needed for capturing HSI noise components (Ref. [44]). Therefore, the placement of the ADS is critical for the acoustic content to be captured by the solution of FW-H equation. In the case of coincidence between ADS and actual blade surface, only the source terms located across the blade surface are considered, which are the acoustic monopoles and dipoles. Hence thickness and loading noise are efficiently resolved using impermeable ADS, but HSI noise components are neglected. An alternative version of the FW-H equation was developed by di Franciscantonio (Ref. [44]), which employs a permeable ADS enclosing the solid blade. The potential advantage of the permeable, or porous, ADS is that it can capture HSI noise when the aerodynamic solver is capable of capturing non-linear phenomena associated with transonic or supersonic flow around the blade surface. Therefore, the quadrupole term of the FW-H equation can be legitimately

neglected during the aeroacoustic calculations, without losing significant degree of accuracy in HSI noise predictions (Ref. [45]). In this case, CFD aerodynamic solutions would be required inside the ADS, which would significantly increase the computational cost of the aerodynamic computations. However, it is again noted that emission of HSI noise is usually limited in civil rotorcraft missions, since operation in the transonic advancing blade region is typically avoided.

Numerical solution of the FW-H equation would be computationally inefficient given the size of computational domain, as well as, the spatial and temporal grid resolution required to capture the high-frequency acoustic phenomena. Instead, integral solutions of the FW-H equation can be derived based on the free-space Green's function for the three-dimensional inhomogeneous wave equation. Farassat (Ref. [46]) was among the first who proposed theoretical solutions of the FW-H equation to predict the far-field noise produced by rigid bodies in arbitrary motion. In Ref. [47], a formal expression for the integral solution of the FW-H equation was presented. Specifically, Farassat recognised that the wave operator in the FW-H equation is linear, which means that the temporal and spatial derivatives commute. This alleviated the complexity of time and space differentiation of the source terms, leading to the formal derivation of "Formulation 1 of Farassat". However, the formulation still required the calculation of time derivative of acoustic integral which was a source of computational overhead and numerical errors.

The speed and accuracy of noise predictions has been improved by eliminating the time derivative of the integral of Formulation 1 (Ref. [14]). Specifically, Farassat expressed the observer time derivative as function of source time derivative, based on the fact that noise radiation distance is a function of observer and source time. In this manner, Farassat managed to move the time derivative inside the acoustic integral, resulting in improvement of the efficiency and robustness of noise predictions. The improved formulation was named "Formulation 1A of Farassat" (Ref. [48]) and is capable of predicting rotor thickness and loading noise, as quadrupole terms were excluded in the original derivation. A computer program named RAPP (Rotor Acoustic Prediction

computer Program), developed by NASA, was one of the first attempts to model aerodynamic noise generated by hovering helicopter rotors using Formulation 1 (Ref. [49]). NASA WOPWOP (Ref. [50]) is one of the most widely used and validated code for the prediction of helicopter thickness and loading noise, based on Formulation 1A of Farassat. The code required time-accurate rotor surface pressures and their temporal derivatives for loading noise prediction (Ref. [51]).

Schmitz and Yu (Refs. [52], [53]) demonstrated the role of the quadrupole source term in the FW-H equation. The effect of the quadrupole field was replaced by a distribution of singularities on the mean blade surface. It was shown that this simplification was leading to significant underprediction of acoustic pressure at in-plane observers at high tip Mach number. Brentner *et al.* (Refs. [54], [55]) derived a collapsing sphere formulation for subsonic quadrupole noise prediction in the rotor tip-path plane, named as “Formulation Q1A”. This has been implemented in the WOPWOP+ software, a new version of WOPWOP. Farassat and Brentner (Ref. [18]) extended the quadrupole noise theory to supersonic regimes, deriving an emission-surface formulation, named as “Formulation Q2”. A new demonstration code has been developed (WOPWOP2+), which uses emission-surface models for both rotational and HSI noise prediction. Both quadrupole formulations were robust and accurate in predicting HSI noise. Nevertheless, the computational time required for the aerodynamic calculations in the vicinity of the rotor is prohibitive for routine noise evaluations, especially at mission level.

Ianniello (Ref. [56]) proposed a supersonic quadrupole noise prediction technique, using an emission-porous surface formulation. The proposed supersonic formulations were based on the modification of the original monopole and dipole equations of Formulation 1A, utilising the permeable surface implementation of di Francescantonio (Ref. [44]). The aforementioned studies achieved the prediction of subsonic and supersonic HSI noise. Nevertheless, they have rarely been used in routine aeroacoustic calculations such as complete missions or optimisation, due to the high computational cost

associated with the aerodynamic calculations required to capture transonic flow phenomena. Additionally, HSI noise is typically omitted in civil rotorcraft aeroacoustic analysis, as operation near the transonic tip Mach number limit it is commonly avoided.

Marcolini *et al.* (Ref. [57]) achieved the prediction of BVI noise utilising the WOPWOP software and measured aerodynamic data gathered from the Duits-Nederlandse Windtunnel (DNW). Brentner *et al.* (Ref. [58]) investigated the sensitivity of acoustic predictions to variations of input parameters, using experimental data from Ref. [57] for the evaluation of accuracy of predictions. Both the aforementioned works concluded that, theoretical noise modelling based on Formulation 1A has a high degree of maturity; however, measured airloads were required as prerequisite.

Advances in unsteady aerodynamics (Ref. [25]) and wake modelling (Ref. [59]) enabled the prediction of BVI events in the vicinity of the rotor. Bernardini *et al.* (Ref. [60]) coupled an aeroelastic steady periodic solver with a three-dimensional free-wake aerodynamic model and a boundary integral model based on Formulation 1A to achieve BVI noise prediction. Gennaretti *et al.* (Ref. [40]) extended this work to numerically characterise helicopter noise hemispheres. The method was limited to descending flight conditions and was applied on demonstrative short-duration trajectories. Brès *et al.* (Ref. [11]) developed a manoeuvring noise prediction code named “PSU-WOPWOP”, whilst Shirey *et al.* (Ref. [61]) extensively validated it. The code was an implementation of the Formulation 1A of Farassat, featuring a compact expression for dipole noise sources (Ref. [62]).

The compact loading approximation reduced the aerodynamic input requirements from surface to line integral, rendering the code capable of direct coupling with lifting-line-based aerodynamic input. The employment of chordwise compact aeroacoustic formulations considerably reduced the computational requirements without significant loss of accuracy up to the high-harmonics of BVI noise [62]; hence it adequately captured the deterministic part of the noise spectrum of typical helicopter rotor designs. PSU-WOPWOP does

not feature compact formulations for thickness noise prediction which would further reduce the associated computational requirements. Romani and Casalino (Ref. [63]) coupled a Lattice-Boltzmann solver with Formulation 1A to model helicopter BVI noise. Although the method was efficient in the prediction of BVI noise, prescribed blade deformations, derived from experimental data, were required (Ref. [64]) to capture the miss-distance between the blades and trailed vortices. The alternative would be to couple the Lattice-Boltzmann solver with a Computational Structural Dynamics (CSD) solver, but this would become excessively expensive in terms of computational requirements. This is the major disadvantage of this method compared to comprehensive codes which typically include physics-based analytical models for the prediction of elastic blade motion (Refs. [65]–[67]).

Brooks and Burley (Refs. [68], [69]) introduced and validated a semi-empirical method for the prediction of broadband noise generated from blade-wake interactions and self-noise. However, the method required measured blade pressures as input which limited its applicability and generality. Farassat and Casper (Refs. [20], [70], [71]) developed “Formulation 2B” for the prediction of broadband noise, after modifying the original Formulation 1A. The drawback of this method was the requirement of extensive turbulence simulation which is prohibitive for the purposes of mission analysis. Pegg (Ref. [72]) proposed a semi-empirical method for broadband noise prediction in the preliminary stages of rotor design. The method only required high-level aerodynamic inputs, easily obtainable by any comprehensive code. For this reason the method found application in a virtual certification study (Ref. [73]) where it was shown that it accounts for the deficiency in ground noise predictions which was observed when broadband noise was excluded from the predictions.

2.3 Uncertainty in Rotorcraft Aeroacoustic Simulation

Comprehensive models are routinely used for the prediction of aerodynamic, aeroelastic and aeroacoustic response of complex rotorcraft systems. However, many input parameters associated with the aforementioned disciplines are accompanied with inherent uncertainty. Uncertainty can be found in structural

blade properties, rotor geometric parameters, aerodynamic properties of aerofoils, wake vortex core diffusion properties and operational or ambient conditions. Regardless of nature, uncertainty essentially propagates into the aeroacoustics calculations. Consequently, non-deterministic evaluation of rotorcraft noise constitutes a step towards improving the robustness of aeroacoustic predictions. This step effectively contributes towards enhanced modelling capabilities through mitigating the risk of under-predicted acoustic impact of modern rotorcraft simulation.

Murugan *et al.* (Ref. [74]) investigated the effect of uncertainty in material properties on the aeroelastic response of composite rotor blades. The stochastic analysis was carried out through Monte Carlo simulation which required 6,000 samples for convergence. It was shown that in forward flight, the torsional response presented considerable deviations from the corresponding deterministic solution, which essentially affected airloads, stability and control. Refs. [75], [76] presented a scattering of -7 to +4% in the value of the vibration index typically used in aeroelastic optimisation studies. They both concluded that the randomness in material properties should be considered in the aeroelastic analysis, design and optimisation of rotorcraft.

Siva *et al.* (Refs. [77], [78]) quantified the impact of blade geometrical and aerodynamic uncertainty on the performance of helicopter rotors. The uncertainties were implemented through random variations of rotor blade chord and radius, aerofoil lift-curve slope, blade profile drag coefficient and blade pitch and twist angles. A Monte Carlo scheme was employed for the uncertainty analysis requiring 100,000 samples to achieve convergence of the thrust and power coefficient of the main rotor. The required number of deterministic evaluations did not permit the use of a sophisticated aerodynamic model: the rotor model featured linear aerofoil aerodynamics, without accounting for stall, unsteady effects or elastic blade motion. It was shown that, the performance of untwisted blades presented smaller output uncertainty compared to linearly twisted blades. It was also shown that, in forward flight, power coefficient

presents an uncertainty of 5.32%, which should be considered in the selection of powerplant for new helicopter designs.

A benchmark rod-aerofoil configuration was assessed in terms of geometric uncertainty and its propagation in aeroacoustics (Ref. [79]). A Large-Eddy Simulation (LES) solver was coupled with a FW-H solver within a non-intrusive stochastic collocation (SC) method (Ref. [80]). The SC method relies upon Lagrange interpolating polynomials for the generation of a response surface model of the uncertain output response, utilising the deterministic model solutions at specific collocation points. The SC method required orders of magnitude less deterministic evaluations compared to traditional Monte Carlo approaches for uncertainty quantification (Ref. [81]). It was demonstrated that an uncertainty of ± 0.004 m in the position of the rod led to an uncertainty of 0.5-1 dB in the Overall Sound Pressure Levels (Ref. [79]).

A robust optimisation approach was presented in Ref. [82], which aimed at minimum aerofoil drag coefficient through morphing camber line. The variables considered as random were the angle of attack and Mach number encountered by the two-dimensional aerofoil. The quantification of uncertainty was implemented via a Polynomial Chaos (PC) expansion method (Ref. [83]). PC expansion is a non-intrusive and non-interpolating stochastic method, which estimates the uncertain model response by utilising orthogonal polynomials to construct a polynomial response surface function. SC and PC present many similarities as they have both demonstrated superior computational efficiency relative to traditional Monte Carlo approaches (Ref. [84]). A practical difference is that SC requires a structured set of collocation points, whilst PC is more flexible. The selection of collocation points can allow greater simulation fault tolerance (Ref. [84]). The Non-dominated Sorting Genetic Algorithm (Ref. [85]) was employed for the concurrent minimisation of mean value and standard deviation of the aerofoil drag coefficient. Improvements of 1% and 4% were achieved for the rotor advancing and retreating side, respectively.

Olsman (Ref. [86]) quantified the uncertainty associated with the ground acoustic footprint of noise abatement flight procedures in the presence of

randomness in the operational flight path. A noise hemisphere approach was employed for the prediction of noise footprint and incorporated into a PC expansion based framework for uncertainty quantification. Standard deviations of up to 3 dB were calculated with respect to the deterministic solution, which dictates the potential importance of non-deterministic noise footprint prediction in helicopter mission planning.

Ricks *et al.* (Refs. [87], [88]) investigated the impact of operational and geometrical uncertainty in the acoustic propagation stage of hybrid computational aeroacoustics. A CFD solver based on the inviscid Euler equations was coupled with an integral solution of the FW-H equation for the prediction of noise generation and propagation. A non-lifting helicopter rotor in hover was considered and an in-plane noise observer located 3.09 rotor radii upstream of the rotor. PC expansion was employed for the quantification of uncertainty and sensitivity analysis using Sobol indices (Ref. [89]). Uncertainty of 7.36% was observed in the peak value of acoustic pressure, primarily attributed to the randomness in blade tip Mach number. The investigations of this study were carried out only for in-plane observers, which are dominated by thickness noise.

With respect to the aforementioned research in the existing literature on non-deterministic analysis, it becomes clear that the quantification of the impact of modelling uncertainty on aeroacoustic predictions should essentially be considered towards improving the robustness of noise evaluations. The aforementioned studies have quantified the uncertainty related to structural blade properties, rotor geometric parameters, aerodynamic properties of aerofoils, and operational or ambient conditions, primarily employing CFD or simplified steady aerodynamic approaches. Nevertheless, as discussed in section 2.1.2, free-wake models are routinely used in aeroacoustic analysis of helicopter rotors. Despite their efficiency in predicting complex aerodynamic phenomena such as BVIs, the semi-empiricism associated with vortex core diffusion parameters is essentially accompanied with uncertainty which affects airloads and subsequently, noise.

Bhagwat and Ramasamy (Ref. [90]) examined the effect of tip-vortex aperiodicity on the measurement uncertainty of vortex-core properties. Uncertainties in the order of 10% were calculated for the vortex core radius of the HART II rotor (Ref. [32]) in BVI conditions. These levels of uncertainty are representative of the state-of-the-art in tip-vortex measurements, although they have never been incorporated into helicopter rotor aeroacoustic simulation. In some cases, robust experimental evidence referring to tip-vortex core radius is not available, forcing analysts to resort to empiricism for the determination of the associated physical properties necessary for aerodynamic calculations. The vortex core radius is a critical aerodynamic parameter which essentially affects airloads and consequently, noise. However, the impact of tip-vortex core modelling uncertainty on the robustness of aeroacoustic predictions has not yet been systematically examined. Further investigations for acoustic observers dominated by BVI noise would shed light on the robustness of advanced aeroacoustic modelling of helicopter rotors.

2.4 Acoustic Impact of Helicopter Trajectories

Rotorcraft operations are governed by strong variations in flight speed, flight path angle, altitude and gross weight. These variations are essentially reflected on the acoustic ground footprint throughout the mission-time. Hence, generalized methodologies, capable of resolving the entire flight trajectory are required to address the noise prediction problem from a holistic perspective. Present ground footprint prediction codes mostly rely on source noise databases given in terms of acoustic disturbance over hemispheres surrounding the helicopter. Noise hemispheres are typically provided by either computational or experimental campaigns (Ref. [8]). The majority of existing mission simulation methods do not directly generate acoustic spheres corresponding to each and every mission element. Instead, two-dimensional interpolation is typically required among a set of available hemispheres to better match the operating conditions of each trajectory waypoint. Acoustic prediction methods based on pre-stored noise hemispheres have been employed within flight

procedure impact assessment (Ref. [9]), as well as mission trajectory optimization (Ref. [91]).

Schmitz *et al.* (Refs. [10], [92]) introduced a quasi-static acoustic mapping method to predict BVI noise in manoeuvring flight. The term of rotorcraft acceleration parallel to the flight path was included in the calculation of rotor tip-path-plane angle. For each waypoint of the manoeuvring trajectory, a radiation sphere which corresponded to an equivalent trimmed state was acquired by interpolating among a pre-calculated tabulated set of acoustic spheres. The governing parameters selected for the interpolation were the advance ratio and tip-path-plane angle. Rotor thrust coefficient was not directly included in the set of governing parameters, although it is a representative descriptor of rotor disc loading which fundamentally affects loading noise. Greenwood (Ref. [93]) included rotor thrust coefficient in the set of operating parameters which govern the generation of the acoustic sphere databases. A quasi-static method was developed to predict ground noise impact for short-duration demonstrative manoeuvres. The method neglected rotor broadband noise which is a potential source of disturbance for out-of-plane ground observers.

In Ref. [94], the GENHEL flight dynamics code was coupled with the PSU-WOPWOP noise prediction code and applied on low altitude demonstrative manoeuvres. The inflow model employed in GENHEL did not treat discrete flow structures such as tip-vortices; hence it was not able to capture BVIs. After including a free-vortex wake model (Ref. [37]), it was proven that BVI, when it occurs, is the dominant noise source; however, its prediction comes with high computational expense: up to 50 rotor revolutions were required, even for short-duration manoeuvres. Yin and Buchholz (Ref. [35]) employed integral solutions of the Acoustic Analogy to generate noise hemispheres, utilising unsteady free-wake aerodynamic input. The integrated methodology was deployed for the purposes of noise abatement procedure design. Specifically, it was applied to simplified hypothetical flyovers, rather than complete helicopter mission trajectories. Gennaretti *et al.* (Refs. [40], [95]) introduced a hemisphere-based approach to predict the acoustic footprint of helicopter unsteady manoeuvres. A

database of noise spheres was developed, relied on steady flight aeroelastic, aerodynamic and aeroacoustic predictions. The method capabilities were demonstrated for a representative approach trajectory.

In the aforementioned studies, the acoustic footprint of helicopters was evaluated for demonstrative, short-range and short-duration flight procedures. Predictions were almost entirely based on pre-stored noise hemisphere data-sets, corresponding to a finite number of flight states, representative only of the designated vehicle. However, a “real-world” helicopter mission can have such duration and range that hundreds of square kilometres can be affected for hundreds of minutes time. Along their three-dimensional trajectory, rotorcraft encounter notable variations in flight speed, flight path angle, altitude, fuel weight and payload (Ref. [96]). In this context, trim control angles, fuselage attitude angles, as well as rotor blade kinematics and loading are continuously varying with time. These fluctuations are directly reflected on the ground noise impact. Nevertheless, the applicability of generalised and physics-based methods on the prediction of acoustic footprint of complete helicopter missions and without any pre-stored noise data has not yet been investigated.

2.5 Morphing Rotor Helicopter Concepts

Helicopter noise reduction has emerged a high priority for the aerospace sector. The most efficient way to reduce helicopter noise emission is through mitigating adverse aerodynamic effects at the source, i.e. in the vicinity of the main rotor. Passive noise reduction methods have been proposed (Ref. [97]), primarily implemented through blade tip design modification or modulated blade spacing (Ref. [13]). Despite the significant noise reductions observed, the proposed passive concepts came with performance penalties or did not offer noise benefits across the entire flight envelope. The diverse nature of helicopter flight states across “real-world” missions dictates that active rotor noise reduction technologies have the potential to collectively mitigate the acoustic impact and improve performance of helicopter rotors.

Variable rotor speed (VRS) has been proven to be an efficient method for engine performance enhancement across the entire flight envelope (Refs. [98],

[99]). Conventional helicopter rotors operate at constant rotational speeds. Nevertheless, there is a potential of trimming the helicopter at rotational speeds different to the nominal, whilst simultaneously achieving considerable performance benefits. The rotational speed of the main rotor can be controlled through governing the engine's Full Authority Control System (FADEC) by adjusting the fuel flow. This effectively determines the rotational speed of the free-power turbine (FPT) and main rotor.

Active rotor morphing (Ref. [100]) aims to mitigate adverse effects associated with rotor aeromechanics through the modification of blade geometry. Various technologies have been tested for active rotor morphing; variable rotor radius (Ref. [101]), variable blade chord (Ref. [102]) and variable blade pitch (Ref. [12]). Variable chord or radius offer great performance benefits from a theoretical perspective, however the increased complexity associated with their implementation prevents their practical utilisation. Variable blade pitch (Ref. [103]) can be implemented either individually, through individual blade control (IBC), or collectively, through higher harmonic control (HHC) or active blade twist (ABT).

Variable rotor speed and variable blade pitch constitute two concepts with potential for improving helicopter rotor performance across a wide range of the flight envelope. Ultimately, the combination of the two aforementioned concepts would offer radical re-distribution of unsteady airloads across the rotor disc. This fundamentally affects aerodynamic noise generation; hence the investigation of combined rotor-speed and pitch variation from a noise perspective would shed light on the aeroacoustics of modern morphing-rotor helicopters.

2.5.1 Variable rotor speed

Steiner (Ref. [104]) investigated the performance benefits and trim requirements arising from the implementation of variable speed on a UH-60 helicopter. The results showed 12% and 17% reduction of power requirement in hover and cruise near 60 m/s, respectively. Furthermore, it was shown that at sea-level conditions, reductions in power requirement can be achieved across the entire range of attainable airspeeds. Bowen-Davies and Chopra (Ref. [105])

investigated the impact of main rotor speed reduction on the aeromechanics and performance of a UH-60A helicopter in steady level flight. For low-thrust requirements, power reductions of up to 20% were observed at low-speed cruise with a rotor speed of 76% of its nominal value.

Mistry and Gandhi (Ref. [101]) examined the impact of variable rotor speed combined with variable rotor radius, on the performance of a UH-60A helicopter. Rotor speed reduction alone offered power savings of up to 14% in low-and-light conditions. It was observed that, at low airspeeds, the effectiveness of rotor speed variation in improving performance diminishes as induced power is not significantly affected by rotor speed variation. The combination of rotor speed and radius variation led to power reductions of up to 30% in heavy-and-high conditions. In Ref. [106], the performance improvements after the implementation of variable-speed tail rotors on a UH-60A helicopter with variable-speed main rotors were explored. Main rotor speed reduction leads to main rotor performance enhancement, but increases anti-torque requirements for the tail rotor. It was shown that variable-speed tail rotors can allow larger speed reductions for the main rotor in hover. Reduction of tail rotor speed led to 1.4% reduction of total helicopter power requirement in cruise at 36 m/s. No performance benefits were observed in high-speed flight associated with variable-speed tail rotors.

Misté *et al.* (Ref. [107]) presented an approach for coupling helicopter-engine performance to assess the benefits arising from the implementation of variable rotor speed on a UH-60 helicopter equipped with a pair of GE T700 turboshaft engines. A methodology to determine the optimal rotor speed for minimising fuel consumption was proposed. Fuel savings of about 7-8% were obtained within a $\pm 15\%$ range of the nominal rotor speed.

With respect to the aforementioned literature, VRS helicopters offer considerable performance benefits across a wide range of the flight envelope, primarily attributed to the reduction of blade profile power requirement. The effects of rotor speed variation on rotor noise generation has not yet been investigated, although improvements would be expected due to the reduction of

the magnitude of blade local Mach vector in the direction of radiation. Rotor speed variation is inefficient in low speed flight, where induced power dominates. In this flight condition, variation of blade pitch angle would be efficient in mitigating adverse aerodynamic effects and further enhancing performance and reducing noise.

2.5.2 Active blade twist

The effects of HHC of blade pitch on helicopter rotor BVI noise and vibration were examined in Ref. [108]. A dynamically scaled, four-bladed, articulated rotor was tested in low-speed descent conditions. It was observed that 4/rev HHC achieved up to 5.6 dB BVI noise reduction, however, low-frequency noise was increased by 6 dB. Splettstoesser *et al.* (Refs. [109], [110]) conducted measurements on a 40% dynamically scaled Bo 105 model rotor, to quantify the impact of HHC on noise and vibration. Although HHC has demonstrated clear benefits in terms of BVI noise reduction in low-speed descent, its application came with concurrent increase in low-frequency harmonic noise which is generally not desirable. As discussed in Ref. [111], the introduction of HHC resulted in an increase in the excitation of the first torsional mode of vibration, due to the proximity of the first torsional moment and the rotor 3P frequency. The increased amplitude in torsional blade twist azimuthal variations yielded increased low-frequency amplitudes in elementary normal force azimuthal distributions which eventually led to the low-harmonic noise penalties reported in Refs. [109], [110]. Furthermore, limitations with respect to technical implementation of a blade control and actuation system for HHC variations has prevented the practical implementation of this concept (Ref. [112]).

Jacklin *et al.* (Ref. [113]) investigated the impact of IBC on power, vibrations and BVI noise of a Bo 105 helicopter rotor. Performance improvements of up to 7% were obtained using 2/rev IBC at high advance ratios. The use of multi-harmonic IBC input led to 85% simultaneous reductions in both BVI noise and hub vibrations. In Ref. [114] an IBC system was assessed in terms of BVI noise ground footprint reduction. Flight tests were conducted on a Bo 105 helicopter featuring a 2/rev IBC control strategy. BVI noise reductions of up to 5 dB were

measured for different slope angles of the descending helicopter. Despite the promising results obtained by both HHC and IBC configurations, none of these concepts found application on a helicopter, primarily due to the complexities associated with modifications required to the swashplate to realise their implementation (Ref. [115]).

Booth and Wilbur (Ref. [116]) emphasised on the acoustic aspects of rotor blade twist variation. The largest BVI noise reduction was 2.8 dB and was achieved through 5/rev variation of blade twist. However, low-frequency noise penalties of up to 7 dBs were observed at an advance ratio of 0.14. Zhang *et al.* (Ref. [117]) quantified the benefits of variable twist on the aerodynamics and performance of a Mach scaled Bo 105 helicopter rotor. Power savings of up to 14% were observed in forward flight with advance ratio 0.378. The impact of twist variation on the aerodynamics of a hovering rotor was investigated in Ref. [118]. Twist actuation frequencies of 1/rev and 3/rev were examined in terms of blade tip-vortex convection. The 1/rev variation led to a reduction of 18% in the initial vortex strength. Reductions of up to 35% were achieved in the magnitude of swirl velocity via 3/rev twist variation. The impact of active blade twist (ABT) on tip-vortex aerodynamics should effectively be reflected on BVI noise generation; hence further investigation of the acoustic aspect of this concept would be required.

It has been proven that the ABT concept is capable of improving rotor performance, whilst offering the potential of easier implementation compared to HHC and IBT concepts (Ref. [119]). It has been also shown that ABT can effectively reduce BVI noise generation, although low-frequency noise penalties have been observed. Ultimately, the combination of VRS and ABT would further improve the performance and acoustic benefits associated with each concept.

2.5.3 Combined variable rotor speed and active blade twist

Han *et al.* (Ref. [119]) presented a combined variable rotor speed and blade twist approach to improve the performance of a UH-60A helicopter. It was demonstrated that at a cruise speed of 250 km/h, individual variation of rotor speed and blade twist leads to 17.8% and 10.4% power savings, respectively.

Combined variation led to 20.9% reduction in power requirement at the same cruise speed. It was also shown that the improvements in power decrease at high take-off weights.

Goulos and Bonesso (Ref. [112]) investigated the combined VRS/ABT concept at multiple levels of assessment, including aircraft-engine and mission levels. A Bo 105 helicopter model was employed within an elaborate optimisation approach comprising methods applicable to flight dynamics, aeroelasticity, engine performance, gaseous emissions and flight path analysis. The integrated framework was deployed to derive globally optimum rotor speed and blade twist schedules to minimise engine fuel flow. The devised rotor control schedules were utilised in the simulation of realistic, three-dimensional mission scenarios. It was shown that the benefits associated with combined VRS/ABT morphing are mission dependent. Performance improvements of up to 5% and 8% in terms of fuel consumption and NO_x emissions, respectively, were observed.

It has been discussed that, the combined VRS/ABT morphing can lead to increased performance benefits compared to individual application of the two aforementioned concepts. Reduction of rotor speed leads to lower dynamic head and eventually reduced blade profile drag. However, a concurrent increase in local blade pitch angle is required to restore thrust at the required levels. Additionally, in low speed cruise, induced power, which typically dominates at these conditions, is not significantly affected by rotor speed variation. Helicopter performance is fundamentally correlated with the associated environmental impact in terms of pollutant emissions. Blade pitch variation constitutes an efficient means of mitigating adverse effects associated with VRS. Additionally, it has been shown that blade twist variation affects the diffusion of blade tip-vortices and eventually, BVI noise. On the other hand, low-harmonic noise penalties have been associated with twist variation, which dictate that further investigation of the VBT concept, ultimately combined with VRS, would potentially offer improved global benefits. The combined VRS/ABT concept has not yet been investigated from an aeroacoustic perspective, thus research on this concept would shed light on the suitability of this promising

morphing rotor technology towards the concurrent minimisation of environmental and acoustic impact of modern helicopters.

2.6 Discussion

Helicopter noise sources have been discussed with emphasis on the major noise source, the main rotor. Helicopter rotor noise has been classified based on the contextual aerodynamic sources in the vicinity of the rotating blades. Capture of the assorted individual noise components requires coupling of high-resolution aerodynamic solutions with aeroacoustic formulations. With respect to the objectives of this work, which include mission-level noise prediction, uncertainty quantification in aeroacoustics, as well as optimisation of novel rotor concepts, it becomes clear that computational efficiency constitutes an additional requirement, along with predictive accuracy and robustness.

It has been discussed that, for the purposes of this work, analytical aerodynamic formulations based on free-wake models constitute the most suitable tool for the prediction of unsteady airloads including BVI phenomena. Free-wake aerodynamic models can be effectively coupled with analytical elastic blade models and blade-element-type unsteady aerodynamic models to comprehensively treat the aeroelastic response of helicopter rotor blades. It has been shown that integral aeroacoustic solutions of the FW-H equation represent an accurate and modular approach for rotor noise prediction in the time domain.

It has been demonstrated that, quantification of uncertainty in aeroacoustic modelling has the potential to enable non-deterministic noise predictions. Various sources of uncertainty have been studied in the open literature related to structural blade properties, rotor geometric parameters, aerodynamic properties, and operational or ambient conditions. Despite that free-wake aerodynamic models are routinely used in rotorcraft analysis for the purposes of aeroacoustic simulation, their impact on the robustness of acoustic predictions has not yet been thoroughly quantified. Free-wake models typically incorporate semi-empirical tip-vortex core diffusion models. Vortex core modelling is therefore associated with inherent uncertainty which essentially propagates into the aeroacoustic predictions. The impact of wake modelling uncertainty on

aeroacoustic predictions has not yet been quantified. Quantification of the impact of this source of uncertainty will contribute towards: (i) assessing the accuracy and robustness of acoustic prediction when experimental data are available for the tuning of tip-vortex core growth models and (ii) providing fundamental modelling guidelines for the coupled aerodynamic-aeroacoustic simulation in the absence of experimental data for correlations.

It has been shown that application of state-of-the-art modelling methods towards the assessment of rotorcraft acoustic impact should be treated at mission-level. Helicopters encompass strong variations in flight speed, flight path angle, altitude and gross weight, which are essentially reflected on the acoustic ground footprint of three-dimensional operations. Present ground footprint prediction methods mostly rely on pre-stored noise datasets corresponding to a finite number of pre-defined flight states, rather than generating the aerodynamic and acoustic content associated with each specific flight state of the mission. Additionally, the proposed approaches have only been demonstrated for short-duration and -range trajectories, rather than complete mission scenarios. The applicability of generalised and physics-based methods on the prediction of acoustic footprint of complete helicopter missions and without any pre-stored noise data has not yet been investigated.

The establishment of generalised methods for helicopter rotor noise prediction constitutes a crucial step towards source noise reduction. The diverse nature of helicopter flight states across “real-world” missions dictates that morphing rotor noise reduction technologies have the potential to collectively mitigate the acoustic impact and concurrently improve performance. The combined VRS/ABT concept can offer considerable performance and environmental improvements. It has also been discussed that ABT can effectively reduce BVI noise, although with low-harmonic noise penalties. The aeroacoustic analysis of combined VRS/VBT concept has not yet been conducted. Ultimately, the multi-disciplinary optimisation of this concept would shed light on the suitability of this promising morphing rotor technology towards the concurrent mitigation of the adverse environmental and acoustic effects of rotorcraft aviation.

3 Methodology

3.1 Introduction

This chapter elaborates on the methodology developed within this work. Helicopter noise prediction requires the capture of unsteady aerodynamic phenomena which are the fundamental noise generating mechanisms in the vicinity of the main rotor. Helicopter rotor induced flowfield is strongly affected by tip-vortices which generate high fluctuations to the local blade element flow incidence angles and consequently, rotor airloads. The latter gives rise to high-intensity BVI noise generation. To capture this effect, an existing free-wake aerodynamics model is utilised. The model is capable of predicting unsteady aerodynamics (including BVIs) and blade motion required for aeroacoustics assessments.

A mathematical formulation is deployed for modelling helicopter rotor noise. The method is provided in a form that can be directly coupled with the aerodynamic representation adopted in the free-wake rotor model. Compact integral expressions are utilised towards the computationally efficient prediction of rotor thickness and loading noise, including BVI components. A semi-empirical formulation for the prediction of broadband noise is also employed.

A statistical method for the quantification of the impact of tip-vortex core modelling uncertainty on the robustness of acoustic predictions is developed. The method is based on meta-modelling of the system's uncertain response to enable the efficient treatment of uncertainty in computationally demanding systems, where the application of traditional random methods would be prohibitive. A sensitivity analysis method is also described which allows the identification of the contribution of each individual uncertainty to the overall non-deterministic response.

An integrated method is presented for the prediction of helicopter rotor noise at mission level. The method couples the aforementioned rotor airloads and aeroacoustic models, with an existing flight dynamics model for the

comprehensive prediction of source noise on-the-fly. A dedicated numerical procedure is developed for the evaluation of the acoustic footprint of complete three-dimensional rotorcraft operations.

Finally, a multi-disciplinary optimisation strategy based on surrogate modelling is presented targeting the concurrent mitigation of adverse noise and NO_x emission effects for any set of flight conditions. The method is deployed for the generation of generalised optimal variable rotor speed and active blade twist control schedules covering the descent-region of the helicopter flight envelope, where BVI noise effects are more pronounced. The same method is applicable towards optimising and evaluating active rotor control technologies of a twin-engine light helicopter operating in a realistic descent trajectory, representative of EASA-standard noise certification procedures.

3.2 Mathematical Modelling of Helicopter Rotor Source Noise

3.2.1 Rotor aerodynamic modelling

A free-wake and unsteady aeroelastic rotor model is utilised to predict elastic rotor blade unsteady motion and unsteady airloads, including BVIs. The model has originally been developed by Castillo Pardo (Ref. [111]). The integrated model comprises a series of fundamental methods related to aeroelasticity and aerodynamics. A minimum potential energy method, similar to the one reported in Ref. [65], is deployed for the estimation of coupled flap-lag-torsion vibration characteristics of the elastic rotor blades (Ref. [67]). The matrix/vector based formulation presented in Ref. [66] is employed for modelling the flexible rotor blade kinematics and the Leishman-Beddoes indicial response method (Ref. [25]) for the prediction of unsteady blade airloads. The aforementioned models have been extensively described in the corresponding references; thus further discussion will be omitted.

A relaxation-type free-wake aerodynamic model is employed for the modelling of unsteady rotor inflow, able to capture the complex blade-vortex interaction phenomena (Ref. [59]). Each vortex is discretised as a series of straight vortex filaments with Lagrangian markers located at the ends of each filament, as

shown in Fig. 3.1. The governing equation for the convection of the Lagrangian markers is the unsteady Vorticity Transport Equation for incompressible and isotropic fluids (Ref. [17]):

$$\frac{D\vec{\omega}}{Dt} = (\vec{\omega} \cdot \nabla) \vec{V} + \nu \nabla^2 \vec{\omega}, \quad (3.1)$$

where $\vec{\omega}$ is the vorticity of the fluid element, $\vec{V}$ is the velocity vector at the same point and ν is the fluid kinematic viscosity.

It is assumed that viscous effects are significant only within the vortex core region of the filaments. Subsequently, the flow can be considered as inviscid outside the vortex singularity (Ref. [37]). This allows the application of Helmholtz's second law, which states the vortex lines move as material lines (Ref. [17]). Under this condition and following Leishman's textbook (Ref. [17]), Eq. (3.1) can be rewritten as:

$$\frac{\partial \vec{r}(\psi, \zeta)}{\partial \psi} + \frac{\partial \vec{r}(\psi, \zeta)}{\partial \zeta} = \frac{1}{\Omega} \vec{V}[\vec{r}(\psi, \zeta)], \quad (3.2)$$

where ψ is the azimuthal position of the blade, ζ is the age of trailed tip-vortices, $\vec{r}(\psi, \zeta)$ is the position vector of the Lagrangian marker and Ω is the

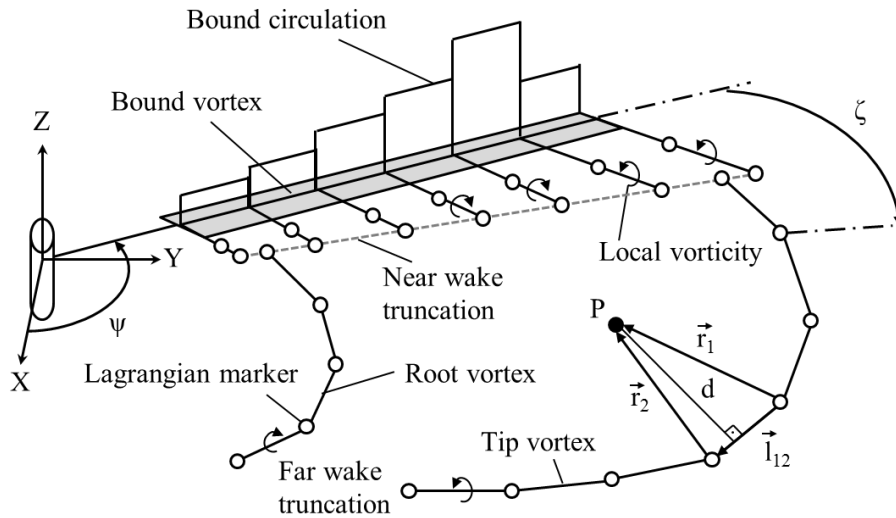


Figure 3.1: Free-wake rotor airloads model overview.

rotational speed of the rotor. Eq. (3.2) is a partial differential equation (PDE) expressed in the blade-fixed coordinate system. The left-hand side is a one dimensional wave equation which is solved by using the pseudo-implicit predictor-corrector, relaxation free-vortex method proposed by Bagai and Leishman (Ref. [59]). The right-hand side is a highly nonlinear velocity term which includes the flight velocity components and the velocity induced by all vortex filaments. Induced velocity is evaluated through the Biot-Savart law for straight vortex filaments including the Vatisas vortex core model (Ref. [120]):

$$\vec{V}_{ind} = \frac{\Gamma_v}{4\pi} \left(\frac{d}{(r_c^{2n} + d^{2n})^{1/n}} \right) \frac{\vec{l}_{12}}{|\vec{l}_{12}|} \cdot \left(\frac{\vec{r}_1}{|\vec{r}_1|} - \frac{\vec{r}_2}{|\vec{r}_2|} \right) \cdot \left(\frac{\vec{l}_{12} \times \vec{r}}{|\vec{l}_{12} \times \vec{r}_1|} \right), \quad (3.3)$$

where Γ_v is the circulation strength of the vortex, $\vec{l}_{12}$ is the vortex filament vector, $\vec{r}_1$ and $\vec{r}_2$ are the position vectors of the arbitrary point P relative to the Lagrangian markers, $\vec{r}$ is the position vector of point P relative to the vortex segment. The perpendicular distance between P and filament is $d = |\vec{l}_{12} \times \vec{r}| / |\vec{l}_{12}|$. The tangential velocity profile of the Vatisas model is defined by the general constant n which best matches the experimental velocities measured in Ref. [120]. Finally, the vortex core radius r_c is given by the Squire's equation:

$$r_c(\zeta) = \sqrt{r_{c0}^2 + 4\alpha\delta\nu \frac{\zeta}{\Omega}}, \quad (3.4)$$

where r_{c0} is the initial core radius and $\alpha = 1.25643$ is the Oseen constant. The eddy viscosity parameter δ is calculated by:

$$\delta = 1 + \alpha_1 \frac{\Gamma_v}{\nu}. \quad (3.5)$$

Bhagwat *et al.* (Ref. [30]) suggested values in the order of 10^{-4} to 10^{-3} for the Squire parameter α_1 for small scale rotors. Although the initial vortex core radius of 5-10% chord is typically employed in forward flight, a larger initial core radius $r_{c0} = 0.6c$ has been used in some studies to reduce spurious BVI

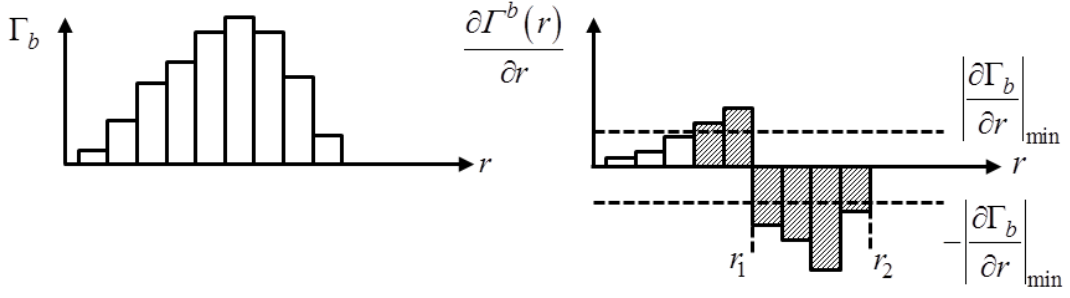


Figure 3.2: Vortex roll-up model overview.

response in descent (Ref. [32]). Different core radii may be used for the evaluation of induced velocity for wake calculations from those used to compute rotor inflow, to improve convergence stability and speed (Ref. [32]). Typically, the values of vortex core model parameters that achieve the best match between predicted and measured noise data are selected.

The wake is split to near wake and far wake regions, as depicted in Fig. 3.1. The near wake is typically truncated at a specific age. The far wake comprises root and tip-vortices, resolving a finite number of wake rotations for the calculation of induced flowfield. The near and far wake regions are linked via Betz's roll-up model (Ref. [121]) based on the distribution of bound circulation Γ_b and local vorticity $\partial\Gamma_b(r)/\partial r$. The total circulation of the rolled-up vortex

$\Gamma_v = \int_{r_1}^{r_2} (\partial\Gamma_b(r)/\partial r) dr$ is calculated based on radial integration of elementary

blade vorticities above an empirically-defined threshold $\left| \frac{\partial\Gamma_b}{\partial r} \right|_{\min}$ as described in

Ref. [122] and illustrated in Fig. 3.2. The radial location

$r_v = \int_{r_1}^{r_2} (\partial\Gamma_b(r)/\partial r) r dr / \Gamma_v$ at the vortex emission time is estimates as a

circulation-weighted average of all blade elements. Therefore, contributions from all the near-wake vortex filament are accounted in the calculation rolled-up vortex strength and trajectory. Shed vortices are not resolved in the wake model as they are fundamentally accounted for in the unsteady blade aerodynamic model through the deficiency functions applied on the circulatory normal force coefficient (Ref. [25]).

The presence of fuselage below the rotor gives rise to aerodynamic effects which affect airloads. The semi-empirical formulation for the prediction of velocity induced by the fuselage, proposed by van der Wall *et al.* (Refs. [29], [123]) is employed. The method provides analytical shape functions of induced velocity, derived from correlations with experimental or CFD data (Ref. [123]).

The aforementioned individual aerodynamic and structural models are incorporated into a globally-convergent trim algorithm based on Broyden's numerical method. The trim algorithm solves for the rotor collective δ_0 and longitudinal δ_{1s} and lateral δ_{1c} cyclic controls for given flight conditions and rotor thrust C_T , rolling C_L and pitching moment C_M coefficients.

3.2.2 Near-field noise prediction method

The primary sources of helicopter noise are the main and tail rotors (Ref. [14]). Rotor noise can be divided into discrete-frequency and broadband. Discrete frequency noise includes thickness noise, loading noise and high-speed impulsive noise. A new code, ICARUS, developed by the author, is deployed to model helicopter rotor aeroacoustics (Ref. [124]). This code is an integral implementation of the Ffowcs Williams-Hawkings equation (Ref. [43]), which is the most general form of Lighthill's acoustic analogy (Ref. [42]). The FW-H equation is an exact rearrangement of the continuity equation and the Navier-Stokes equations into the form of an inhomogeneous wave equation with monopole and dipole terms across the acoustic data surface and a quadrupole source distribution surrounding it.

$$\begin{aligned} \left(\frac{1}{c_0^2} \frac{\partial^2}{\partial t^2} - \frac{\partial^2}{\partial x_i^2} \right) p'(\vec{x}, t) = & \frac{\partial}{\partial t} \left\{ \left[\rho_0 \vec{v} \cdot \hat{n} + \rho (\vec{u} \cdot \hat{n} - \vec{v} \cdot \hat{n}) \right] \delta(f) \right\} \\ & - \frac{\partial}{\partial x_i} \left\{ \left[\Delta P_{ij} \hat{n}_j + \rho u_i (\vec{u} \cdot \hat{n} - \vec{v} \cdot \hat{n}) \right] \delta(f) \right\}, \quad (3.6) \\ & + \frac{\partial^2}{\partial x_i \partial x_j} \left[T_{ij} H(f) \right] \end{aligned}$$

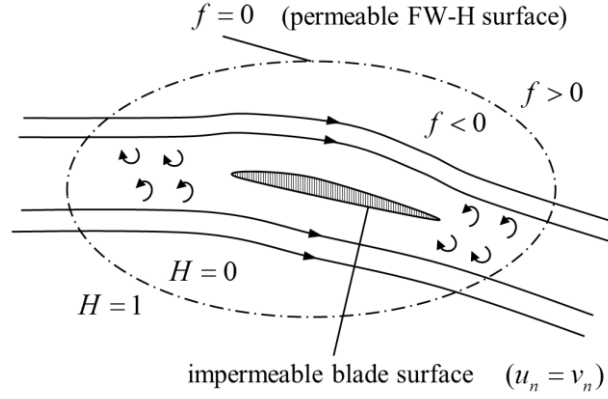


Figure 3.3: Concepts of permeable and impermeable acoustic data surface.

where $p'(\vec{x}, t)$ is the acoustic disturbance evaluated at the observer position $\vec{x}$ and observer time t and c_0 and ρ_0 are the speed of sound and density of the quiescent acoustic medium, respectively. Moreover, $\vec{v}$ is the source surface velocity vector, $\vec{u}$ is the fluid velocity vector and $\hat{n}$ is the unit outward normal vector to the source surface. The Lighthill stress tensor is defined as:

$$T_{ij} = \rho u_i u_j + P_{ij} - c_0^2 (\rho - \rho_0) \delta_{ij}, \quad (3.7)$$

where ρ is the local fluid density and δ_{ij} the Kronecker delta. The compressive stress tensor is defined as $\Delta P_{ij} \hat{n}_j = (p - p_0) \delta_{ij}$ after neglecting the viscous forces. The source surface is defined by the function $f = 0$ and $H(f)$ is the Heaviside function. The left-hand side term of Eq. (3.6) represents the three-dimensional wave operator. On the right-hand side of the equation, there are three noise source terms, which are of monopole, dipole and quadrupole nature, respectively.

Formulation 1A of Farassat (Refs. [47], [48]) is the most extensively utilised and validated integral solution of the FW-H equation. It is based on the free-space Green's function solution of the three-dimensional wave equation. Farassat recognised that the wave operator is a linear operator, so both the temporal and spatial derivatives commute (Ref. [14]). Application of the Formulation 1A considering impermeable ADS implies that the fluid velocity is equal to the

blade velocity $\vec{u} = \vec{v}$ on an arbitrary location on the impermeable blade surface, as depicted in Fig. 3.3. Formulation 1A resolves thickness (Eq. 3.8) and loading (Eq. 3.9) noise components, which correspond to the monopole and dipole acoustic terms, respectively. The high-speed impulsive noise component which corresponds to the quadrupole term of the FW-H equation is neglected. This is valid as long as the rotor tip Mach number is below the transonic limit, which is the typical condition for the majority of civil rotorcraft operations. The derived monopole p'_m and dipole p'_d components (Ref. [41]) are presented below:

$$\begin{aligned} \frac{4\pi}{\rho_0} p'_m(\vec{x}, t) = & \int_{f=0} \frac{\frac{\partial \vec{v}}{\partial \tau} \cdot \hat{n} + \vec{v} \cdot \frac{\partial \hat{n}}{\partial \tau}}{r |1 - \vec{M} \cdot \hat{r}|^2} \bigg|_{ret} dS \\ & + \int_{f=0} \frac{\vec{v} \cdot \hat{n} \left(r \frac{\partial \vec{M}}{\partial \tau} \cdot \hat{r} + c_0 (\vec{M} \cdot \hat{r} - |\vec{M}|^2) \right)}{r^2 |1 - \vec{M} \cdot \hat{r}|^3} \bigg|_{ret} dS \end{aligned} \quad (3.8)$$

$$\begin{aligned} 4\pi c_0 p'_d(\vec{x}, t) = & \int_{f=0} \frac{\frac{\partial \vec{l}}{\partial \tau} \cdot \hat{r}}{r |1 - \vec{M} \cdot \hat{r}|^2} \bigg|_{ret} dS \\ & + \int_{f=0} \frac{\vec{l} \cdot \hat{r} - \vec{l} \cdot \vec{M}}{r^2 |1 - \vec{M} \cdot \hat{r}|^2} \bigg|_{ret} dS \\ & + \int_{f=0} \frac{\vec{l} \cdot \hat{r} \left(r \frac{\partial \vec{M}}{\partial \tau} \cdot \hat{r} + c_0 (\vec{M} \cdot \hat{r} - |\vec{M}|^2) \right)}{r^2 |1 - \vec{M} \cdot \hat{r}|^3} \bigg|_{ret} dS \end{aligned} \quad (3.9)$$

The vectors $\vec{v}$, $\vec{M}$ and $\vec{l}$ correspond to the local velocity vector, the local relative Mach number vector and the local force intensity vector, respectively. The vectors $\hat{n}$ and $\hat{r}$ correspond to the unit outward normal vector and the unit radiation vector, respectively. It is noted that the time derivatives are being evaluated in the source time. The subscript "ret" represents numerical integration within the retarded-time (i.e. source-time). The aforementioned non-

compact expressions of Formulation 1A require both spanwise and chordwise discretisation of the blade surface. Hence, the blade surface defined by the ADS function $f = 0$ is required to be divided into panels of elementary area dS .

Chordwise compact acoustic expressions of the Formulation 1A can be utilised when the blade chord can be small compared to the wavelength of the radiated sound [37]. Brentner *et al.* (Ref. [58]) introduced a compact form of the loading noise term, utilising the blade lift distribution $\vec{L}$ across the span. Consequently, the computation reduced from a surface integral to a line integral. For representative values of rotor blade passage frequency (BPF), compact chordwise approximation is valid up to the 35th harmonic of the radiated noise spectrum (Ref. [125]), which is deemed as adequate to capture all the major harmonics of BVI noise. Lopes (Ref. [126]) presented and validated a compact expression of the monopole term based on the geometrically compact assumption that the radiation distance is large compared to blade chord. The proposed formulation utilises the spanwise distribution of the aerofoil cross-sectional area A_{cs} with differential span length dy . Hence, thickness acoustic pressure is also evaluated as a line integral defined by the line equation $f = 0$. Employing the shorthand notation introduced by Lee *et al.* in Ref. [127] and followed by Lopes in Ref. [126], the compact acoustic expressions are:

$$\begin{aligned} \frac{4\pi}{\rho_0} p'_m(\vec{x}, t) = & \int_{f=0} A_{cs} \left(W + \frac{\partial W}{\partial \tau} \right) R(2, 4) \Big|_{ret} dy \\ & + \int_{f=0} 2A_{cs} W^2 R(3, 5) \Big|_{ret} dy, \\ & - \int_{f=0} c_0 A_{cs} \frac{\partial \vec{M}}{\partial \tau} \cdot \hat{r} R(2, 3) \Big|_{ret} dy \end{aligned} \quad (3.10)$$

$$\begin{aligned} 4\pi c_0 p'_d(\vec{x}, t) = & \int_{f=0} \frac{\partial \vec{L}}{\partial \tau} \cdot \hat{r} R(1, 2) \Big|_{ret} dy \\ & + \int_{f=0} c_0 (\vec{L} \cdot \hat{r} - \vec{L} \cdot \vec{M}) R(2, 2) \Big|_{ret} dy, \\ & + \int_{f=0} \vec{L} \cdot \hat{r} W R(2, 3) \Big|_{ret} dy \end{aligned} \quad (3.11)$$

where $\vec{L}$ is the blade element loading vector. The following shorthand functions are being used, as suggested by Lopes (Ref. [126]):

$$R(n, m) = r^{-n} (1 - \vec{M} \cdot \hat{r})^{-m}, \quad (3.12)$$

$$W = r \frac{\partial \vec{M}}{\partial \tau} \cdot \hat{r} + c_0 \left(\vec{M} \cdot \hat{r} - |\vec{M}|^2 \right), \quad (3.13)$$

The source-time-derivative of the function W has been analytically determined by Lee (Ref. [127]):

$$\frac{\partial W}{\partial \tau} = r \frac{\partial^2 \vec{M}}{\partial \tau^2} \cdot \hat{r} - 3c_0 \frac{\partial \vec{M}}{\partial \tau} \cdot \vec{M} + c_0 \left(\frac{\partial \vec{M}}{\partial \tau} \cdot \hat{r} + \frac{c_0}{r} \left(\left(\frac{\partial \vec{M}}{\partial \tau} \cdot \hat{r} \right)^2 - |\vec{M}|^2 \right) \right), \quad (3.14)$$

where r is the radiation distance and $|\vec{M}|$ is the magnitude of the local relative Mach number vector.

The overall discrete frequency acoustic pressure $p'(\vec{x}, t)$, evaluated at the observer position $\vec{x}$, located at distance r from the source point and at the observer time t can be expressed as the summary of monopole $p'_m(\vec{x}, t)$ and dipole $p'_d(\vec{x}, t)$ acoustic disturbance contributions:

$$p'(\vec{x}, t) = p'_m(\vec{x}, t) + p'_d(\vec{x}, t), \quad (3.15)$$

The compact acoustic formulations adopted in this study are compatible with the lifting-line-type aerodynamic input provided by the free-wake airloads model. Moreover, computational savings of up to 99.5% have been reported with only slight deviations from the corresponding non-compact deterministic acoustic pressure computations (Refs. [58], [126]). This enables the incorporation of on-the-fly and physics-based rotor source noise prediction into the simulation of complete rotorcraft operations.

Broadband noise is a stochastic noise source; hence turbulence statistics would be required for its first-principles prediction. Furthermore, extensive

computational resources would be required which goes beyond the purpose of mission-level predictions. The semi-empirical model developed by Pegg (Ref. [72]) is employed for the prediction of broadband noise in the frequency domain. Pegg's model accounts for contributions from boundary layer noise, inflow turbulence noise and vortex noise. It has been shown that the inclusion of Pegg's model accounts for the deficiency in ground noise predictions observed when broadband noise is neglected (Ref. [73]).

3.3 Uncertainty Quantification in Aeroacoustic Simulation

As discussed in Section 3.2.1, a free-wake aerodynamics model is employed to capture complex BVI phenomena. Free wake models are routinely used in aeroacoustic assessment of helicopter rotors; however, their semi-empiricism is essentially accompanied with uncertainty related to wake tip-vortex core growth physical parameters. In some cases, analysts have to resort to empirical adaption of these parameters based on previous experimental evidence. Vortex core radius is a critical parameter which significantly affects airloads and consequently, noise. The quantification of the impact of this source of uncertainty on the robustness of aeroacoustic predictions has not yet been quantified in the open literature.

Traditional approaches for uncertainty analysis involve Monte Carlo (MC) simulation which is based on random sampling of the uncertain input parameters. Subsequently, deterministic evaluations of the investigated model are performed for each sample point and the statistics of output quantities of interest are explored. Monte Carlo methods are widely utilised due to the ease of implementation, but the number of samples required for convergence of the statistical properties of the outputs is typically in the order of thousands of evaluations (Refs. [76], [78]). This number becomes prohibitive when the deterministic model evaluations are computationally demanding. The coupled approach adopted for the prediction of BVI noise within this work comprises a series of fundamental modelling methods associated with aeroelasticity, flight dynamics, unsteady aerodynamics including treatment of wake tip-vortices and aeroacoustics. With respect to the aforementioned employed modelling

methods, it becomes clear that an alternative approach, requiring fewer deterministic evaluations is needed to investigate the impact of wake modelling uncertainty on noise predictions.

Point collocation methods constitute an efficient alternative to the computationally expensive MC-based methods for uncertainty quantification. As discussed in Section 2.2, Polynomial Chaos expansion (PCE) has been utilised for the quantification of uncertainty in aeroacoustics when the deterministic model is computationally demanding. A PCE approach (Ref. [83]) is employed for the quantification of uncertainty in aeroacoustic analysis due to the uncertainty in wake tip-vortex core physical parameters utilised in this work. PCE describes the response $Y(\xi)$ of a system encompassing non-deterministic input ξ of known distribution as an infinite series of orthogonal polynomial basis functions $\varphi_i(\xi)$, provided that the response behaviour is smooth. Specifically:

$$Y(\xi) = \sum_{i=0}^{\infty} \varphi_i(\xi) a_i, \quad (3.16)$$

where a_i are the coefficients of the orthogonal polynomial basis functions. For practical engineering applications, the infinite series is truncated to a number of $p+1$ basis functions:

$$Y(\xi) \approx \sum_{i=0}^p \varphi_i(\xi) a_i, \quad (3.17)$$

The orthogonality condition of polynomial basis functions is expressed in a normalised form as following:

$$\langle \varphi_i(\xi), \varphi_j(\xi) \rangle = \int_A f(\xi) \varphi_i(\xi) \varphi_j(\xi) d\xi = \delta_{ij}, \quad (3.18)$$

where $\langle \cdot, \cdot \rangle$ denotes the inner product, $f(\xi)$ is the joint probability density function of the distribution with support A and δ_{ij} is the Kronecker delta. It is noted that the orthogonal polynomial basis functions $\varphi_i(\xi)$ are solely specified

by the known random distribution, hence no input from the deterministic model is yet required. As an example, the first four Hermite polynomials (0th to 3rd order), which are the orthogonal polynomials corresponding to a Gaussian distribution with expected value 0 and standard deviation 1, are expressed below:

$$H_0(\xi) = 1, \quad (3.19)$$

$$H_1(\xi) = \xi, \quad (3.20)$$

$$H_2(\xi) = \xi^2 - 1, \quad (3.21)$$

$$H_3(\xi) = \xi^3 - 3\xi. \quad (3.22)$$

Hence, the estimation of the non-deterministic response of the system, according to Eq. (3.17) requires only the determination of the coefficients a_i of the polynomial terms. These are determined by solving the linear system of equations formed by the expansion of Eq. (3.17) as follows:

$$\begin{bmatrix} y(\xi_0) \\ \vdots \\ y(\xi_m) \end{bmatrix} = \begin{bmatrix} \varphi_p(\xi_0) & \cdots & \varphi_0(\xi_0) \\ \vdots & \ddots & \vdots \\ \varphi_p(\xi_m) & \cdots & \varphi_0(\xi_m) \end{bmatrix} \cdot \begin{bmatrix} a_p \\ \vdots \\ a_0 \end{bmatrix}, \quad (3.23)$$

where ξ_i are the $m+1$ collocation points at which the system's deterministic response is evaluated, $y(\xi_i)$ is the response of the system at each collocation point and a_i the unknown coefficients of the polynomial of order p . The minimum number of collocation points to solve Eq. 3.23, which corresponds to the number of deterministic evaluations of the system with n uncertain input parameters, is the cardinality of the dataset $\mathcal{E}$ and is given by:

$$\text{card}(\mathcal{E}) = 1 + \sum_{i=1}^p \frac{1}{i!} \prod_{j=0}^{i-1} (n+j) = \frac{(n+p)!}{n!p!}. \quad (3.24)$$

The selection of collocation points for the deterministic evaluation of the system has a strong impact on the accuracy of the Polynomial Chaos approximation. The low-discrepancy Halton sequence is employed for the determination of collocation points, which offers better coverage of the design space compared to random sampling methods due to the inline minimisation of mathematical discrepancy within the sequence. Ref. [128] presented the improved coverage properties of low-discrepancy sampling relative to quasi-random Latin Hypercube Sampling and random sampling. This enables the construction of polynomial response approximation with an affordable number of collocation points. Typically, over-sampling is required to improve the accuracy of the system's response approximation with PCE (Ref. [128]). The effective polynomial expansion is finally derived through least-squares regression (Ref. [129]). PCE treats the deterministic model as a black-box; hence it is a non-intrusive method for uncertainty quantification, thereby rendering it a suitable approach for engineering applications.

Orthogonal polynomials are based on the probability density function of the random parameters; hence they feature convenient properties for statistical analysis. According to Ref. [130], the mean and variance of the system's output can be analytically derived from:

$$E(Y(\xi)) = a_0, \quad (3.25)$$

$$V(Y(\xi)) = \sum_{i=1}^p a_i^2 \langle \varphi_i(\xi), \varphi_i(\xi) \rangle. \quad (3.26)$$

In this work, the developed PCE-based approximation will be used as meta-model to be provided to an MC simulation method which, apart from statistical moments, can provide an extensive, but now computationally affordable dataset for detailed statistical analysis. MC is also used for construction of probability histograms for comparisons between non-deterministic simulation and randomly-picked inputs in the absence of reliable experimental evidence. Based on Ref. [131], a distribution cannot be uniquely defined by its statistical moments. Hence, the use of MC simulation is required to obtain the probability

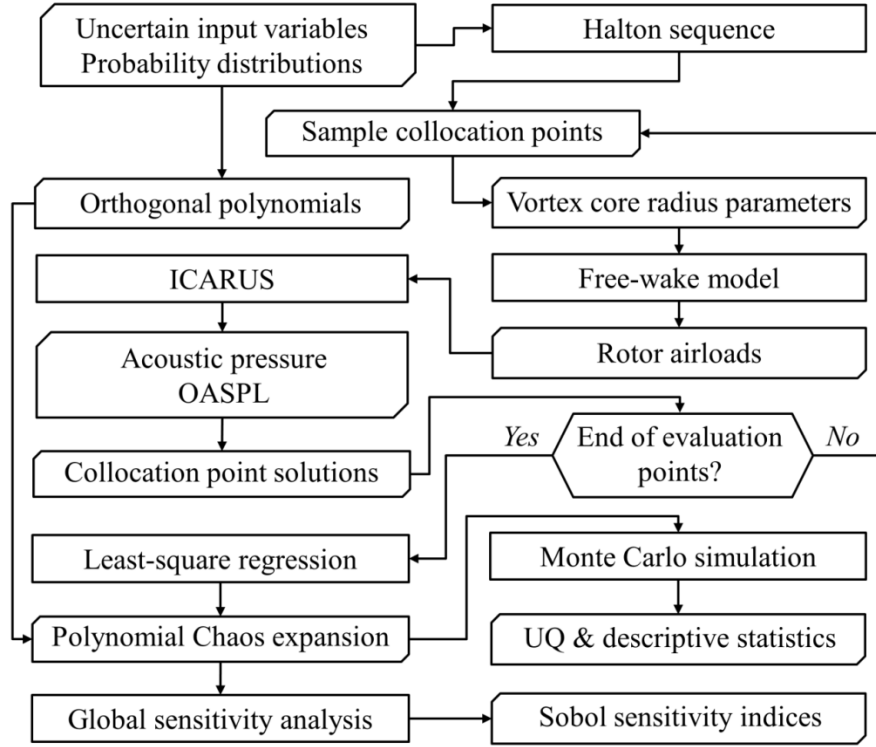


Figure 3.4: Uncertainty quantification method overview.

density functions of the output distributions. The coupled PCE-MC approach has also been followed in Refs. [132], [86], [133], [134].

PCE enables the capability of directly performing variance-based sensitivity analysis to identify the relative sensitivity of the non-deterministic output to the uncertain input parameters. Sensitivity analysis based on Sobol indices (Refs. [89], [135]) is allowed when the response of the system can be decomposed in summands that are mutually orthogonal. Thus, first- or higher-order sensitivity indices S_i can be derived in the following format:

$$S_i = \frac{V(E(Y|\xi_i))}{V(Y(\xi))}. \quad (3.27)$$

A statistical method based on the aforementioned approach has been developed in Python. An overview of the method is presented in Fig. 3.4. Ultimately, the proposed method can be used for fundamental investigations on the impact of input uncertainty on aeroacoustic predictions.

3.4 Helicopter Rotor Noise Prediction at Mission Level

3.4.1 Mission flight dynamics model

The comprehensive framework “HECTOR” (Helicopter Omni-disciplinary Research Platform) is employed for the calculation of flight kinematics and engine performance across the mission. HECTOR, developed by Goulos, has been extensively described in Refs. [28], [65], [66], [136], therefore only a brief description of the individual modelling methods is provided in this section. The aeroelastic model for the simulation of rotor blade flexibility in the time-domain, is based on the numerical method described in Ref. [66]. A minimum potential energy approach, based on Lagrangian kinematics, is employed to obtain the natural vibration characteristics of the main rotor blades for flap-lag-torsion (Ref. [65]). The structural formulation of the rotor model is coupled with the Leishman–Beddoes unsteady nonlinear blade element aerodynamics model (Ref. [25]), the Peters-He finite state induced flow model (Ref. [137]) and the Goulos dynamic wake curvature model (Ref. [136]). A globally convergent Newton-Raphson approach is incorporated to obtain the required trim control and fuselage attitude angles for any set of flight conditions utilising the individual models mentioned above.

Fig. 3.5 presents the mission analysis numerical procedure incorporated in HECTOR which has been introduced in Ref. [28]. The mission path is described in a global geographical system (longitude-latitude) based on the expressions employed by WGS 84 Ref. [138]. The overall process involves the breakdown of each designated mission task element into discrete segments with a predefined time step. The helicopter is assumed to be operating in trim and the engine in steady-state off-design during each segment. The initial rotorcraft all-up mass (AUM) is determined based on a user-defined initial guess of the total mission fuel consumption. The helicopter is trimmed depending on the flight conditions defined within the mission. The numerical scheme is applied towards the solution of the Euler’s equations of motion for the aircraft fuselage which is treated dynamically as a rigid body with six degrees of freedom, three translational and three rotational.

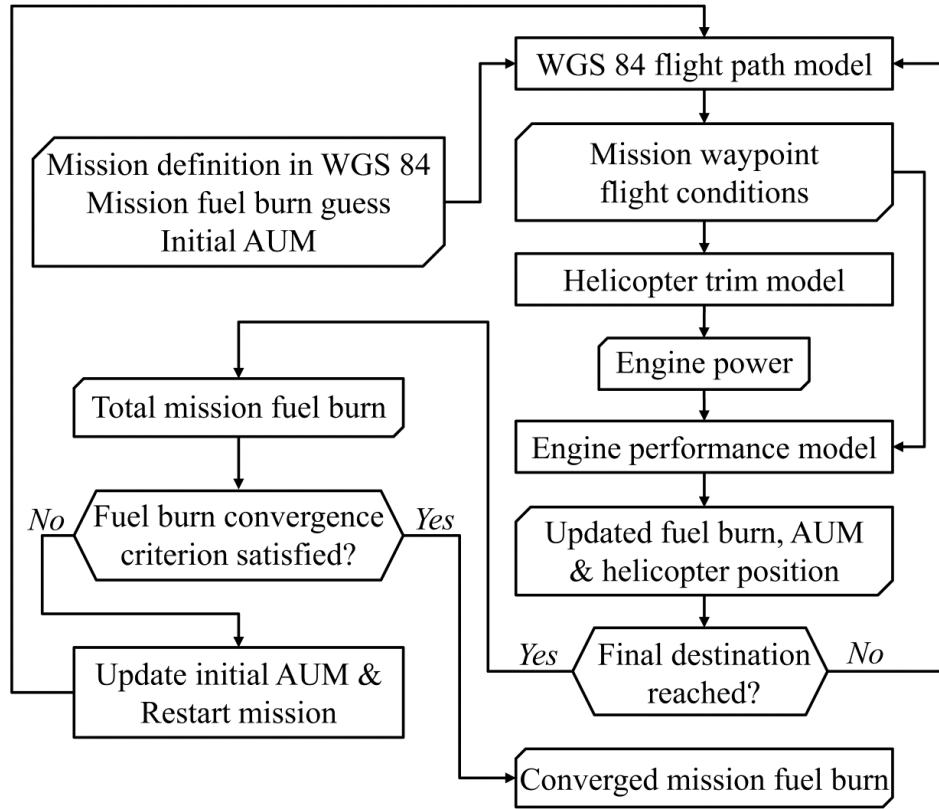


Figure 3.5: Rotorcraft flight dynamics model overview.

The engine performance model (Turbomatch) employed in HECTOR has been developed and improved for over three decades at Cranfield University (Ref. [139]). Turbomatch is based on zero-dimensional analysis of the aerothermodynamic processes occurring throughout the engine gas path, employing discrete component maps. Therefore, the engine fuel flow is determined among other off-design performance parameters. The time-dependent fuel consumption is calculated by applying a numerical time integration scheme over the corresponding mission segments. This value is subtracted from the AUM to simulate the helicopter's gradual weight reduction along the mission.

The helicopter's spatial location is updated based on the flight conditions dictated by the flight path model. Having completed a first-pass along the entire mission, the estimated value of total fuel consumption is compared with the initial fuel burn guess. If the obtained percentage difference in mission fuel burn

between successive flight path iterations is within a user-defined tolerance, the process is halted. Otherwise, the initial AUM is updated based on the calculated total mission fuel consumption. The overall process is re-iterated in a fixed-point manner until the deviation of the calculated fuel consumption from the AUM is below the user-specified convergence criterion.

The procedure described above determines the required rotor thrust (C_T), rolling moment (C_L) and pitching moment (C_M) coefficients, as well as fuselage attitudes in a computationally inexpensive manner, based on the resolved rotorcraft trim state of each trajectory waypoint. These requirements are then passed to the free-wake aeroelastic rotor model to trim the main rotor and provide the high-resolution airloads necessary for the source-noise predictions.

3.4.2 Far-field noise propagation model

A source-time dominant algorithm has been developed for the numerical implementation of the retarded-time acoustic formulation, as described in Ref. [45]. The source position and time are determined a-priori according to the blade kinematics and flight dynamics calculations. Next, for the requested observer positions, the corresponding observer times are explicitly calculated using the retarded-time equation:

$$t = \tau + |\vec{x} - \vec{y}| / c_0, \quad (3.25)$$

where t is the observer (i.e. retarded) time, τ is the source time and $\vec{x}$, $\vec{y}$ are the observer and the source position vectors, respectively, relative to and expressed in the ground fixed reference frame.

The drawback of this method is that the acoustic pressure time-histories for each observer are unequally spaced in time, thus time-interpolation is needed to resolve acoustic pressure at the desired time intervals. However, co-ordinate transformations between source-observer are evaluated only once per source point, thus significant computational savings occur compared to the traditional observer-time dominant (or retarded-time) numerical algorithm (Ref. [45]).

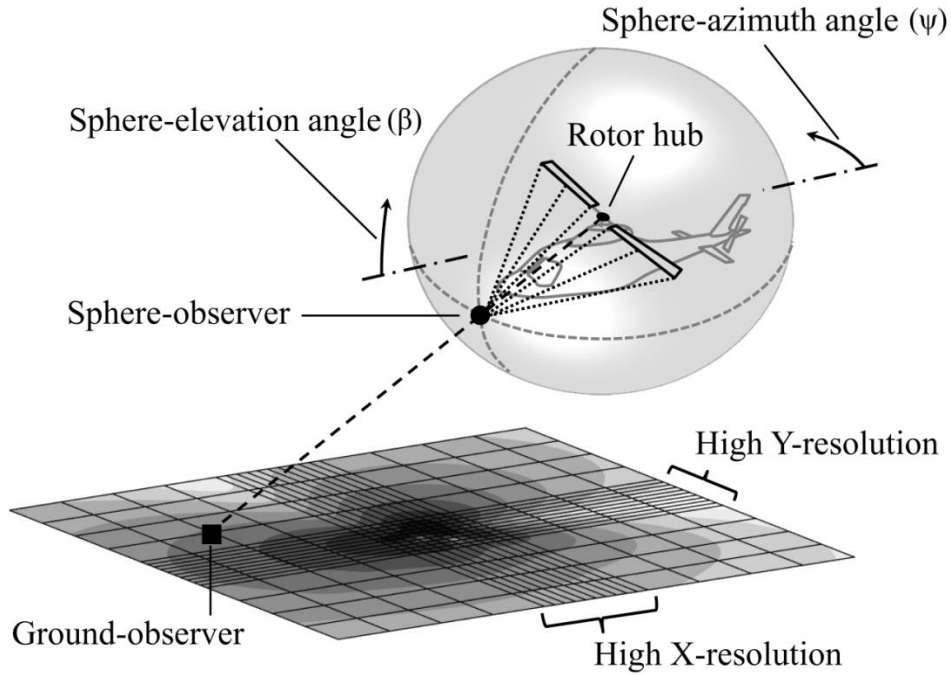


Figure 3.6: Rotor aeroacoustics model overview.

Fig. 3.6 presents the summary of the rotor noise model implementation for the prediction of the acoustic impact across the grid of ground observers. A two-step procedure is adopted for the prediction of source noise and its far-field propagation. A fictitious acoustic sphere is considered around the rotorcraft and fixed to the main rotor hub which will represent the source-noise to be propagated to the ground. For each ground observer, a ray-approach is adopted to determine the intersection point between the ray and the sphere. This intersection point serves as an intermediate sphere-observer for the evaluation of acoustic pressure time-history for one rotor revolution. The time-domain acoustic signal is then spectrally analysed via fast Fourier transformation (FFT) to provide a frequency-domain expression of the rotor acoustic impact on the designated sphere-observer. The broadband noise contribution is then added to the deterministic frequency-domain noise components.

Formulation 1A does not account for sound attenuation and propagation losses. However this is essential for mission-level applications considering the altitude

variations typically encountered by helicopters. The propagation losses during the sound wave travel from the acoustic sphere to the ground observer are accounted for in the frequency domain, assuming no losses within the sphere volume. Atmospheric absorption is assessed using the method proposed by Sutherland *et al.* (Ref. [140]) and standardised by ISO 9613-1 and ANSI S1.26-1995. The employment of the acoustic sphere aims to the consistent application of far-field propagation from a common geometric starting point for all the individual noise contributions coming from the rotor blades. The sphere has a radius of 150 m, as followed in Ref. [40] to properly apply the inverse square law for spherical geometric spreading.

The procedure described above is repeated for the entire set of ground observers. Following this methodology, the evaluation of source noise levels across the entire acoustic sphere surface is avoided, since only the ray-sphere intersection points are considered, as shown in Fig. 3.6. Consequently, the transition from pre-stored hemisphere databases to the generation of acoustic spheres on-the-fly, across the mission, is realised. Additionally, dedicated acoustic spheres are constructed reflecting the specific flight state of the helicopter. Consequently, the proposed methodology can be considered as generalised and independent of pre-stored databases as input. Moreover, there is no need for interpolation on the non-regular and usually sparse polar grid of conventional noise hemispheres.

For the purposes of this work, only the acoustic content of the main rotor has been included in the analysis. However, the same methodology can be adopted for the prediction of tail rotor noise, or multi-rotor aircraft. Other noise sources such as engine noise or drivetrain noise are of lower intensity and easily attenuated with distance due to their high frequencies. These sources will be neglected as in Refs. [93] and [35], although there might be an underprediction of ground noise levels at specific instances of the mission and for observers located very close to the helicopter.

3.4.3 Integrated numerical procedure

A three-step integrated procedure has been developed to reduce computational time and enable the generation of acoustic spheres on-the-fly based on high resolution computation of unsteady airloads. This involves the coupling of flight dynamics, elastic blade motion, unsteady airloads and aeroacoustics at mission level. First, the comprehensive code HECTOR is used to compute the flight path, aircraft motion, and engine performance and to update the AUM based on the iterative process described in section 3.4.1. The converged HECTOR mission output is used to set up the rotor trim requirements for the free-wake airloads model. The airloads model is trimmed for thrust, rolling moment and pitching moment coefficients at each element of the mission for the flight conditions specified by HECTOR. The final unsteady blade airloads (including BVIs), elastic blade motion, as well as blade design are provided by the free-wake airloads model to the aeroacoustics code, as shown in Fig. 3.7. The rotor airloads prediction procedure described in section 3.2.1 is executed every time flight conditions change or if the variation of helicopter mass exceeds a user-defined tolerance. In this way, the computationally demanding airloads calculation is only performed when substantial changes in helicopter trim state occur without any significant loss of accuracy. The acoustic sphere generation, as well as, far field noise propagation procedure explained in section 3.4.2 is executed at each and every trajectory waypoint, as described in the flowchart of Fig. 3.7. In this context, the noise data calculated on the acoustic sphere surface directly reflects the specific flight state of the rotorcraft, with the sphere observer positions being directly adjusted to each individual ground observer position.

The acoustic sphere generation method proposed in this study becomes beneficial when combined with an appropriate set-up of the ground observer grid. The structure of the ground observer grid determines the accuracy and computational efficiency of the proposed approach at mission level. Regular rotorcraft operations cover hundreds of km² areas. The complex ground noise patterns require high resolution to adequately capture the associated acoustic phenomena. However, the use of a uniform and dense grid across the entire

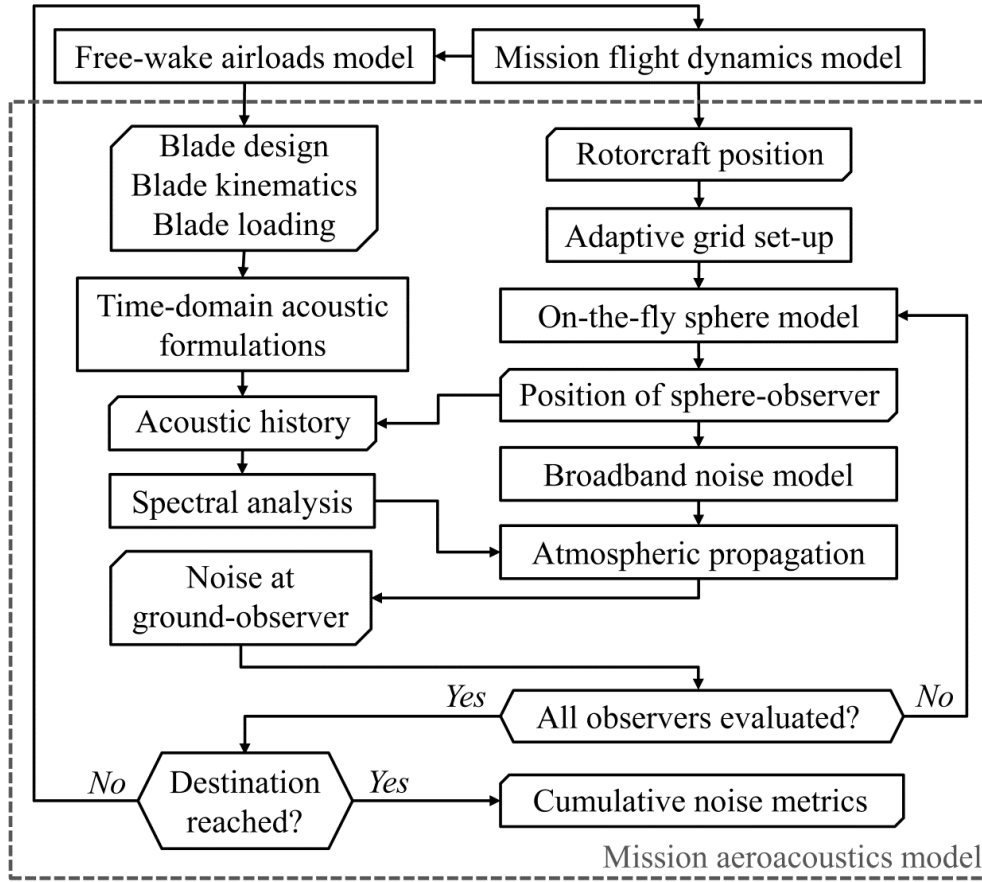


Figure 3.7: Integrated mission simulation numerical procedure.

mission space is prohibitive, due to the limitations in computational time and Random Access Memory (RAM) capabilities of present workstations. Considering that the prominently affected ground regions at each trajectory waypoint are those that lie in the vicinity of the rotorcraft, it can be understood that high grid resolution is only necessary at limited ground regions across the trajectory.

A non-uniform and trajectory-adaptive ground grid is implemented for this study. The entire mission area is covered by a relatively coarse grid. For the area below the rotorcraft, a dense grid is utilised to capture the complex ground noise patterns, as shown in Fig. 3.6. The position of the dense-grid region is continuously updated during the mission, according to the rotorcraft motion. The width as well as the resolution of both coarse and dense-grid regions is

determined by the user. An observer grid sensitivity analysis is carried out and analysed in section 6.2 to form a reliable set of rules for the tuning of the adaptive grid and determination of the employed resolutions. Finally, the free wake code, as well as the aeroacoustics code have been parallelised utilising the Open Multi-Processing (OpenMP) application programming interface (Ref. [141]), which supports shared memory multiprocessing parallel programming. The computational savings of parallelisation are reported in section 6.7.

3.5 Multi-Disciplinary Optimisation of Morphing Helicopter Rotors

3.5.1 Design space exploration and system approximation

It has been shown that the three step integrated procedure described in Section 3.4.3 can be deployed for the aeroacoustic characterisation of helicopter rotors at any designated operating conditions. This method can now be used for the multi-disciplinary optimisation of morphing helicopter rotor systems. A dedicated multi-disciplinary optimisation framework is developed to obtain optimum rotor speed and blade twist control of the main rotor, leading to concurrent reductions in both noise and NO_x emissions. The optimisation algorithm is an expansion of the work presented in Ref. [112], implementing multiple objective functions in a multi-disciplinary manner. The method caters for the inherent non-linearity associated with rotorcraft aerodynamics, aeroelasticity, aeroacoustics and engine and combustor performance deploying a computationally efficient strategy capable of evaluating multiple design iterations. The system's response is evaluated in terms of two objective functions representing specific disciplines of rotorcraft aviation: emitted noise and emitted NO_x .

The three-step integrated procedure comprising HECTOR, the free-wake aerodynamic solver and ICARUS is deployed for the evaluation of rotor aeroacoustics referring to a single trim state, as described in Section 3.4.3. In this manner, an acoustic sphere is obtained, representative of the designated trim state and flight conditions. The acoustic sphere is defined as a set of individual observers forming a spherical fictitious surface surrounding the main rotor. A unique OASPL value corresponds to each of the sphere points,

calculated via the mathematical formulations described in Sections 3.2.2 and 3.4.3. It is noted that, the highest annoyance to human hearing occurs at frequencies between 1-5 kHz which is the frequency range where BVI noise usually occurs. On the other hand, low-frequency noise between 20-100 Hz, is related to adverse physiological (cardiovascular and respiratory) as well as psychological effects on human health (Ref. [142]). No low-frequency weighted metrics exist in the open literature. Therefore, the un-weighted version of the OASPL metric has been selected for the elementary noise evaluations in order to resolve both low and higher frequency harmonics with no bias.

A cumulative metric of acoustic disturbance representative of the entire acoustic sphere is deployed, calculated through energetically averaging the OASPL over the acoustic sphere data points:

$$EAOASPL = 10 \log_{10} \left(\frac{1}{N} \sum_{i=1}^N 10^{\frac{OASPL(i)}{10}} \right), \quad (3.26)$$

where $EAOASPL$ is the energetically averaged OASPL in dB and N is the number of data points across the surface of the acoustic sphere. The $EAOASPL$ metric has been successfully utilised in trajectory optimisation for helicopter noise abatement (Refs. [143], [144]).

The rate of nitrogen oxides (NO_x^{rate}) constitutes the second objective of the optimisation process, to investigate the trade-off between pollutant and noise emissions in descent. The prediction of NO_x^{rate} at a specific flight state is carried out with HEPHAESTUS generic software, originally developed by Celis *et al.* at Cranfield University for the modelling of combustor emissions of civil aero-engines (Ref. [145]). HEPHAESTUS has been further developed by Goulos *et al.* (Ref. [146]) for the prediction of pollutant emissions of helicopter turboshaft engines. HEPHAESTUS is based on a stirred reactor model combined with a set of simplified chemical reactions. HEPHAESTUS has been extensively described and utilised in a number of turboshaft engine studies (Refs. [146], [147]) hence further elaboration on modelling aspects will be omitted.

Having established the objectives of the optimisation process, a Design Space Exploration (DSE) is essential for the identification of system's response to variation of design parameters (variable rotor speed and active blade twist angle) and operational conditions (rotorcraft AUM, density altitude, flight speed and climb rate). The DSE approach comprises two parts; (a) the Design of Experiments (DOE) to populate the design space, and (b) the construction of Response Surface Models (RSMs) to approximate the system's non-linear response. A DOE is a systematic approach to get the maximum amount of information out of a given sample (Ref. [112]). A full-factorial design is implemented to populate the design space. Factorial sampling allows different weighting to be given to each of the design and operational variables based on importance. Additionally, it allows expansion of the design space without distorting the original set of solutions. Fig. 3.8 presents an overview of the DSE method implemented for the purposes of this work.

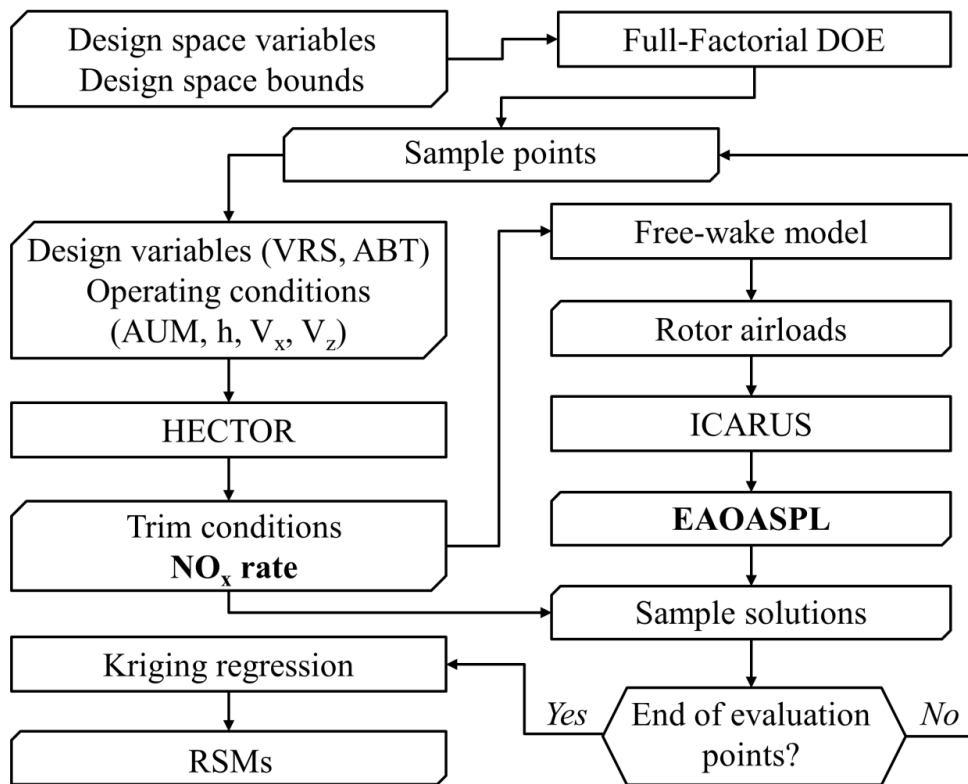


Figure 3.8: Overview of Design Space Exploration approach.

Subsequently, the RSMs are structured using Gaussian Process Regression (Ref. [148]) or Kriging interpolation across the sample data obtained from DOE. Kriging interpolation has been successfully utilised in a number of studies on rotorcraft engine performance optimisation (Refs. [91], [112]). The purpose of using RSMs instead of direct HECTOR/Free-wake/ICARUS runs at each evaluation of the optimisation process is to mitigate the excessive computational overhead associated with the multiple simulations related to rotorcraft aeromechanics, aeroacoustics, engine performance and combustor emissions. The predictive accuracy of the derived RSMs is assessed using the classical Leave-One-Out Cross-Validation (LOOCV) method, as presented in Ref. [149]. Having established an accurate and computationally efficient DSE approach for the approximation of the non-linear system under investigation, the available design space can be systematically explored to identify optimum VRS and ABT control schedules targeting the concurrent mitigation of environmental and acoustic impact of morphing rotor helicopters.

3.5.2 Optimisation approach

The Non-dominated Sorting Genetic Algorithm II (NSGA-II), originally developed by Deb *et al.* (Ref. [150]) has been selected for the multi-disciplinary optimisation carried out in this work. NSGA-II is an elitist evolutionary algorithm which performs sorting of the population into a hierarchy of non-dominated Pareto fronts. The elite solutions of each generation are directly transformed to the next generation, which prevents back-tracking of the algorithm. The NSGA-II is a global optimisation algorithm, which makes the method immune to the danger of getting trapped within design space regions containing local optimum solutions (Ref. [112]). The final output of the multi-disciplinary optimisation process is essentially a Pareto front, which is a set of optimal (non-dominated) solutions. The Pareto-optimal set of solutions are directly utilised to investigate the trade-off between the multiple objectives of the optimisation process. The onus is on the designer to select one or more specific point(s) of the Pareto front, based on problem requirements and design intent.

3.6 Discussion

This chapter has described the methodology developed to address the objectives set within this work. An existing free-wake aeroelastic rotor model has been employed to predict unsteady rotor airloads and elastic blade motion. A rotor noise prediction method has been developed, implementing mathematical formulations for the integral solution of FW-H equation. Formulations for the prediction of thickness and loading noise (including BVI) in the time-domain have been employed. A semi-empirical model for the rapid estimation of broadband noise contribution has been utilised. Compact expressions of the acoustic formulations have been employed to reduce computational time and enable direct coupling with the lifting-line-based approach adopted in the free-wake aerodynamics model.

A computationally-efficient method has been developed for the quantification of uncertainty in aeroacoustics due to vortex core modelling uncertainty. The method has been based on non-intrusive Polynomial Chaos expansion implemented via least-squares regression. A sensitivity analysis method utilising Sobol indices has been deployed to quantify the impact of each source on the overall system uncertainty.

An integrated numerical procedure has been established for the prediction of rotor acoustic impact across complete three-dimensional rotorcraft operations. The flight dynamics code HECTOR has been coupled with the free-wake airloads model and the developed aeroacoustic model to predict rotor source noise on-the-fly, without the need of pre-stored acoustic data. A far-field noise propagation module has been utilised to account for atmospheric absorption and spreading losses. A trajectory-adaptive ground observer grid has been introduced to reduce computational requirements and enable mission-level simulation capability.

A method for the multi-disciplinary optimisation of novel morphing rotor helicopters has been deployed. The novel rotor incorporates combined variable rotor speed and active blade twist. A design space exploration strategy has been implemented based on full-factorial sampling to identify the interaction

between the multiple noise generation mechanisms across the complete investigated design space. The system's response has been approximated through Kriging-based interpolation. A global multi-objective optimisation algorithm has been employed for the identification of optimal solutions for minimum noise and NO_x at any given operating condition of the descent flight envelope. The optimal solutions have been expressed as Pareto fronts, which allow the direct investigation of the trade-off between noise and NO_x emissions of helicopters in descent.

4 Results and Discussion; Helicopter Rotor Source Noise Prediction

4.1 Introduction

The mathematical formulation presented in Section 3.2 is numerically implemented for modelling helicopter rotor aeroacoustics. This formulation comprises fundamental modelling methods for the prediction of helicopter rotor thickness and loading noise in the time domain based on the integral solutions of the original FW-H equation. The aerodynamic input necessary for noise predictions is provided by a free-wake aeroelastic rotor model, capable of resolving unsteady blade aerodynamics including BVIs. The coupled aerodynamics-aeroacoustics model will serve as the core of the methods developed in Sections 3.3-3.5 referring to uncertainty quantification, mission simulation and novel-concept optimisation, respectively. Hence, a thorough validation is required to assess the accuracy, as well as limitations of the developed models.

Ref. [111] describes the development and validation of free-wake aerodynamics model in terms of elastic blade motion, wake geometry and unsteady airloads. For this reason, further elaboration on aeroelasticity and wake geometry validation will be omitted in this work. Nevertheless, an in-depth investigation of specific parameters of the tip-vortex core growth model is carried out in the present work. The tip-vortex core radius is one of the governing parameters which significantly affect airloads and consequently, noise. Thus, a detailed validation of the airloads predictive capability is carried out utilising publicly available experimental data. A dedicated investigation of the capability of capturing BVIs is also carried out and the impact of BVIs on source noise generation is discussed.

The accuracy of acoustic pressure prediction in the time domain is assessed by comparing predictions against published experimental data. An extensively instrumented and validated rotor model is employed to highlight the applicability and robustness of the noise prediction methodology. FFT analysis of both predicted and experimental acoustic histories is carried out to identify the spectral limitations of the developed method. Finally, predictions of ground acoustic impact are compared against measured noise carpets, primarily focusing on the capture of the most challenging component of helicopter acoustic spectrum: BVI noise.

4.2 Unsteady Airloads

The hingeless HART II rotor model (Ref. [32]) is employed for the assessment of unsteady airloads prediction conducted in this work. The HART II rotor is a 40% scale model of the hingeless Bo 105 helicopter rotor. The four-bladed rotor has a radius $r = 2\text{ m}$, a linear twist angle $\theta_{tw} = -8\text{ deg}$ per unit span, a chord $c_b = 0.121\text{ m}$ and rotates at $\Omega = 109.12\text{ rad/s}$. The summary of all rotor

Table 4.1: HART II rotor geometrical characteristics, retrieved from Ref. [32]

Parameter	Value	Unit
Blade count	4	blades
Rotor radius	2.00	m
Blade chord	0.121	m
Rotor solidity	0.077	-
Rotational speed	109.12	rad/s
Non-dimensional root cut-out distance	0.22	-
Blade linear twist per span	-8	deg
Precone angle	2.5	deg
Aerofoil	NACA23012	-

geometrical parameters is provided in Table 4.1. The HART II rotor was used in the Higher harmonic control Aeroacoustics Rotor test program (Ref. [151]) conducted in the German-Dutch Wind Tunnels (DNW). Specifically, the rotor was operated in the 8 m x 6 m cross-section open jet configuration of the DNW (Ref. [32]). The aim of HART II project was to comprehensively assess a geometrically and dynamically scaled helicopter rotor in terms of elastic blade deformation, leading-edge pressure distributions, tip-vortex flowfield, acoustic pressure and ground noise. The HART II database is considered one of the most comprehensive and reliable data sets available for model rotors.

The rotor operating conditions represent low-speed descent flight, where BVIs are most pronounced (Ref [32]). Specifically, the rotor is trimmed at thrust coefficient $C_T = 0.00457$, rolling moment coefficient $C_L = 0.00001386$ and pitching moment coefficient $C_M = -0.00001386$. The rotor reference shaft angle of attack is $\alpha_s = 5.3 \text{ deg}$ nose-up. Due to wind tunnel wall interference effects,

Table 4.2: HART II wind tunnel ambient and operational conditions, retrieved from Ref. [32]

Parameter	Value	Unit
Air pressure	100,970	Pa
Air temperature	290.45	K
Air density	1.2055	kg/m ³
Freestream velocity	32.9	m/s
Rotor shaft angle	5.3	deg
Wind tunnel interference angle	-0.8	deg
Thrust coefficient	0.00457	-
Rolling moment coefficient	0.00001386	-
Pitching moment coefficient	-0.00001386	-

an effective angle of attack $a_{eff} = a_s + \Delta a_s = 4.5 \text{ deg}$ has been suggested to be used in simulation campaigns (Ref. [152]). The advance ratio is $\mu = 0.151$ where it was observed that at rotor azimuth angles $\psi = 50 \text{ deg}$ and $\psi = 300 \text{ deg}$, the axis of tip-vortices is almost parallel to the blade leading edge. This results in strong BVI phenomena (Ref. [152]). Table 4.2 summarises the wind tunnel operating conditions of the HART II experiment.

In the HART II test, the impact of Higher Harmonic Control (HHC) on rotor noise and vibrations was assessed. HHC was applied through 3/rev variation at the blade pitch with an angle $\theta_3 = 0.8 \text{ deg}$ and a phase angle ψ_3 (Ref. [32]). The total blade pitch distribution is then defined as function of the rotor azimuth angle ψ and the normalised radial location r/R :

$$\begin{aligned} \theta(r/R, \psi) = & \theta_{tw}(r/R - 0.75) \\ & + \theta_{el}(r/R, \psi) \\ & + \theta_{.75} + \theta_{1s} \sin \psi + \theta_{1c} \cos \psi, \\ & + \theta_3 \cos(3\psi - \psi_3) \end{aligned} \quad (4.1)$$

where θ_{tw} is the blade pre-twist angle, θ_{el} is the elastic twist deformation, $\theta_{.75}$ is the rotor collective pitch angle applied at $r = 0.75R$, θ_{1s} is the rotor longitudinal cyclic pitch control angle and θ_{1c} is the lateral cyclic pitch control angle. Three different HHC cases were defined based on the value of phase angle: the baseline case (BL) where HHC was not applied and $\psi_3 = 0 \text{ deg}$, the minimum noise case (MN) where HHC was applied with $\psi_3 = 300 \text{ deg}$ and the minimum vibration case (MV) where HHC was applied with $\psi_3 = 180 \text{ deg}$.

As discussed in Section 3.2.1, different tip-vortex core radii and Squire parameter values may be used for the evaluation of induced velocity for wake calculations from those used to compute induced velocity for rotor inflow calculations, to improve convergence stability and speed (Ref. [32]). For the present analysis, the values of $r_{c0} = 0.1c$ and $\alpha_1 = 0.001$ are used for wake calculations, and $r_{c0} = 0.6c$ and $\alpha_1 = 0.008$ for rotor inflow calculations. These

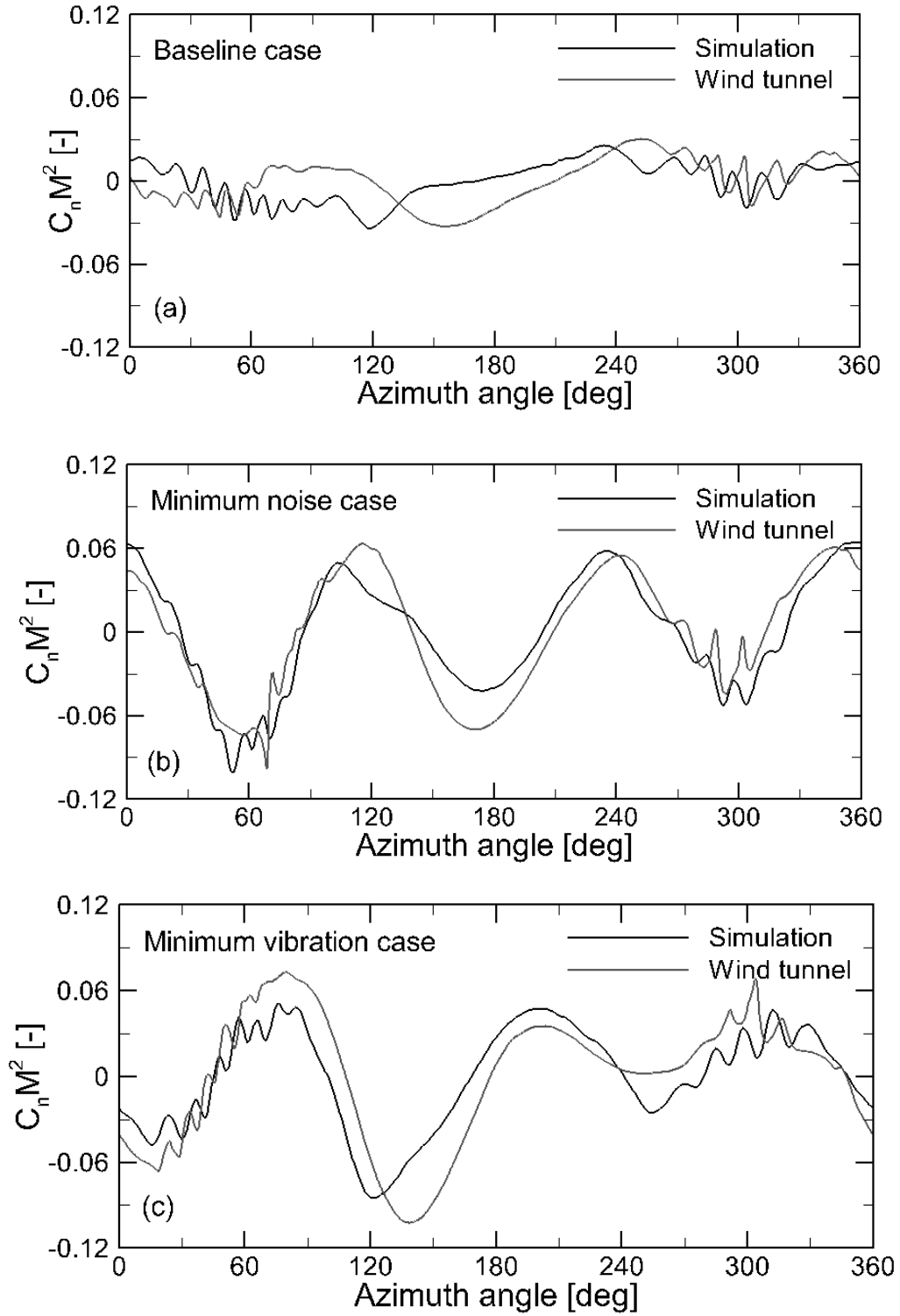


Figure 4.1: Comparison of calculated normal force coefficient with HART II experimental data retrieved from Ref. [32]; (a) Baseline case, (b) Minimum noise case, (c) Minimum vibration case.

values provide the best match between predicted and measured wake, airloads and noise data. The rotor azimuthal step implemented is $\Delta\psi = 1 \text{ deg}$ to efficiently capture the BVI airloads. The blades are discretized into 30 radial elements and ten vibration modes are considered. The aforementioned resolutions have been selected after sensitivity analyses and literature suggestions (Ref. [37]). The near wake is discretized with an azimuthal step of $\Delta\zeta = 10 \text{ deg}$ and truncated at the age of $\zeta = 30 \text{ deg}$. The far wake comprises tip-vortices, resolving three wake rotations for the calculation of induced flowfield.

The normal force coefficient $C_n M^2 = T / (0.5 \rho_0 c_0^2 c_b)$ is the metric employed for the assessment of unsteady airloads predictive capability. This metric is a normalised expression of the elementary thrust force T acting normal to the rotor tip-path-plane. Fig. 4.1 a) presents the calculated azimuthal distribution of normal force coefficient at 0.87R, compared with wind tunnel data referring to the BL case of the HART II test, extracted from Ref. [32]. The mean value of normal force coefficient has been removed from both distributions. It is observed that high frequency fluctuations of the normal force coefficient occur around rotor azimuth angles $\psi = 50 \text{ deg}$ and $\psi = 300 \text{ deg}$, which can be attributed to the close proximity between wake tip-vortices and blades, as it will be shown in Section 4.3. These essentially constitute a potential source of BVI noise. It can be seen that the phase and amplitude of normal force coefficient high-frequency fluctuations are fairly predicted by the rotor airloads model. On the other hand, a phase shift $\Delta\psi = 35 \text{ deg}$ between predicted and measured data is observed at the low frequency negative peak of the normal force coefficient. According to Ref. [111], this is due to the under-prediction of blade elastic twist by the aeroelastic rotor model at this azimuthal region.

Fig. 4.1 b) presents the predicted and measured time-histories of normal force coefficient for the MN case. It is observed that the amplitude of low-frequency loading is significantly increased compared to the BL case. On the other hand, the amplitude of high-frequency loading has been reduced in the MN case around $\psi = 30 \text{ deg}$ and $\psi = 90 \text{ deg}$, as well as around $\psi = 270 \text{ deg}$ and

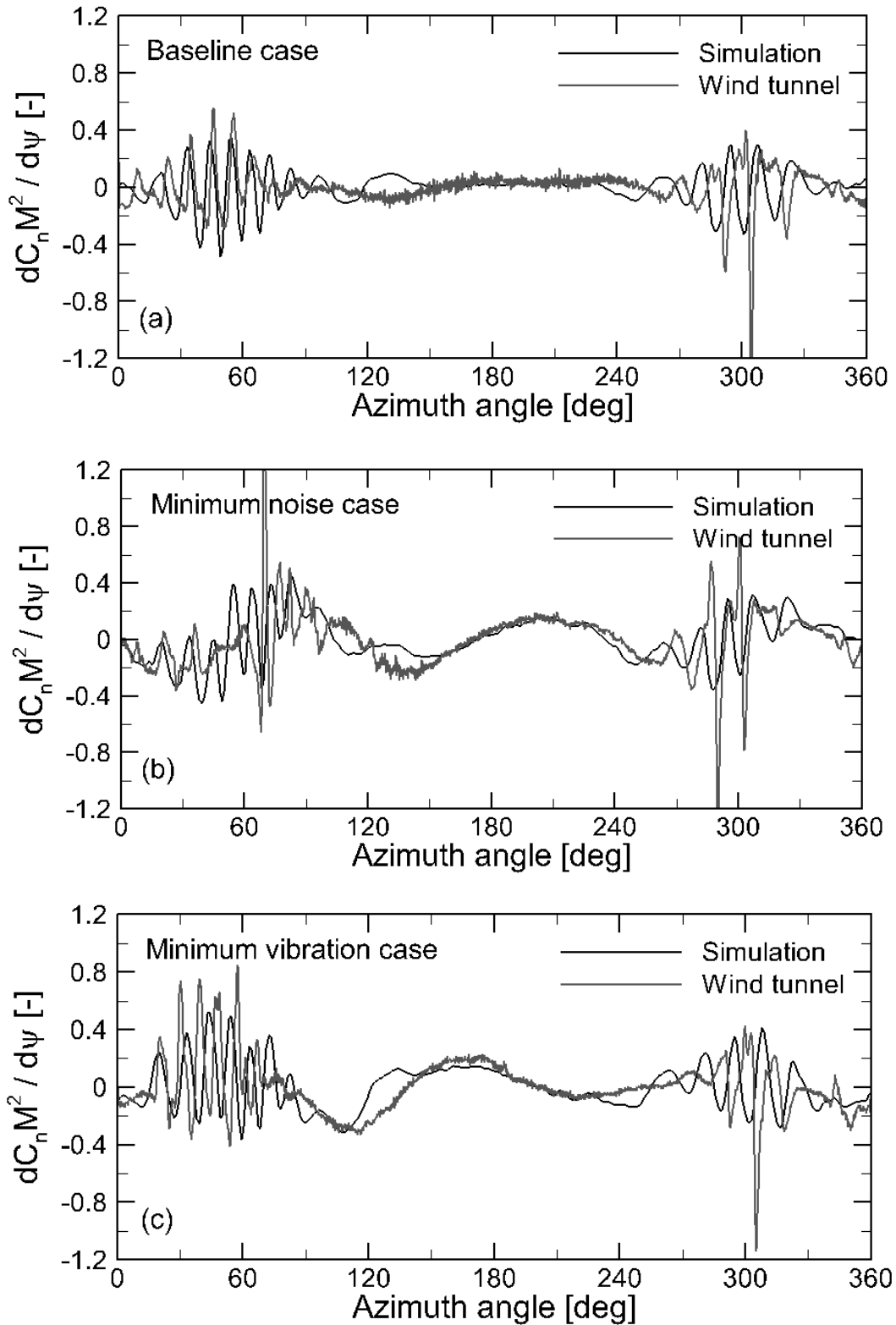


Figure 4.2: Comparison of calculated time-derivative of normal force coefficient with HART II experimental data retrieved from Ref. [32]; (a) Baseline case, (b) Minimum noise case, (c) Minimum vibration case.

$\psi = 330 \text{ deg}$, relative to the BL case. Nevertheless, the amplitude of high-frequency loading at $\psi = 50 \text{ deg}$ and $\psi = 300 \text{ deg}$ is comparable to the amplitudes encountered in the BL case. In terms of predictive capability, very good agreement is shown between predicted and measured normal force coefficient, both in terms of amplitude and phase. Fig. 4.1 c) depicts the corresponding normal force coefficient distributions of the MV case. Very good agreement is observed, albeit with slightly higher discrepancies which are attributed to the under-prediction of torsion elastic angle around $\psi = 90 \text{ deg}$ and $\psi = 150 \text{ deg}$, as reported in Ref. [111].

Fig. 4.2 a) illustrates the time-derivative of normal force coefficient calculated for the predicted and measured airloads of the BL case. The time-derivative is evaluated using a five-point stencil finite difference scheme; hence achieving fourth-order accuracy of the derivative estimation. The time-derivative of normal force coefficient constitutes a metric representative of unsteady loading noise terms for out-of-plane observers, where BVI noise usually dominates. It can be confirmed that the most intensive fluctuations occur at rotor azimuth angles $\psi = 50 \text{ deg}$ and $\psi = 300 \text{ deg}$. Very good phase agreement between measured and predicted fluctuations is observed at the advancing side, with slight amplitude discrepancies, which are attributed to the $C_n M^2$ discrepancies identified around $\psi = 30 \text{ deg}$.

As regards the retreating side, a slight phase shift between the simulation and wind tunnel results is observed, due to the phase shift encountered in the retreating side $C_n M^2$ distribution. Moreover, a strong negative peak at $\psi = 305 \text{ deg}$ calculated in the experimental data cannot be captured by the airloads model. This strong peak is a result of the sudden reduction of $C_n M^2$ at the same azimuthal location, which can be attributed to the high flap angle measured for the Blade 1 of the HART II experiment (Ref. [111]). In the HART II measurements, it was found that Blade 1 consistently presented higher flapping angles compared to the rest of the blades of the model rotor. This was due to

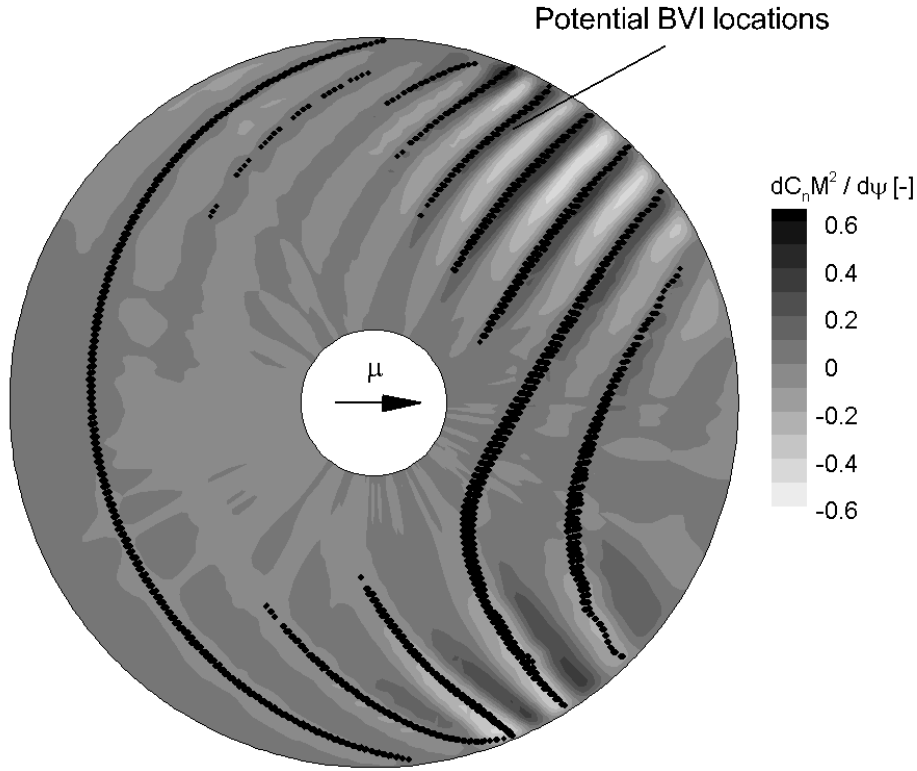


Figure 4.3: Distribution of the time-derivative of normal force coefficient across the rotor disc for the HART II baseline case and comparison with potential BVI locations calculated with the spherical detection method.

the retro-fit of additional pressure sensors on Blade 1 (Ref. [32]), which affected its elastic behaviour and consequently, local induced velocity and airloads.

Fig. 4.2 b) presents the corresponding time-histories of normal force coefficient time-derivative of the MN case, for both predicted and wind tunnel data. A slight phase shift is observed, mainly at the advancing side. Additionally, the strong peaks calculated at $\psi = 70 \text{ deg}$ and $\psi = 290 \text{ deg}$ of the experimental distribution data are not captured by the model. These peaks are attributed to the higher flapping angles measured for Blade 1 at the corresponding azimuthal locations, as documented in Ref. [32]. Overall, good amplitude agreement is observed for the rest of the distribution of time-derivative of $C_n M^2$. Similarly for the MV case, the strong negative peak measured at $\psi = 310 \text{ deg}$ is not captured by the

simulation, as illustrated in Fig. 4.2 c). Apart from that discrepancy, only slight phase and amplitude deviations between predicted and wind tunnel data are recorded.

4.3 Blade-Vortex Interactions

The capability of predicting high-frequency unsteady airloads has been verified in Section 4.2. The aerodynamic phenomenon which acts as dominant source of these high-frequency oscillations is the interaction between blades and tip vortices. BVIs occur when the rotor tip-vortices are in close proximity with the rotating blades. According to Eq. 3.3, the small miss-distance between blades and tip-vortices, results in high components of induced velocity on the rotor disc, which eventually lead to strong fluctuations of aerodynamic loading. Fig. 4.3 presents the distribution of the time-derivative of normal force coefficient across the rotor disc, as calculated for the BL case of the HART II experiment. As discussed in Section 4.2, the most intensive oscillations are located around $\psi = 50 \text{ deg}$ at the advancing side and $\psi = 300 \text{ deg}$ at the retreating side of the rotor disc. In the same figure, the potential BVI locations are superimposed and represented with circle markers. The potential BVI locations are identified via the spherical detection method (Ref. [34]) using a critical miss-distance equal to one blade chord. It can be seen that the detected potential BVI locations coincide with the regions of strong fluctuations of unsteady airloads, which confirms the BVI predictive capability of the employed models. Specifically, at the advancing side, a trailed tip-vortex propagates at a direction opposite to the direction of rotation of the upcoming blade, which induces an upwash to the leading edge of the upcoming blade and eventually, a positive impulse in the airloads history. At the retreating side, the trailed tip-vortices propagate at a direction same to the blade rotation direction, which leads to an upwash in the vicinity of the trailing edge of the retreating blade; hence a negative impulse in the airloads history. For this reason, the potential BVI locations of the advancing side coincide with the positive peaks of the time-derivative of normal force coefficient, whilst at the retreating side, they match the negatives.

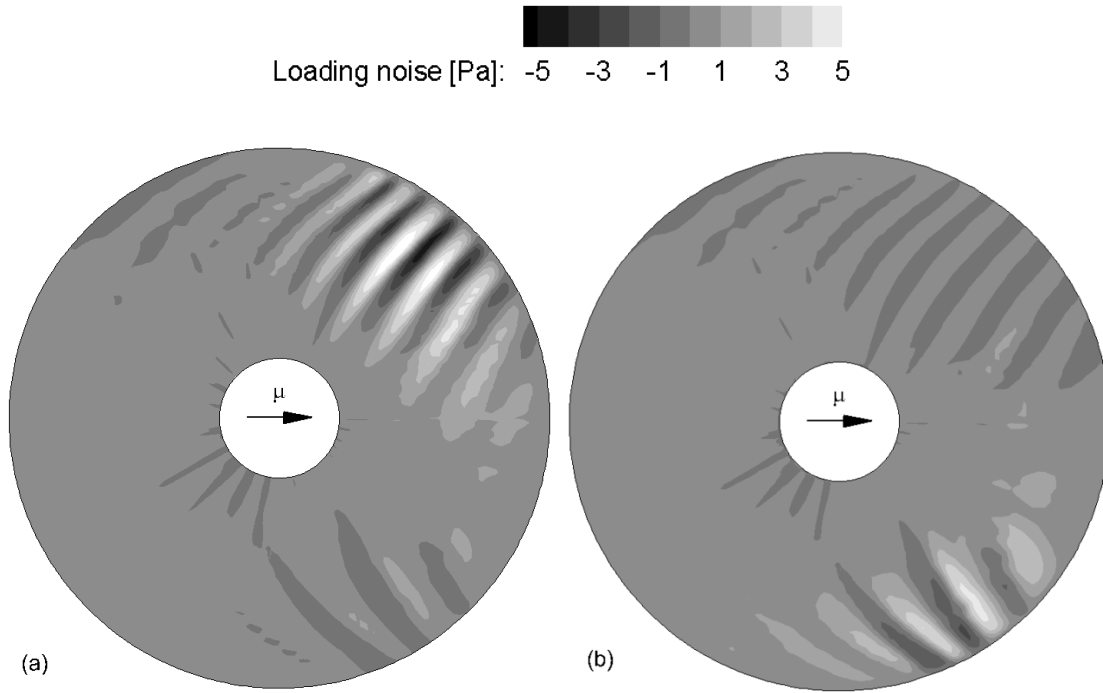


Figure 4.4: Loading noise generation across the rotor disc for the HART II baseline case; (a) advancing side observer, (b) retreating side observer.

Having established and validated a methodology for the prediction of BVI events in the vicinity of the rotor, one can investigate the first-order effects of impulsive airloads on BVI noise generation. Fig. 4.4 a) depicts the distribution of loading noise across the HART II BL case rotor disc, referring to the advancing side observer located 2.215 m below the rotor disc, 0.0 m upstream and 1.81 m at the advancing side of the rotor. It is noted that only the loading noise contributions have been isolated for the calculation of the acoustic pressure at this specific observer within this investigation. Loading noise is calculated using the dipole acoustic term of the Formulation 1A of Farassat, as expressed in Eq. 3.11. The illustrated elementary noise contributions correspond to the source time (i.e. emission time) and no propagation losses have been considered. It can be seen that this acoustic observer is primarily affected by the advancing side BVIs, due to the directivity of the loading vector time-derivative at this region of the rotor disc. At the advancing side of the rotor disc, this vector becomes almost parallel to the unit radiation vector pointing towards the

observer under investigation, leading to up to ± 5 Pa elementary loading noise contributions from the rotor disc.

Similarly, Fig. 4.4 b) presents the corresponding loading noise contributions which refer to the BL case's retreating side acoustic observer located 2.215 m below the rotor disc, 0.92 m downstream and 2.00 m at the retreating side of the rotor (Ref. [153]). The observer is dominated by impulsive noise of BVI nature, primarily generated at the retreating side of the rotor disc, where loading-noise directivity is maximised. Overall, the retreating-side contributions present smaller magnitudes compared to the advancing side. Thus, it is expected that the BVI acoustic impact of the HART II experiment will be stronger in the ground region below the advancing side of the rotor.

4.4 Acoustic Pressure

The generated acoustic pressure across the rotor disc is essentially propagated in the near and far field. As discussed in Section 3.2.2, the Formulation 1A of Farassat (Eq. 3.10-Eq. 3.15) is utilised for the calculation of near field noise in the vicinity of the rotor. An advanced-time (i.e. source-time dominant) algorithm is implemented to evaluate the acoustic signal in the observer time-space, as described in Ref. [124]. The acoustic pressure history is calculated at the two observer locations selected in the HART II experiment and compared with the corresponding microphone measurements, retrieved from Ref. [153]. Fig. 4.5 a) illustrates the comparison of predicted and measured acoustic pressure history at the advancing-side observer located 2.215 m below the rotor disc, 0.0 m upstream and 1.81 m at the advancing side of the rotor. The rotor is trimmed according to the settings of the BL case, as described in Section 4.2. Differential microphones are not able to capture steady noise components (Ref. [48]). For this reason, the mean value is removed from the predicted acoustic signal, to be directly comparable with the measured one. Both thickness and loading noise components are resolved in the calculations, although the contribution of thickness noise to out-of-plane observers is typically negligible. The acoustic signal comprises four distinct pulses due to the blade count of the HART II rotor. It is observed that the measured acoustic signal is not entirely periodic due to

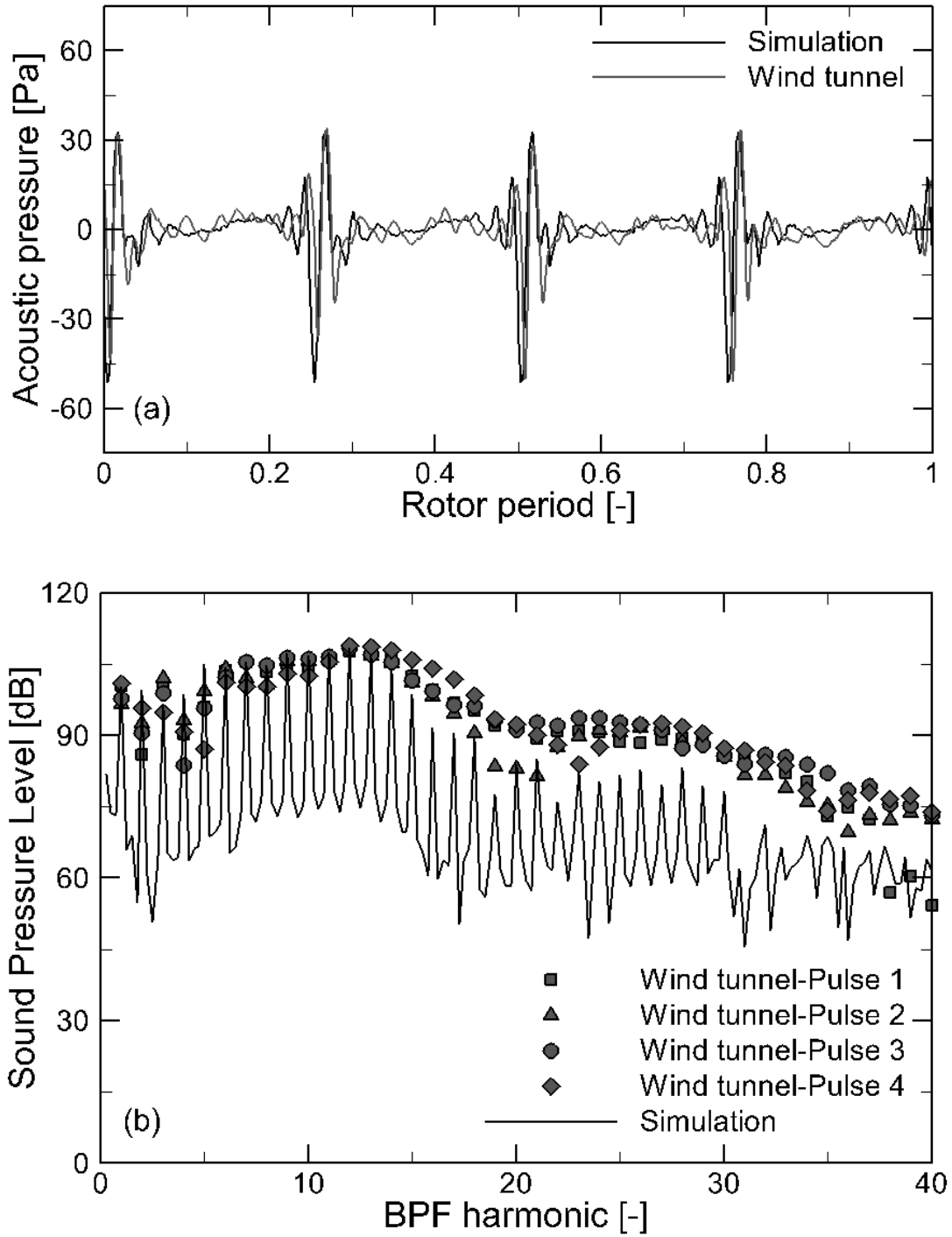


Figure 4.5: Comparison of predicted acoustic pressure history with experimental measurements retrieved from Ref. [153] for the HART II baseline case at a single observer located 2.215 m below the rotor disc, 0.0 m upstream and 1.81 m at the advancing side of the rotor; (a) acoustic pressure, (b) spectral analysis.

the additional pressure sensors installed on Blade 1, which caused flapping and loading discrepancies relative to Blades 2-4. The signal obtained from simulation is essentially periodic. It is illustrated that the developed method accurately predicts the generated BVI noise in terms of amplitude and phase.

A spectral comparison of the acoustic pressure time histories is required to quantify the predictive accuracy of the aeroacoustic formulation. Fig. 4.5 b) presents the FFT of the predicted signal, compared with the FFT of the measured signal for the first 40 harmonics, where BVI noise is dominant (Ref. [32]). Since the steady component of the acoustic pressure has been removed, the 0th harmonic component of the FFT has zero amplitude. Due to the aperiodicity of the acoustic signal, each of the pulses of the measured signal is individually analysed. In this manner, four distinct FFT signals occurred referring to the experimental data. As regards the low-frequency content of the noise signal, good agreement between predictions and measurements is observed for the first five BPF harmonics (deviation between 1-10dB), albeit with discrepancies associated with the 4th and 5th harmonic. These can be attributed to the deviations observed around the second negative peak of each acoustic pulse and especially in pulses 3 and 4. It needs to be mentioned that the deviation calculated between the 2nd and the 4th pulse of the 5th harmonic of the measured FFT signal is over 12 dB, which highlights the level of uncertainty associated with the measured acoustic signal in the low-frequency region.

As regards the higher-frequency BVI content, an average deviation of 3.1dB is observed in the range between the 6th and the 18th BPF harmonic which depicts the BVI predicting capabilities of the developed methodology. Generally, an average deviation of 6.5dB is observed up to the 30th BPF harmonic. Higher discrepancies are recorded at the higher harmonics which showcase the limitations of chordwise compact aerodynamic and aeroacoustic formulations. Nevertheless, the amplitude of higher harmonic noise components is significantly smaller; hence the impact of not accurately capturing noise above the 30th BPF harmonic will not significantly affect the cumulative metrics of ground noise impact.

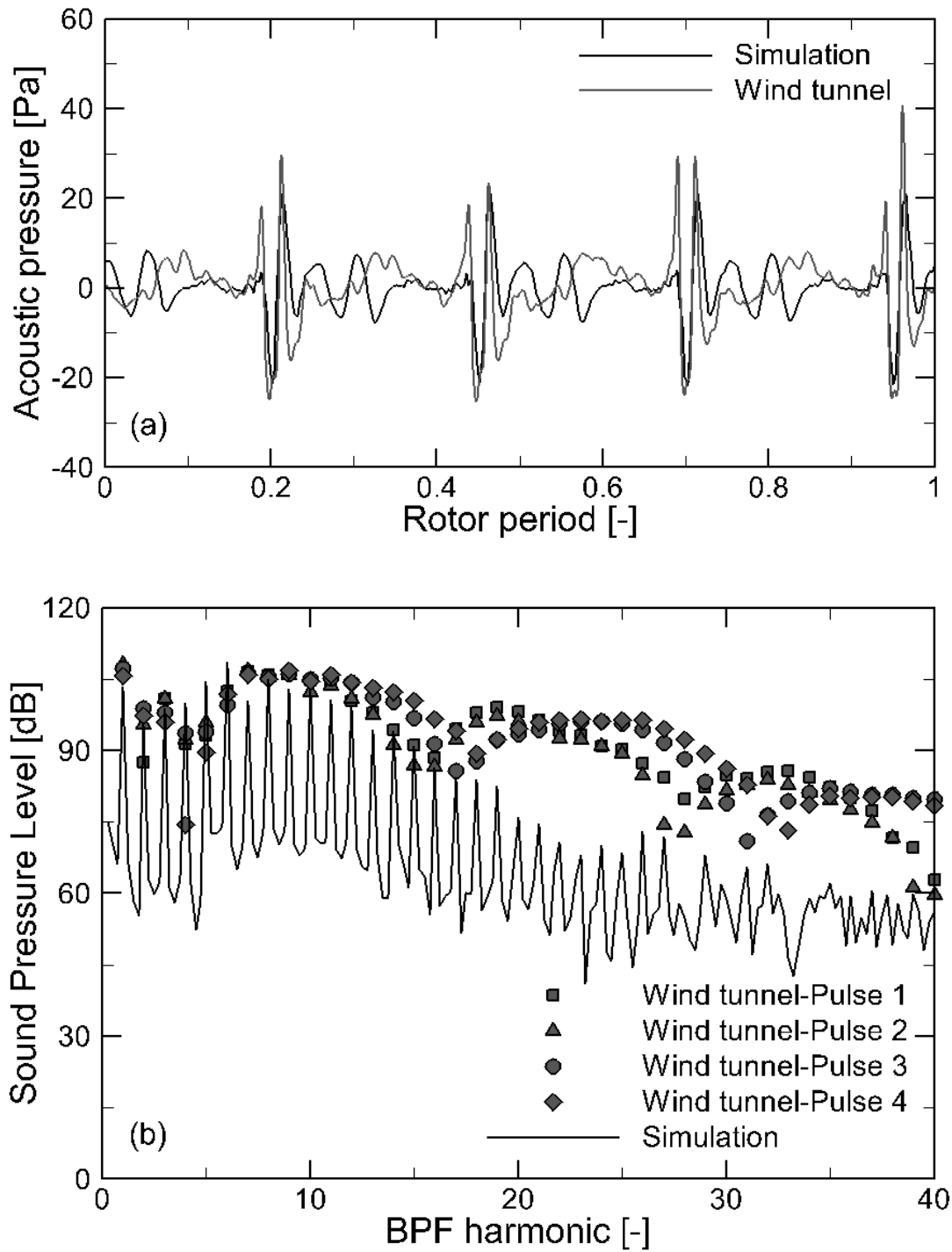


Figure 4.6: Comparison of predicted acoustic pressure history with experimental measurements retrieved from Ref. [153] for the HART II baseline case at a single observer located 2.215 m below the rotor disc, 0.92 m downstream and 2.00 m at the retreating side of the rotor; (a) acoustic pressure, (b) spectral analysis.

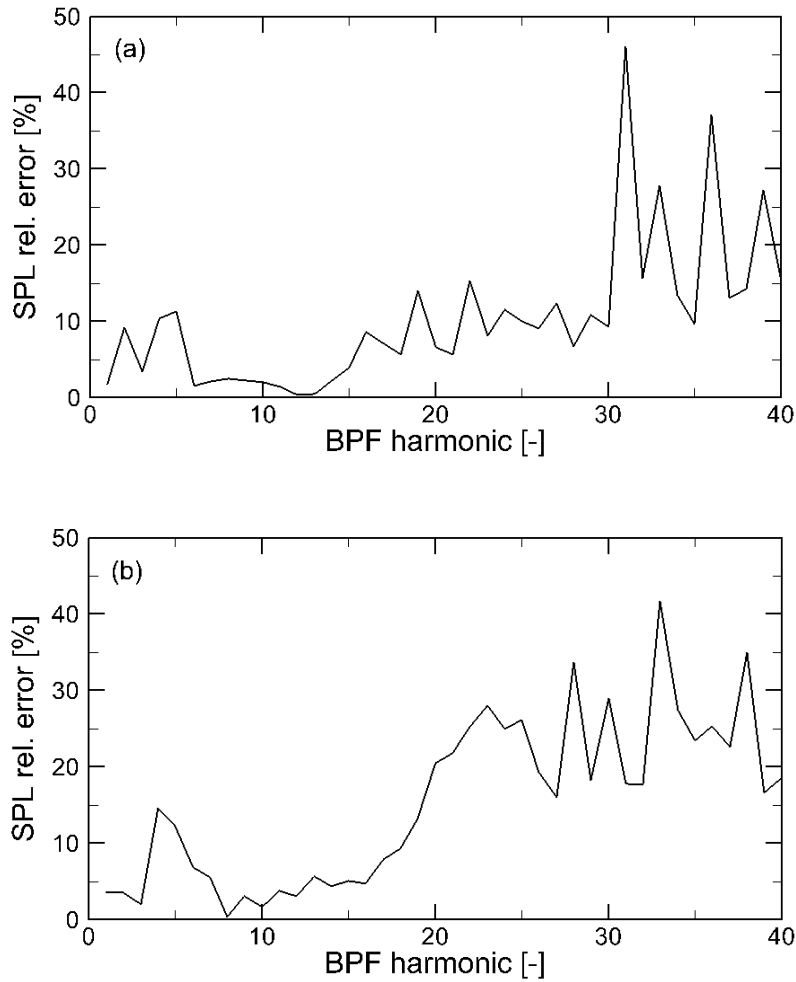


Figure 4.7: Sound pressure level relative error between predicted and measured acoustic pressure in the spectral domain; (a) advancing side observer, (b) retreating side observer.

Fig. 4.6 a) presents the predicted and measured acoustic pressure histories that correspond to the retreating-side observer of the BL case, located 2.215 m below the rotor disc, 0.92 m downstream and 2.00 m at the retreating side of the HART II rotor (Ref. [153]). Significant aperiodicity is observed in the measured data, especially around the second positive BVI peak of each of the four pulses. The simulation is not able to capture the first positive BVI peak observed in the experimental signal due to the inability of capturing the retreating side airloads negative peak, as discussed in Section 4.2. Similar to

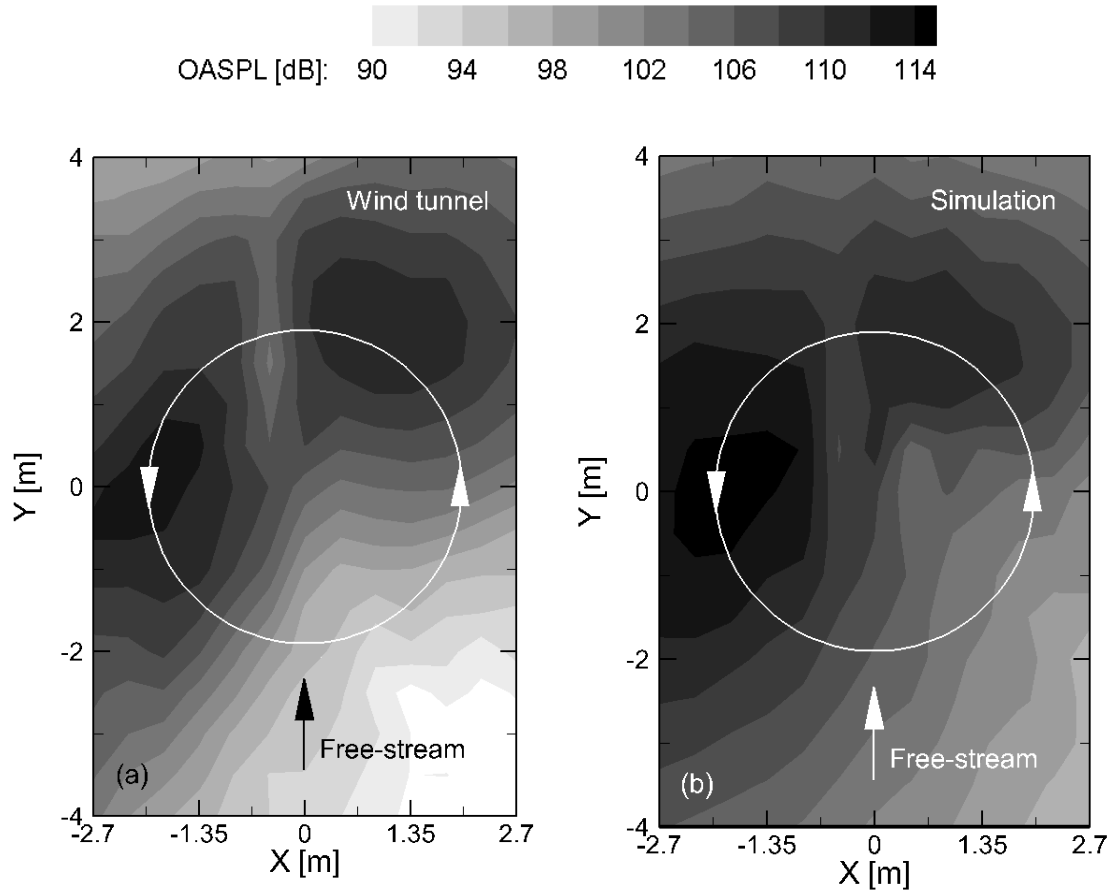


Figure 4.8: Comparison of ground noise 2.215 m below the rotor disc of the baseline case of HART II rotor with measured data retrieved from Ref. [32]; (a) wind tunnel, (b) simulation.

the advancing side, deviations are observed between measured and predicted results at the 4th-6th BPF harmonics. The BVI dominated spectral content (6th-19th) is very well captured by the developed model. High discrepancies are observed at higher BPF harmonics, primarily due to the insufficient capture of the first positive BVI peak of each acoustic pulse.

Fig. 4.7 a) and Fig. 4.7 b) summarise the relative Sound Pressure Levels (SPL) error between experimental and predicted data for the advancing and retreating side observers, respectively. The average value of the four acoustic pulses is utilised at each BPF harmonic of the measured signal for this comparison. Generally, the advancing side predictive capability is superior compared to the

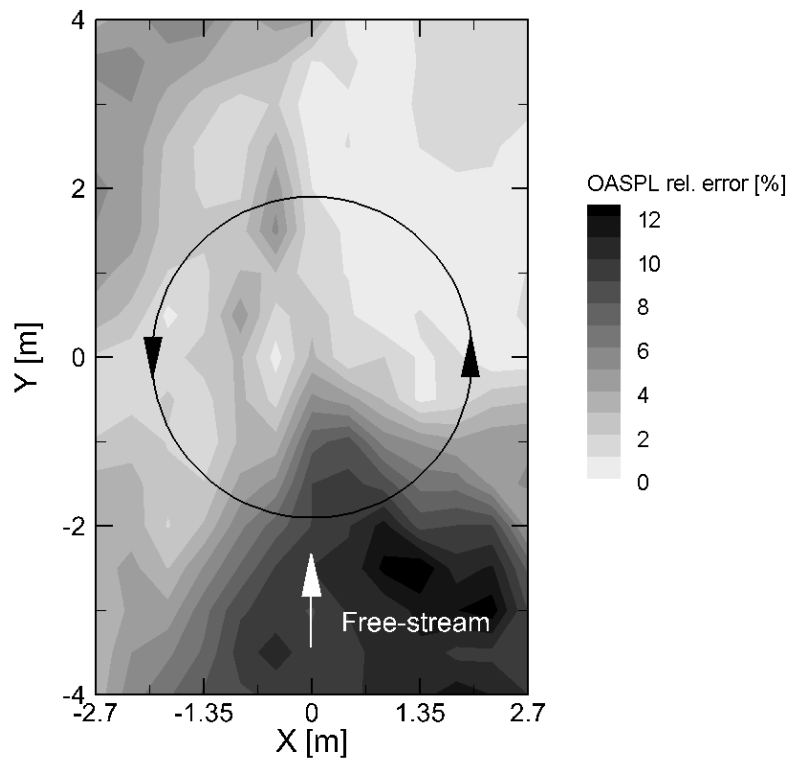


Figure 4.9: Distribution of Overall Sound Pressure Levels average relative deviation between predicted and measured ground noise for the baseline case of the HART II experiment.

retreating side. Specifically, the acoustic content can be reliably predicted at frequencies up to the 30th BPF harmonic for advancing side out of plane observers, with a mean relative error (MAE) of 6.54%. The retreating side observer acoustic history can be properly predicted at frequencies up to the 19th BPF harmonic, at out-of-plane observers, with a MAE of 5.82%. Overall, the developed aeroacoustic methodology is successful in predicting all the major acoustic effects associated with complex BVI phenomena in an accurate manner and only within a fraction of a second.

4.5 Ground Noise

The prediction of complex acoustic patterns has been validated at single observer locations, thus, the investigation should be expanded to complete noise carpets below helicopter rotors. Fig. 4.8 a) illustrates the ground noise

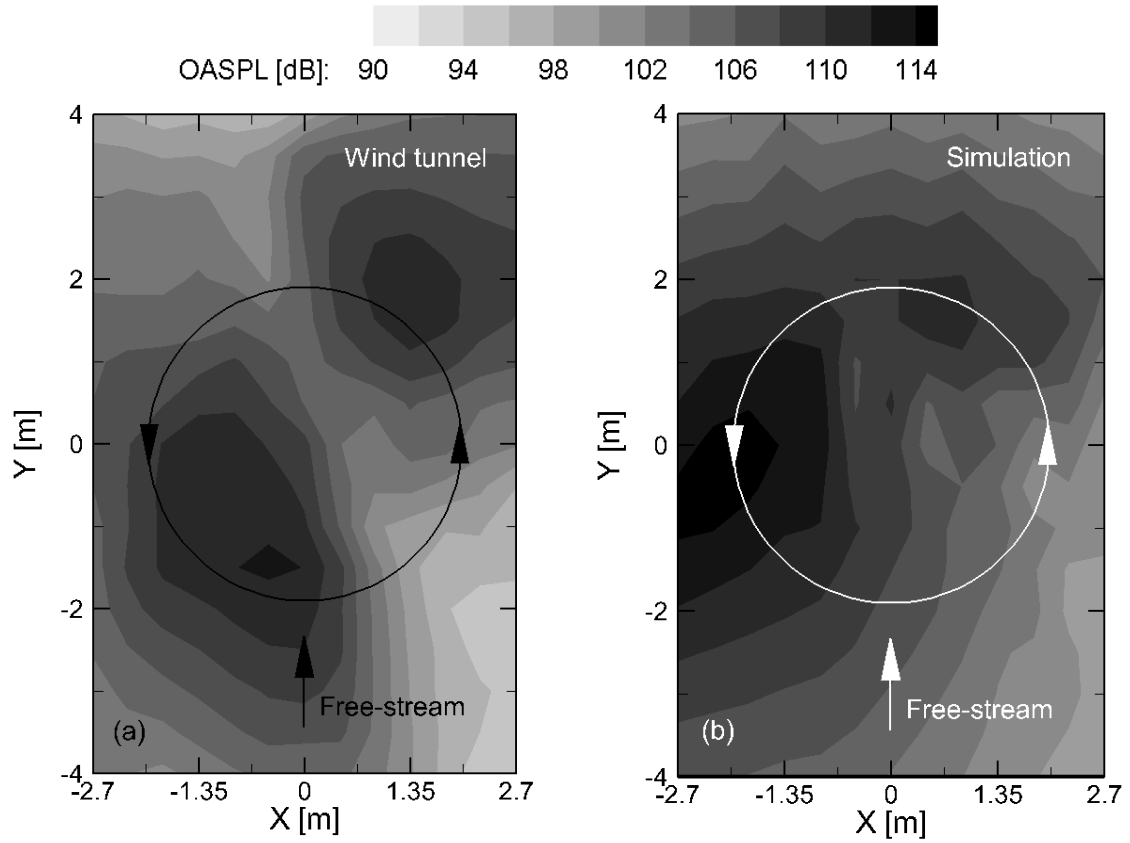


Figure 4.10: Comparison of ground noise 2.215 m below the rotor disc of the minimum noise case of HART II with measured data retrieved from Ref. [32]; (a) wind tunnel, (b) simulation.

contours corresponding to the BL case of the HART II experiment. The measurement plane is located 2.215 m below the rotor disc and extends 4 m upstream and downstream of the hub centre and 2.7 m left and right of it. The measuring microphone array comprises thirteen differential microphones equally distributed in the lateral direction. Measurements were taken at seventeen equally distributed traverse positions of the microphone array, as described in Ref. [32]. The Overall Average Sound Pressure Levels (OASPL) in decibels (dB) is calculated via frequency-integration of the FFT on the original acoustic pressure signal at microphone location. Only BPF harmonics between the 6th and the 40th were considered to isolate the mid-frequency content of the emitted noise, where BVI noise is dominant. A strong BVI noise peak is observed below the advancing side of the rotor, around $\psi = 90^\circ$, as a result

of the impulsive fluctuations observed in the normal force coefficient around $\psi = 50 \text{ deg}$. A BVI peak of lower intensity is formed below the retreating side of the rotor, around $\psi = 330 \text{ deg}$ due to the retreating side normal force fluctuations.

Fig. 4.8 b) presents the predicted ground noise for the same conditions using the developed computational models. A total of 221 ground observers are employed, to ensure consistency with the measurement grid utilised in the HART II experiment. It can be seen that both BVI peaks are accurately predicted in terms of directivity and intensity. Fig. 4.9 depicts the distribution of relative error between predicted and measured results. Very good accuracy is observed at the BVI-dominated ground regions, with the higher deviations located at the low-noise region, upstream of the rotor. The MAE of this

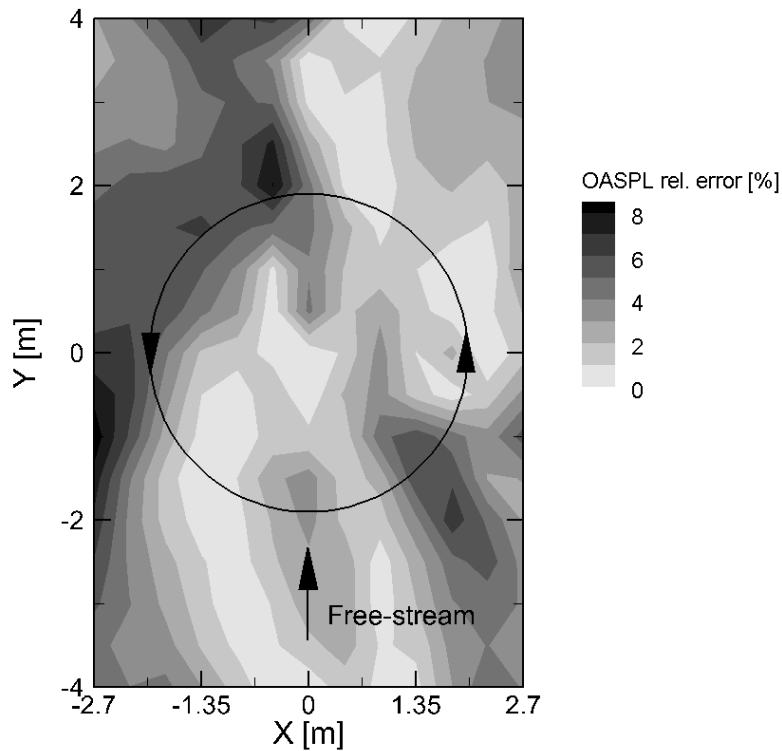


Figure 4.11: Distribution of Overall Sound Pressure Levels average relative deviation between predicted and measured ground noise for the minimum noise case of the HART II experiment.

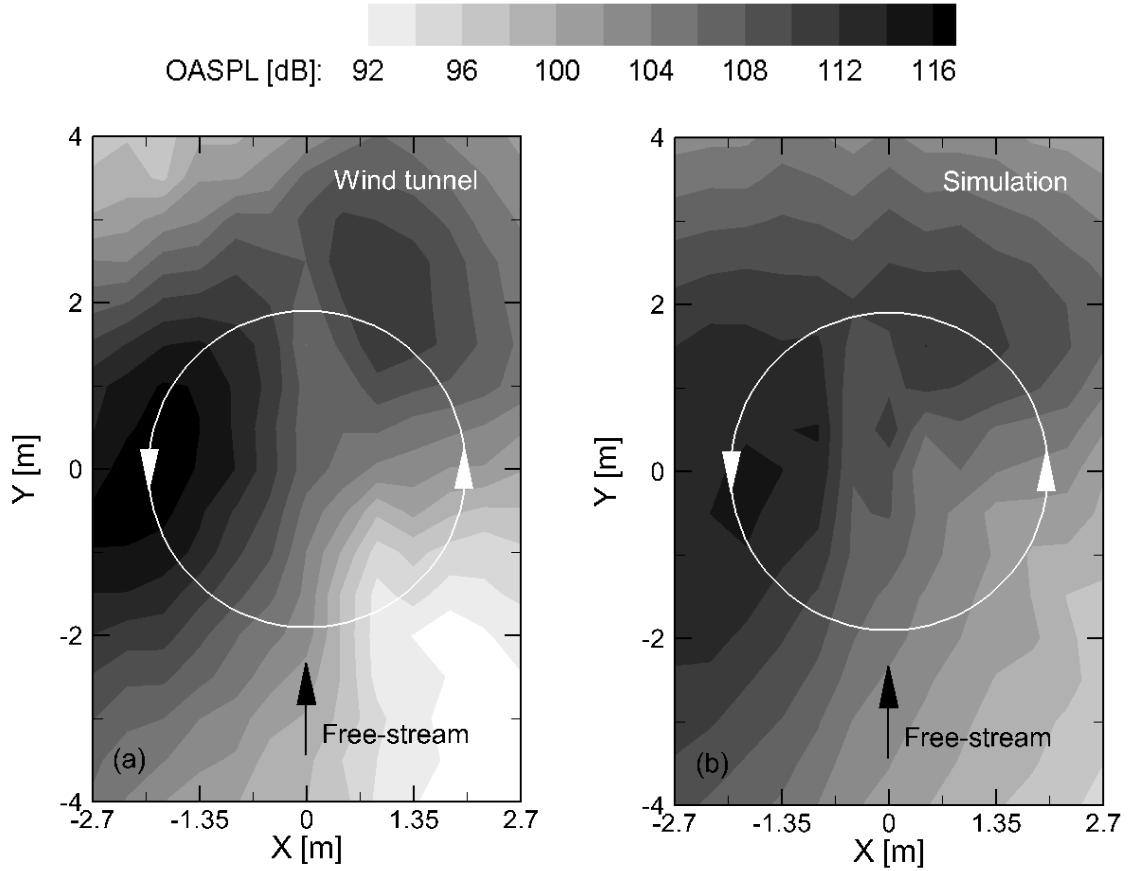


Figure 4.12: Comparison of ground noise 2.215 m below the rotor disc of the minimum vibration case with measured data retrieved from Ref. [32]; (a) wind tunnel, (b) simulation.

distribution is 4.18% which highlights the capabilities of the deployed methodology. The computed MAE can be translated to an average absolute deviation of 4.1 dB, which can be deemed as satisfactory given the complexity of the simulated case. It is noted that the experimental scatter of the noise radiation was found to be up to 2 dB (Ref. [32]) which is comparable to the average absolute error of the BL simulation.

Fig. 4.10 a) illustrates the mid-frequency ground noise contours of the MN case of the HART II experiment. The details of the measurement grid are identical to the BL case, thus further elaboration on the test conditions shall be omitted. It can be seen that the noise levels have been generally mitigated which depicts the effectiveness of HHC on BVI noise reduction. Additionally, the directivity of

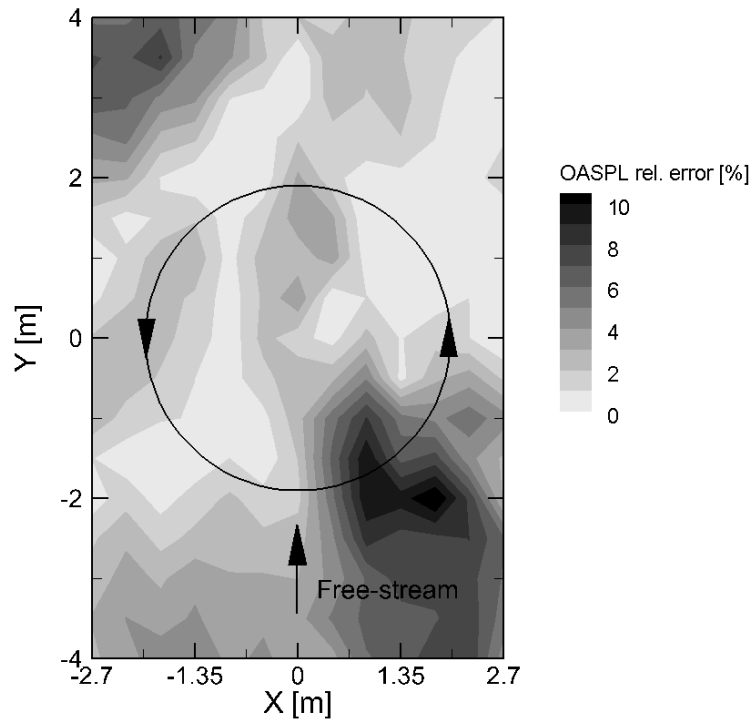


Figure 4.13: Distribution of Overall Sound Pressure Levels average relative deviation between predicted and measured ground noise for the minimum vibration case of the HART II experiment

advancing side peak has moved towards the centre, compared to the BL case. As seen in Fig. 4.10 b), the developed model does not capture this change of directivity, which can be attributed to the advancing side discrepancies identified in the time derivative of normal force coefficient for this case. It is noted though that this change of directivity has not been captured by comparable state-of-the-art noise codes, as reported in Ref. [32]. Overall, the predicted noise carpet reflects the reduction in noise levels due to HHC, as depicted in the error distribution of Fig. 4.11. The MAE calculated between predicted and measured BVI noise levels, is 3.00%, which corresponds to an absolute deviation of 3.11 dB.

Fig. 4.12 a) presents the measured BVI noise contours of the MV case of the HART II experiment, employing the same measurement grid as in BL and MN cases. As reported in Ref. [152], dual vortices are generated in this case, which

interact with the rotor at the advancing side. This essentially introduces increased BVI noise below the advancing side, which is confirmed by the measured noise carpets of this case. This trend is depicted in the noise predictions carried out with the developed method, as illustrated in Fig. 4.12 b). Nevertheless, the advancing-side peak is slightly under-predicted, due to the underprediction of the time-derivative of normal force coefficient, presented in Section 4.2. Generally good amplitude and directivity agreement is observed between the predicted and experimental data. Fig. 4.13 presents the distribution of relative error, which has a MAE of 2.93%. The calculated average absolute noise deviation is 2.93 dB, due to the very good agreement observed between measured and predicted airloads.

4.6 Computational Grid Convergence

The numerical uncertainty associated with the numerical implementation of the noise prediction method needs to be investigated before utilisation in aeroacoustic investigations. In this section, the discretisation error due to the following parameters is discussed: (a) rotor blade radial discretisation, (b) rotor azimuthal discretisation, (c) near-wake azimuthal discretisation, (d) near-wake extent, (e) far-wake extent. The computational grid is assessed in terms of the following two aeroacoustic descriptors: (a) amplitude of acoustic pressure p'_{amp} at the advancing-side ground observer of the HART-II experiment and (b) $OASPL$ at the same observer location, as described in Section 4.4.

Table 4.3: Summary of grid convergence investigated cases.

Parameter		Values	
1 / radial elements	1/60	(1/30)	1/15
Rotor azimuth step [deg]	0.5	(1)	2
Near wake azimuth step [deg]	5	(10)	20
1 / Near wake extent [1/deg]	1/60	(1/30)	1/15
1 / Far wake rotations	1/4	(1/2)	1/1

Roache (Ref. [154]) proposed a method which provides an objective asymptotic approach for the quantification of numerical uncertainty in grid convergence. Specifically, three computational grids are constructed for each of the five parameters under investigation. The grids are constructed based on a constant grid refinement ratio $r_{refine} = 2$. Table 4.3 summarises the values corresponding to each parameter for the three grid cases. A reference value is allocated for each of the five parameters, annotated in parenthesis and bold font. Each parameter is assessed individually with the other four retaining their reference values.

The value of p'_{amp} and $OASPL$ is calculated for each computational grid and all the discretisation parameters. The theoretically exact values of the acoustic objectives are estimated based on Richardson Extrapolation (Ref. [154]),

$$p'_{amp}{}^{exact} = p'_{amp}{}^1 + \frac{p'_{amp}{}^1 - p'_{amp}{}^2}{r_{refine}^n - 1} \text{ for an order } n = \ln \left(\frac{p'_{amp}{}^3 - p'_{amp}{}^2}{p'_{amp}{}^2 - p'_{amp}{}^1} \right) / \ln(r_{refine}), \text{ with}$$

$i=1,2,3$ corresponding to each grid level, from finer to coarser, respectively. Similar extrapolation is carried out for $OASPL^{exact}$. The exact values represent the solution that would theoretically occur after an infinitely fine numerical grid implementation. For this reason, the inverse values of number of radial blade elements, near wake extent and far wake extent have been preferred in order to match the zero value of the discretisation step corresponding to the solution of the Richardson Extrapolation.

Grid convergence is firstly assessed via quantifying the actual fractional error (Ref. [154]) of the three evaluated grids with respect to the exact solution. Relative deviations are provided for acoustic pressure amplitude,

$$\varepsilon_{p'_{amp}}^i = \frac{p'_{amp}{}^i - p'_{amp}{}^{exact}}{p'_{amp}{}^{exact}} [\%], \text{ whilst dB differences are preferred for overall noise,}$$

$$\varepsilon_{OASPL}^i = OASPL^i - OASPL^{exact} [dB], \text{ with } i=1,2,3 \text{ corresponding to each grid level, from finer to coarser, respectively. Fig. 4.14 (a) presents the relative deviations of } p'_{amp} \text{ from } p'_{amp}{}^{exact} \text{ for the five discretisation parameters under}$$

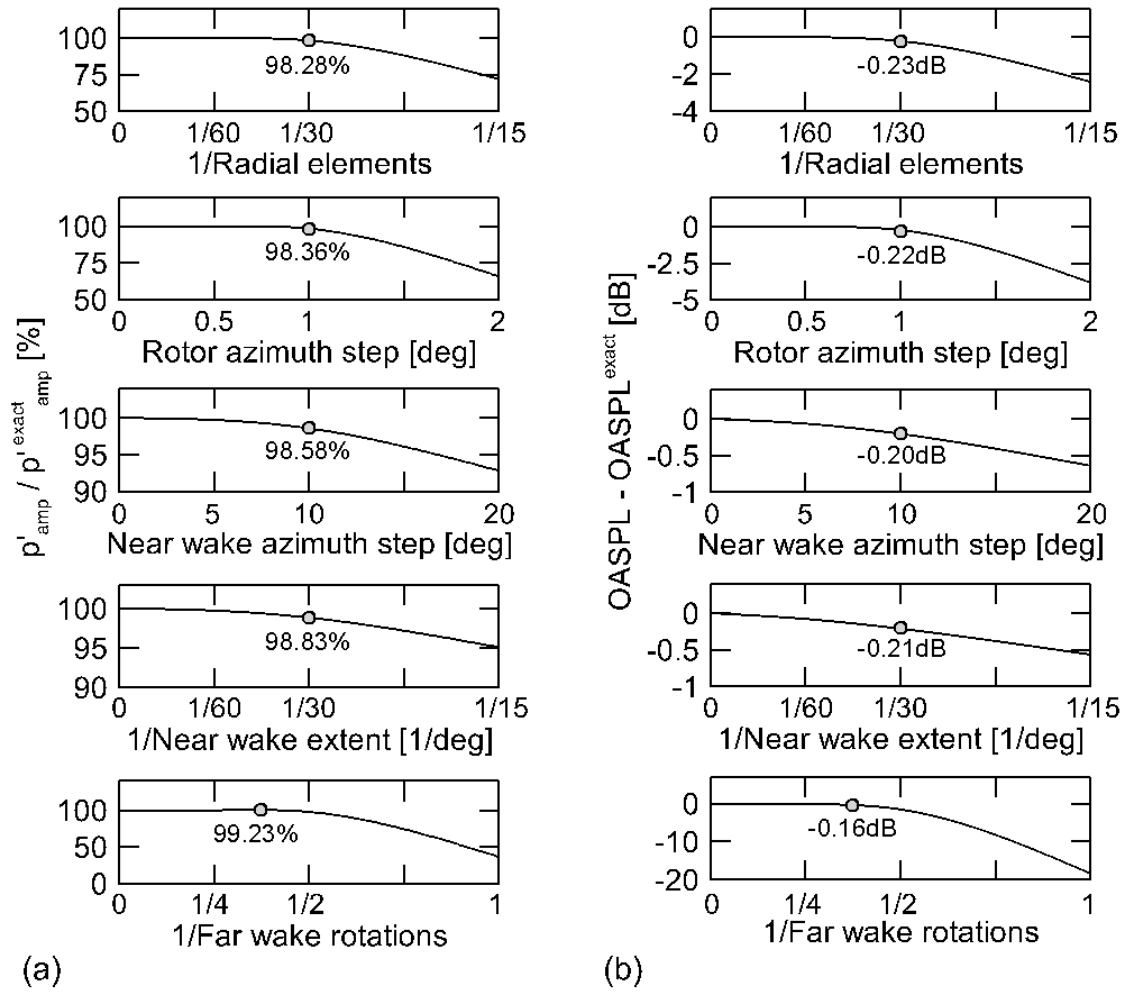


Figure 4.14: Distribution of actual fractional error as function of the discretisation parameters: (a) evaluation for p'_{amp} and (b) evaluation for $OASPL$.

investigation. It can be seen that the settings corresponding to the medium grid fineness feature deviations between 98.28% and 99.23%. The width of the vertical axis is a metric of the sensitivity of the solution to variations of discretisation parameters. It can be seen that rotor radial and azimuthal discretisation, as well as far-wake extent are the parameters with the highest impact on grid convergence.

Fig. 4.14 (b) presents the corresponding dB deviations for the $OASPL$ output objective. Deviations between -0.23dB and -0.16dB are observed between the

Table 4.4: Grid Convergence Indices for p'_{amp} and $OASPL$.

Parameter	p'_{amp}		$OASPL$	
	GCI_{12} [%]	GCI_{23} [%]	GCI_{12} [dB]	GCI_{23} [dB]
1 / radial elements	0.31	5.23	0.06	0.72
Rotor azimuth step	0.23	4.92	0.04	0.69
Near wake azimuth step	0.84	4.31	0.17	0.61
1 / Near wake extent	0.84	3.55	0.21	0.63
1 / Far wake rotations	0.26	7.17	0.33	4.61

medium-fineness grid solution and the exact solution. The highest sensitivity with respect to $OASPL$ prediction is observed for the extent of the far wake. This is primarily due to the mis-capture of high-frequency acoustic pressure variations resulting from interactions between the far wake tip vortices and the rotor blades.

Further to the actual fractional errors, Grid Convergence Indices $GCI = 3|\varepsilon|/\left(r_{refine}^n - 1\right)$ are calculated to quantify the impact of grid refinement on the numerical convergence of the grid, as suggested in Ref. [154]. The calculated GCIs between grid 1 (fine) and grid 2 (medium) as well as between grid 2 and grid 3 (coarse) are provided in Table 4.4. For p'_{amp} it can be seen that GCI_{12} lies between 0.26-0.84%, whilst the corresponding values for $OASPL$ are between 0.06-0.33 dB. The above numbers suggest that the medium-fineness grid is an appropriate selection for the quantifications of this work, therefore the discretisation employed comprises: 30 radial elements, 1 deg. azimuthal blade step, 10 deg. near wake azimuthal step, 30 deg. extent of the near wake and 3 far-wake rotations.

4.7 Discussion

This chapter has presented an extensive validation study of the developed methodology for helicopter rotor noise prediction. The methodology described in Section 3.2 has been deployed for the comprehensive assessment of noise prediction capabilities, utilising the experimental results of the HART II campaign as benchmark. The model rotor has been trimmed at acoustically complex operating conditions representative of low-speed descent, dominated by BVI noise.

The unsteady airloads predictive capability has been assessed, with emphasis on capturing BVI events in the vicinity of the rotor. Very good agreement was observed which was also depicted in the prediction of the time-derivative of normal force coefficient at different rotor HHC settings. Furthermore, the capabilities of spherical BVI detection method have been explored, towards the identification of the impact of unsteady airloads on BVI noise generation across the rotor disc. Very good agreement has been observed between the predicted BVI locations and the impulsive peaks of unsteady airloads time-derivative.

The acoustic pressure predictive capability has been thoroughly assessed for both advancing and retreating side observers, dominated by BVI noise. Very good agreement was observed, especially for the advancing side observer. The accuracy of predictions has been quantified through an elaborate FFT analysis of the predicted signal and comparison with the corresponding experimental spectral content. Good agreement has been noted in the low-frequency region, although significant scatter of the experimental data was identified. In the BVI-dominated frequency range, very good agreement was observed in the advancing side. Specifically, a MAE of 6.54% was calculated for predictions up to the 30th BPF harmonic. A MAE of 5.82% was observed for predictions up to the 19th BPF harmonic at retreating side observers. Overall, the developed method has been proven to be very accurate in predicting noise at single observers, even for complex BVI noise cases, within only a fraction of a second.

Finally, the assessment of ground noise prediction accuracy has been carried out by comparing complete ground noise contours with experimentally

measured noise carpets, representative of BVI-dominated descent. Very good agreement has been found in most of the test cases and especially at the ground regions dominated by complex BVI noise effects. The average deviation between predicted and measured ground noise data was between 3-4 dB, a value comparable to the 2 dB scatter reported in the experimental noise results.

It can be concluded that the method has been capable of capturing the governing aeroacoustic phenomena associated with the HART II rotor. The variety of results available for this rotor allowed the validation of rotor performance, trim controls, aeroelastic behaviour, wake aerodynamics and unsteady rotor airloads prior to the presented noise validation. To the author's knowledge, no other comprehensive experimental campaign of that extent was found in open literature. However, it should be noted that the mathematical foundation of the employed methods is not configuration-dependent, but it models the associated physical mechanisms. Therefore, the developed methodology has the potential for extension to similar helicopter rotors, provided that comprehensive experimental data are available for validation.

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5 Results and Discussion; Uncertainty Quantification in Aeroacoustic Simulation

5.1 Introduction

In this chapter, the impact of tip-vortex modelling uncertainty on the aeroacoustic analysis of helicopter rotors in BVI conditions is undertaken. Free-wake models are routinely used in aeroacoustic analysis of helicopter rotors; however, their semi-empiricism is essentially accompanied with uncertainty related to tip-vortex core geometry physical parameters. In some cases, analysts have to resort to empirical adaption of these parameters based on previous experimental evidence. Two cases of input uncertainty are investigated: (a) when experimental data about the geometry of tip-vortex core are available and (b) when experimental data are absent. A consistent comparison between the aforementioned cases is presented, based on the uncertainty quantification method introduced in Section 3.3. The output objectives for the assessments are effective descriptors of unsteady airloads, acoustic pressure and OASPL noise.

The predictive accuracy of the Polynomial Chaos expansion is quantified for all the investigated aeroacoustic objectives. Having established computationally efficient and accurate meta-models, the statistical properties of output objectives of interest are quantified by incorporating the polynomial expansion into classical Monte Carlo simulation. The relative contribution of the uncertain input parameters to the overall system's response is quantified via global sensitivity analysis. Non-deterministic predictions of full acoustic pressure histories and ground noise carpets are carried out to identify the impact of vortex-core-related uncertainty on the low- and higher-harmonic acoustic spectrum.

Table 5.1: Distributions of uncertain input variables when tip-vortex experimental data are available (Gaussian distribution with $COV = 10\%$).

Uncertain input	μ	σ	Unit
r_{c0}^{wake}	0.10	0.01	[chords]
a_1^{wake}	0.001	0.0001	[-]
r_{c0}^{inflow}	0.60	0.06	[chords]
a_1^{inflow}	0.008	0.0008	[-]

A generalised analysis at different levels and distributions of input uncertainty is undertaken, to derive holistic maps for the robustness of aeroacoustic predictions. These can serve as modelling guidelines, useful in the pre-estimation of the deviation between not-deterministic and deterministic simulation, as well as the expected uncertainty in simulation of helicopter noise. Furthermore, they can be used for the evaluation of the necessity of non-deterministic simulation based on input data robustness and requirements.

5.2 Polynomial Chaos Expansion

The rotor under investigation is the HART II model rotor (Ref. [32]) analysed in Chapter 4, hence no further description is provided. The uncertain input parameters considered are the tip-vortex core model parameters. Specifically:

(a) tip-vortex initial core radius for wake calculations r_{c0}^{wake} , (b) Squire parameter for wake calculations a_1^{wake} , (c) tip-vortex initial core radius for rotor inflow calculations r_{c0}^{inflow} and (d) Squire parameter for rotor inflow calculations a_1^{inflow} .

It is noted that other aerodynamic modelling parameters such as blade aerofoil aerodynamic properties are also prone to induce uncertainty (Ref. [78]). However, the tip-vortex growth parameters govern the distribution of rotor induced flowfield which affects airloads and consequently, noise. This is especially pronounced in descent where BVIs dominate the rotor flowfield.

Table 5.2: Distributions of uncertain input variables when tip-vortex experimental data are not available (Uniform distribution with $COV = 30\%$).

Uncertain input	a	b	Unit
r_{c0}^{wake}	0.0480	0.1519	[chords]
a_1^{wake}	0.00048	0.00152	[-]
r_{c0}^{inflow}	0.2882	0.9117	[chords]
a_1^{inflow}	0.00384	0.01215	[-]

Two different cases are examined: (a) the case when experimental data referring to tip-vortex core geometry are available and (b) the case of not available data. For the case of available data, the probability density function (PDF) of all input parameters will be considered to be Gaussian with coefficient of variation $COV = 10\%$. This is in correspondence with experimental measurements of tip-vortex core radius and circulation strength undertaken by Bhagwat and Ramasamy (Ref. [90]). The associated values of mean μ and standard deviation σ of the uncertain inputs considered are summarised in Table 5.1. As regards the case of not available experimental data for tip-vortex core correlations, a uniform distribution is used for all uncertain inputs, as they can take any value within the designated limits, with equal probability of occurrence. The determination of the upper limit a and lower limit b for each input is predominantly dependent upon the availability of relevant literature and practical experience. Based on the suggestions of Ref. [30] and Ref. [32], as well as the discussion of Section 3.2.1, an uncertainty of $COV = 30\%$ will be used, which can be considered as conservative choice. The resulting limits of the uniform input distributions are provided in Table 5.2. It is noted that in both examined cases, the mean values of uncertain inputs correspond to the values found that they provide the best agreement between predicted and experimental noise data.

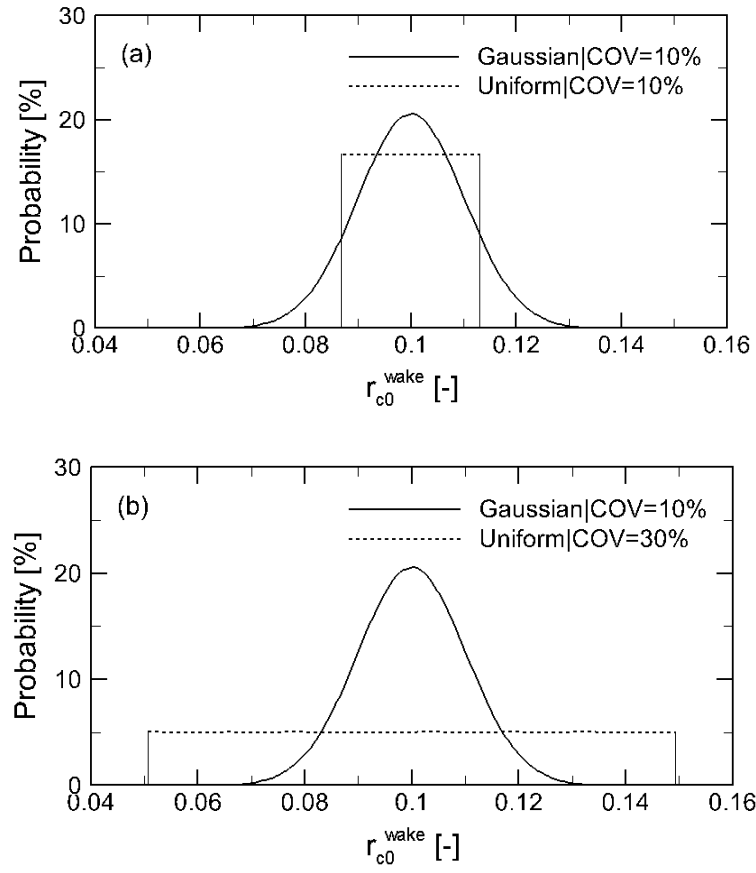


Figure 5.1: Comparison of probability density functions for Gaussian and uniform distributions of r_{c0}^{wake} uncertain input variable; (a) $COV = 10\%$ for both distributions, (b) $COV = 10\%$ for Gaussian and $COV = 30\%$ for uniform.

Fig. 5.1 a) discusses the relative spread and probability density of Gaussian and uniform distributions for r_{c0}^{wake} . It can be seen that for the same levels of uncertainty, $COV = 10\%$, the spread of uniform distribution is significantly narrower than the Gaussian spread. This is due to the definition of σ for both distributions (Eq. 5.1 and Eq. 5.2):

$$COV = \frac{\sigma}{\mu}, \quad (5.1)$$

$$COV = \frac{\sigma}{\mu} = \frac{b-a}{(b+a)\sqrt{3}}, \quad (5.2)$$

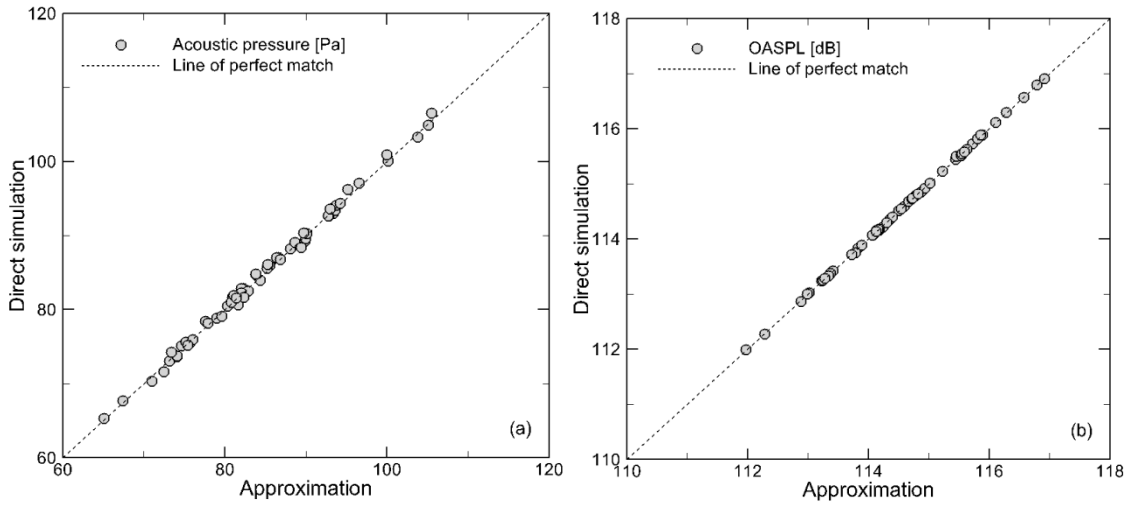


Figure 5.2: Leave-one-out Cross Validation for the case of available tip-vortex experimental data, applied to the developed Polynomial Chaos Expansion, referring to the advancing side observer of HART II experiment; (a) Amplitude of acoustic pressure pulse, (b) OASPL.

Fig. 5.1 b) presents the probability distributions of the r_{c0}^{wake} uncertain input for the cases of availability (Gaussian distribution) and absence (uniform distribution) of experimental data, utilised in this work. The rest of uncertain input parameters are described by similarly shaped distributions. It is noted that for the case of availability of experimental evidence, the COV values reported in Ref. [90] do not exactly match at 10%. However, due to the fundamental character of this study, it has been deemed as more easily interpretable to maintain a common COV for each uncertain variable, thus allowing the expansion and generalization of obtained results, as will be described in Section 5.6.

The design space is discretised based on the sampling method described in Section 3.3. An over-sampling coefficient of 4 is used in this work to generate a reliable sample. A polynomial order $p=2$ is selected in this work, as also followed in Ref. [88]. Based on Eq. 3.24 and for the given number of uncertain inputs $n=4$, a total of 60 collocation points are required for the PCE. The

Table 5.3: Statistics of leave-one-out Cross Validation of the derived PCE when tip-vortex experimental data are available (Gaussian distribution with $COV = 10\%$).

Objective	$N_{\text{Pearson}} [-]$	Gradient [deg]	NRMSE [%]	σ [%]
$\left(dC_n M^2 / d\psi\right)_{amp}^{adv}$	0.999	45.052	0.469	0.473
$\left(dC_n M^2 / d\psi\right)_{amp}^{ret}$	0.995	44.555	1.004	1.011
p'_{amp}^{adv}	0.998	44.683	0.649	0.632
p'_{amp}^{ret}	0.996	45.339	0.977	0.963
$OASPL^{adv}$	0.999	45.004	0.109	0.110
$OASPL^{ret}$	0.996	44.760	0.161	0.169

coupled approach comprising free-wake aerodynamics and ICARUS aeroacoustics is employed for the evaluation of each collocation point, as described in Section 3.2.

The output objectives of interest represent aerodynamic and aeroacoustic parameters which govern BVI noise generation and propagation. Specifically, the objectives chosen for UQ are: (a) amplitude of time-derivative of normal force coefficient at $r = 0.87R$ at the advancing side of the rotor $\left(dC_n M^2 / d\psi\right)_{amp}^{adv}$, (b) amplitude of time-derivative of normal force coefficient at $r = 0.87R$ at the retreating side of the rotor $\left(dC_n M^2 / d\psi\right)_{amp}^{ret}$, (c) amplitude of acoustic pressure signal at the advancing-side microphone p'_{amp}^{adv} , (d) amplitude of acoustic pressure signal at the retreating-side microphone p'_{amp}^{ret} , (e) overall noise levels at the advancing-side microphone $OASPL^{adv}$ and (f) overall noise levels at the

Table 5.4: Statistics of leave-one-out Cross Validation of the derived PCE when tip-vortex experimental data are not available (Uniform distribution with $COV = 30\%$).

Objective	$N_{\text{Pearson}} [-]$	Gradient [deg]	NRMSE [%]	σ [%]
$(dC_n M^2 / d\psi)_{amp}^{adv}$	0.999	45.262	1.823	1.829
$(dC_n M^2 / d\psi)_{amp}^{ret}$	0.998	45.121	1.960	1.951
p_{amp}^{adv}	0.999	45.230	1.701	1.715
p_{amp}^{ret}	0.998	45.196	1.880	1.895
$OASPL^{adv}$	0.999	44.901	0.191	0.191
$OASPL^{ret}$	0.998	44.940	0.232	0.233

retreating-side microphone $OASPL^{ret}$. The microphone positions considered are the ones reported in Ref. [32] and used in Section 4.4.

The PCE method described in Section 3.3 is deployed for the efficient meta-modelling of the response of the aforementioned objectives for the obtained set of collocation points. Before commencing the UQ process, the predictive accuracy of the derived PCE is assessed through Leave-One-Out Cross-Validation (LOOCV). The LOOCV method is based on the construction of a dedicated PCE at each collocation point after excluding this specific point from the dataset. Then, the PCE-predicted value of the objective functions at this sample point is compared with the corresponding value obtained from the direct evaluation of this point using the actual simulation model. The aforementioned process is repeated for the entire set of sample points of the PCE procedure. The quality of the original PCE is assessed in terms of Pearson's product moment of correlation N_{Pearson} along with the gradient of the associated linear

Table 5.5: Uncertainty quantification when tip-vortex experimental data are available (Gaussian distribution with $COV = 10\%$).

Objective [unit]	μ [unit]	σ [unit]	γ_1 [-]	γ_2 [-]
$(dC_n M^2 / d\psi)_{amp}^{adv} [rad^{-1}]$	0.8208	0.1260	0.4242	0.1124
$(dC_n M^2 / d\psi)_{amp}^{ret} [rad^{-1}]$	0.6275	0.0747	0.6080	0.3646
$p'_{amp}^{adv} [Pa]$	84.1204	9.9117	0.3277	0.0462
$p'_{amp}^{ret} [Pa]$	46.6456	5.5634	0.3185	0.0247
$OASPL^{adv} [dB]$	114.4593	1.1787	0.0133	-0.1131
$OASPL^{ret} [dB]$	111.1533	1.1413	0.0276	-0.0649

regression line (Ref. [112]). A perfectly linear correlation would correspond to $N_{Pearson} = 1.00$ and a regression line gradient of 45.00 deg.

Fig. 5.2 a) and b) present the LOOCV applied to the PCE derived to approximate p'_{amp}^{adv} and $OASPL^{adv}$, respectively. The results refer to the case where experimental data are available, i.e. Gaussian input distributions with $COV = 10\%$. Very good agreement can be observed between the approximated and directly simulated values of p'_{amp}^{adv} and $OASPL^{adv}$. The p'_{amp}^{adv} data points depict that acoustic pressure is much more sensitive to vortex core radius uncertainties, however, $OASPL$, which is an integral noise metric is more immune to such disturbances. Table 5.3 summarises the statistical properties of LOOCV for all objectives. Linear correlation is observed for all objectives, with $N_{Pearson}$ above 0.995 and gradients very close to 45 deg. This showcases the ability of the developed PCE to replicate the physical behaviour of the direct simulation. Furthermore, the normalised root-mean-square error (NRMSE) of each objective is lower than 1% which quantified the predictive accuracy of the developed PCE. Finally, the corresponding standard deviation σ of the

calculated error is again in the same order for both variables. This indicates that the PCE error is uniformly distributed throughout the investigated design space. Table 5.4 presents the statistical properties of LOOCV referring to the case of not available experimental data with uniformly distributed uncertain inputs with $COV = 30\%$. Similar trends with the previous case are observed, albeit with slightly higher NRMSE below 2% for p'_{amp} and less than 0.25% for OASPL. This is attributed to the wider spread of the vortex-core related input variables. However, higher values of output uncertainty are anticipated in this case, which brings the relative contribution of LOOCV uncertainty at similar levels to the $COV = 10\%$ Gaussian case. Summarising, the quantification of quality and accuracy of the developed PCE confirms the applicability of the method for the context of uncertainty quantification in BVI noise prediction.

5.3 Uncertainty Quantification

Having derived computationally efficient PCE-based meta-models of verified accuracy through LOOCV, they can be utilised for UQ in BVI noise prediction subject to tip-vortex core radius uncertainty. The quantification of uncertainty is implemented through MC simulation based on results of the PCE meta-model.

Table 5.6: Uncertainty quantification when tip-vortex experimental data are not available (Uniform distribution with $COV = 30\%$).

Objective [unit]	μ [unit]	σ [unit]	γ_1 [-]	γ_2 [-]
$(dC_n M^2 / d\psi)_{amp}^{adv} [rad^{-1}]$	0.8795	0.2876	0.9466	-0.0379
$(dC_n M^2 / d\psi)_{amp}^{ret} [rad^{-1}]$	0.6723	0.1859	0.8416	-0.0081
$p'_{amp}^{adv} [Pa]$	86.9628	21.7400	0.7043	-0.2671
$p'_{amp}^{ret} [Pa]$	48.3714	12.4340	0.7829	-0.0744
$OASPL^{adv} [dB]$	114.6132	3.7109	0.0245	-0.7055
$OASPL^{ret} [dB]$	111.2138	3.3282	0.0612	-0.6188

Table 5.7: Sensitivity of output objectives to uncertain inputs when tip-vortex experimental data are available (Gaussian distribution with $COV = 10\%$).

Objective	Uncertain input Sensitivity index S_i [%]			
	r_{c0}^{wake}	a_1^{wake}	r_{c0}^{inflow}	a_1^{inflow}
$(dC_n M^2 / d\psi)_{amp}^{adv}$	0.0188	0.0098	27.6109	72.3605
$(dC_n M^2 / d\psi)_{amp}^{ret}$	0.0263	0.0298	35.5116	64.1954
p_{amp}^{adv}	0.0107	0.0152	27.3835	72.5907
p_{amp}^{ret}	0.0379	0.0041	46.9239	53.0341
$OASPL^{adv}$	0.0022	0.0139	27.7550	72.2289
$OASPL^{ret}$	0.0232	0.0038	47.1715	52.8016

Table 5.5 provides the results of UQ for the case when vortex-core experimental data are available. It is observed that the mean values of $(dC_n M^2 / d\psi)_{amp}^{adv}$ and $(dC_n M^2 / d\psi)_{amp}^{ret}$ are identical to the deterministic simulation results, whilst the corresponding uncertainties reach 15.35% and 11.90% of the associated mean values, respectively. As regards p_{amp}^{adv} and p_{amp}^{ret} , the mean values are slightly higher than the corresponding deterministic amplitudes (0.36% and 0.88%), with associated uncertainties of 11.78% and 11.93%, respectively. The uncertainty in $OASPL^{adv}$ and $OASPL^{ret}$ is 1.02% for both microphone locations, which represents a standard deviation of approximately 1.1 dB.

The skewness γ_1 of each objective distribution describes the degree of distortion from the symmetrical “bell-shaped” distribution. It can be seen that for the $COV = 10\%$ case with Gaussian input uncertainty distributions, the output

Table 5.8: Sensitivity of output objectives to uncertain inputs when tip-vortex experimental data are not available (Uniform distribution with $COV = 30\%$).

Objective	Uncertain input Sensitivity index S_i [%]			
	r_{c0}^{wake}	a_1^{wake}	r_{c0}^{inflow}	a_1^{inflow}
$(dC_n M^2 / d\psi)_{amp}^{adv}$	0.0402	0.0522	29.6523	70.2554
$(dC_n M^2 / d\psi)_{amp}^{ret}$	0.0253	0.1055	36.7268	62.9151
p_{amp}^{adv}	0.0208	0.0338	28.3730	71.5724
p_{amp}^{ret}	0.1324	0.0406	40.2352	59.5918
$OASPL^{adv}$	0.0045	0.0200	28.8606	71.1149
$OASPL^{ret}$	0.0385	0.0055	40.1943	59.7617

objectives are fairly symmetrical, with values of γ_1 generally below 0.5. Especially for $OASPL^{adv}$ and $OASPL^{ret}$, skewness is below 0.05 which corresponds to an almost fully-symmetrical distribution. The excess kurtosis γ_2 is a measure of outliers present in the distribution. It is noted that the kurtosis of Gaussian distribution is 3 by definition. Hence, a value of $\gamma_2 = 0$ would represent a perfectly Gaussian distribution. It is shown that for the $COV = 10\%$ case the non-deterministic output objectives follow distributions close to mesokurtic, with values of γ_2 generally below 0.5. Overall, it can be concluded that in cases when experimental data referring to the tip-vortex core radius are available, robust aeroacoustic predictions can be made, which confirms the suitability of free-wake/FW-H approaches for complex BVI noise prediction.

Table 5.6 presents the uncertainty analysis referring to the case described by uniformly distributed uncertain input variables with $COV = 30\%$. The uncertainty in p_{amp}^{adv} and p_{amp}^{ret} prediction is approximately 25% for both microphone

locations, whilst the associated mean values are 3.75% and 12.84% higher than the corresponding deterministic predictions. Standard deviations of 3.71 dB and 3.33 dB are calculated for the $OASPL^{adv}$ and $OASPL^{ret}$ objectives, respectively.

Furthermore, the distributions of $\left(dC_n M^2 / d\psi\right)_{amp}$ and p'_{amp} are moderately skewed to the right with γ_1 reaching positive values of almost 1. In this case, the mean value of output objectives is significantly higher than the mode of the distributions, which is the objective value with the highest probability of occurrence. Hence, by neglecting the input uncertainty in this case, the most probable output would be an under-estimated descriptor of acoustic pressure. The symmetry of OASPL is almost immune to even high values of input uncertainty. However, the associated excess kurtosis is considerably negative, with values lower than -0.5, which is primarily attributed to the uniform distribution of input uncertainty. It can be concluded that in the case of absent reliable experimental data for the tip-vortex core radius, considerable uncertainty is introduced to the output aeroacoustic predictions, which highlights the importance of non-deterministic simulation for robust BVI noise predictions.

It needs to be noted that some uncertainty related to PCE modelling is present in the uncertainty quantification carried out in this section. As discussed in Section 5.2, the NRMSE of the LOOCV for the Gaussian $COV=10\%$ and the uniform $COV=30\%$ case, is 0.65% and 1.7%, respectively, as regards the p'_{amp}^{adv} objective. The corresponding levels of normalised uncertainty σ/μ as calculated in this section are 11.8% and 25%. Therefore the relative contribution of PCE modelling uncertainty is more than one order of magnitude lower than the results of uncertainty quantification process. Additionally, the PCE modelling uncertainty is proportionally the same for the $COV=10\%$ and $COV=30\%$ cases.

5.4 Sensitivity Analysis

The relative contribution of individual sources of input uncertainty to the overall non-deterministic response of the system is quantified through global sensitivity analysis based on Sobol indices, as described in Section 3.3. Table 5.7

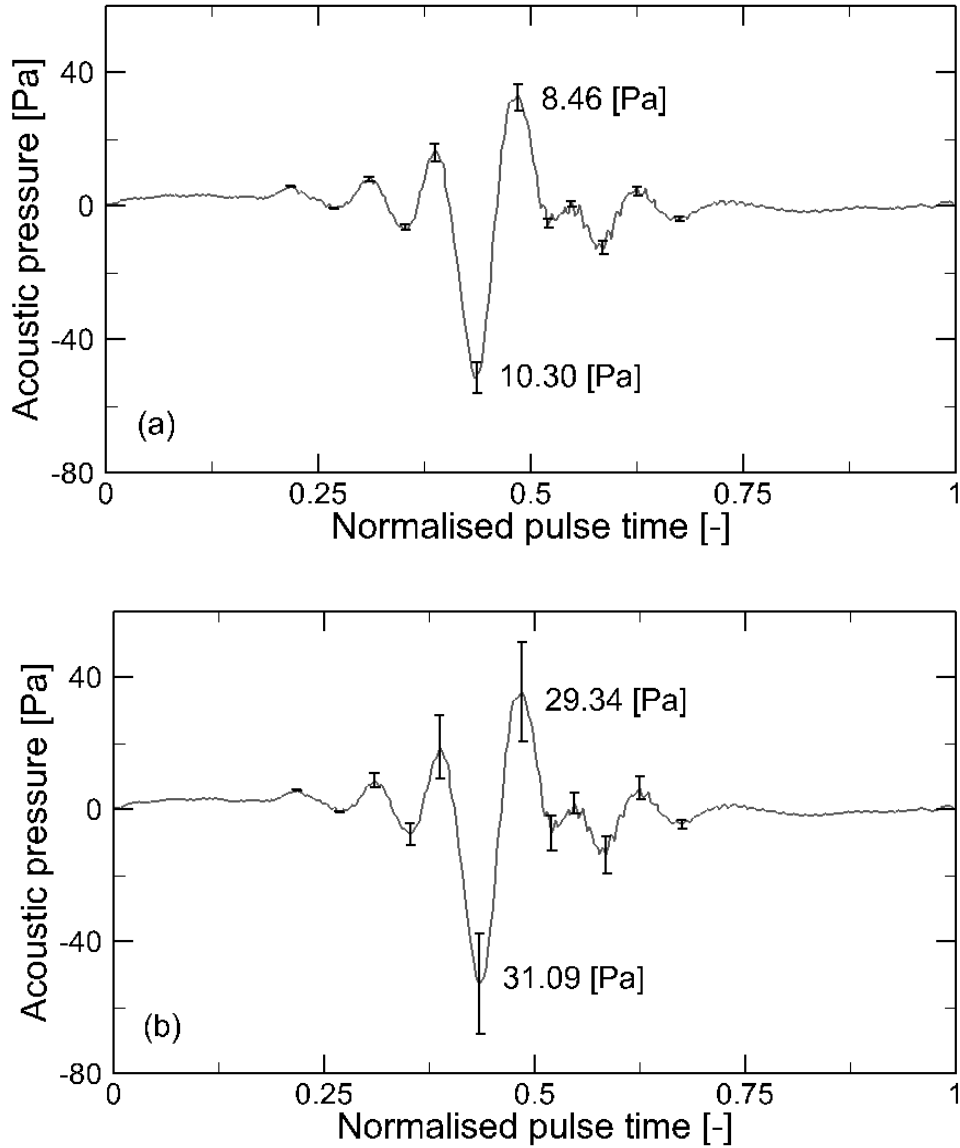


Figure 5.3: Non-deterministic prediction of acoustic pressure at the advancing side observer of the HART II rotor: mean acoustic pressure and uncertainty bars; (a) Gaussian input with $COV = 10\%$, (b) uniform input with $COV = 30\%$.

presents the global sensitivity indices S_i referring to the case of available tip-vortex experimental data with Gaussian $COV = 10\%$. It is observed that uncertainty in the parameters used for wake geometry calculations, r_{c0}^{wake} and a_1^{wake} , has insignificant influence on aeroacoustic outputs. However, inflow-

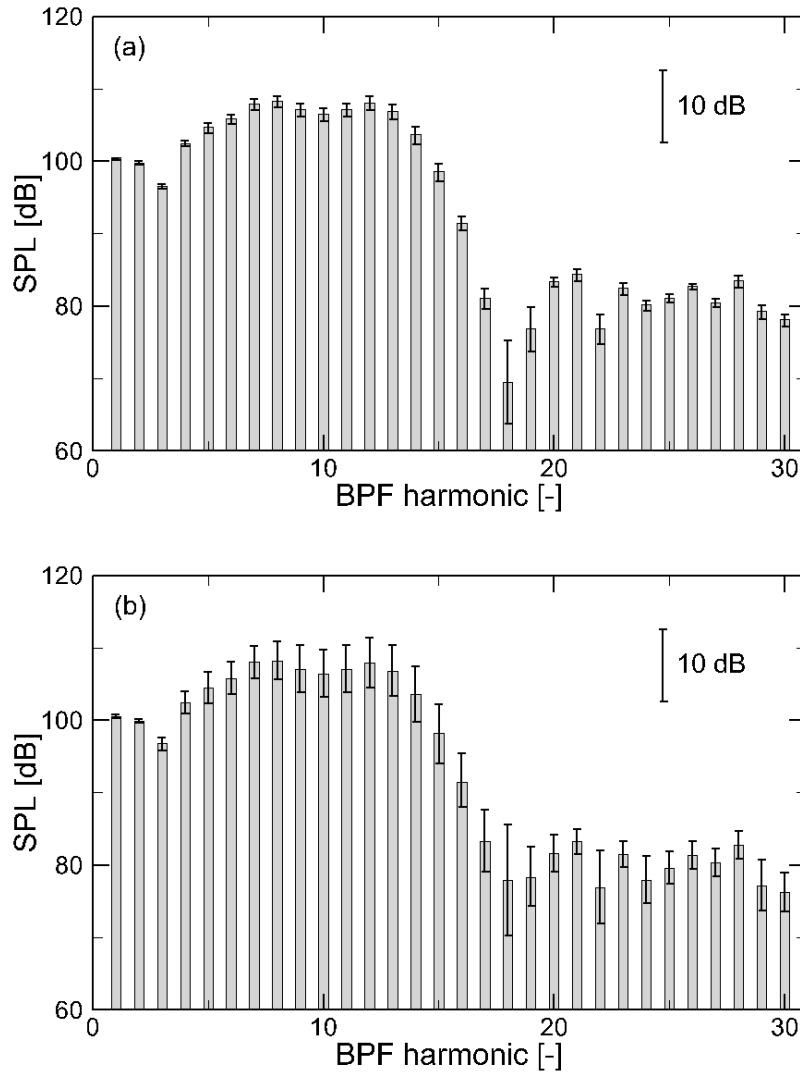


Figure 5.4: Non-deterministic predictions of spectral content at the advancing side observer of the HART II experiment: mean SPL and uncertainty bars; (a) Gaussian input with $COV = 10\%$, (b) uniform input with $COV = 30\%$.

related input uncertainty and especially uncertainty in the value of Squire parameter a_1^{inflow} has approximately 72% contribution to the overall system's response. Similar trends are observed in the sensitivity analysis of the $COV = 30\%$ case with uniform input distributions, as shown in Table 5.8. It is therefore illustrated that the relative contribution of input parameters is not dependent upon the shape and value of input uncertainty.

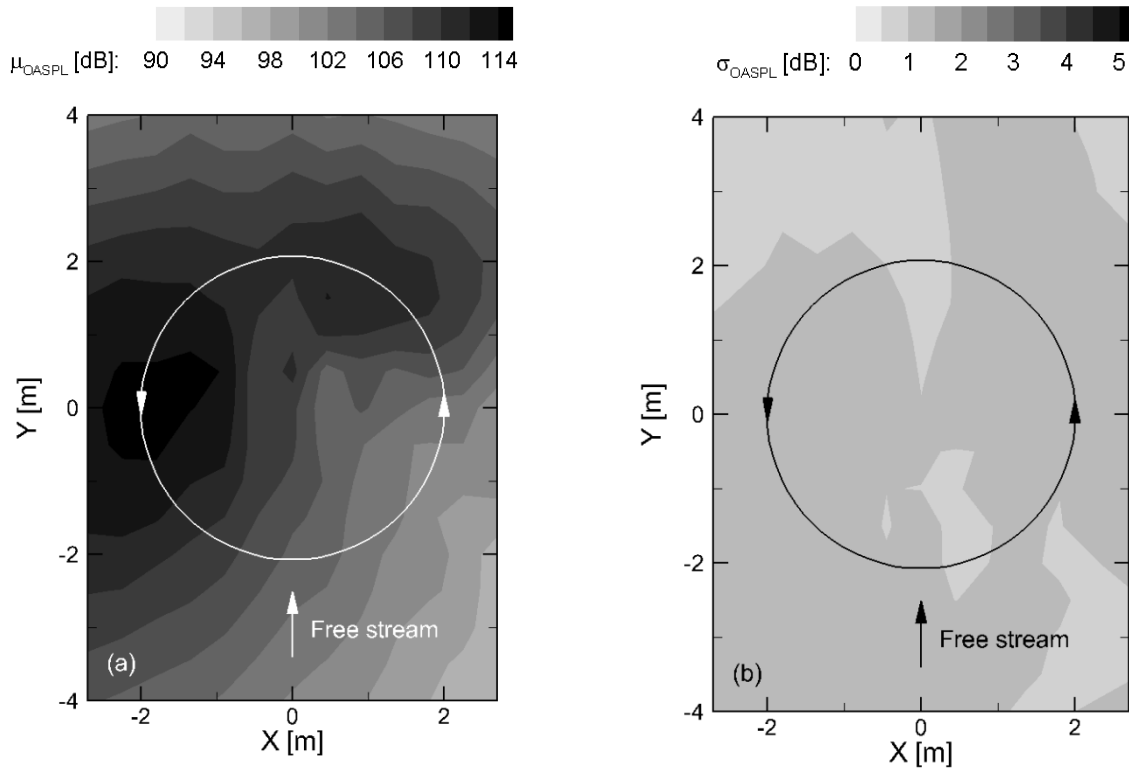


Figure 5.5: Non-deterministic predictions of OASPL below the HART II rotor for Gaussian input with $COV = 10\%$: (a) mean OASPL, (b) standard deviation.

Additionally, in both cases it is observed that the sensitivity indices of a_1^{inflow} are consistently higher at the advancing side, compared with the corresponding indices of retreating side. This is attributed to the tip-vortex behaviour at both sides of the rotor. As shown in Refs. [32], [111], retreating rotor blades interact with trailed tip-vortices of age less than 90 deg. At the early stages of tip-vortex propagation, vortex core growth is minimal, which in combination with a small r_{c0} can lead to small vortex core radii which directly affect induced velocity and airloads. A larger r_{c0} would act as a “damper” of excessively high vortex core radii, which depicts the importance of initial vortex core radius in UQ of retreating-side BVI noise. As regards advancing-side BVIs, most of interactions comprise relatively grown tip-vortex filaments of age greater than 90 deg.,

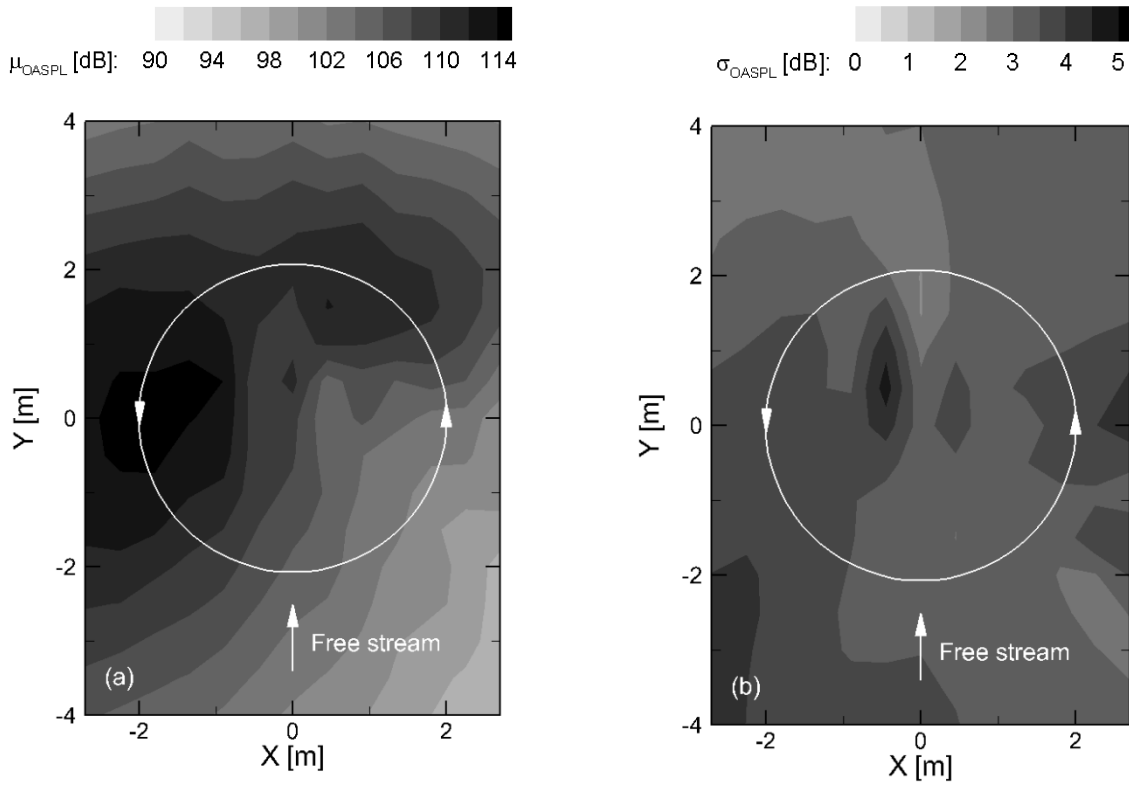


Figure 5.6: Non-deterministic predictions of OASPL below the HART II rotor for uniform input with $COV = 30\%$: (a) mean OASPL, (b) standard deviation.

hence the impact of r_{c0}^{inflow} becomes less important at the advancing side BVI uncertainty.

5.5 Non-Deterministic Aeroacoustic Predictions

Fig. 5.3 a) presents the non-deterministic prediction of full time-history of a single BVI acoustic pressure pulse for the case of Gaussian $COV = 10\%$ input distribution of the tip-vortex core parameters. The pressure time-histories of all sample collocation points have been aligned together to remove any phase shift between the 60 acoustic signals. The illustrated acoustic pressure distribution represents the mean p'^{adv} value at each point of the time-history. The distribution of non-deterministic mean acoustic pressure is almost identical to the deterministic predictions illustrated in Fig. 4.5. Superimposed are the uncertainty bars representing two standard deviations of p'^{adv} at each time-

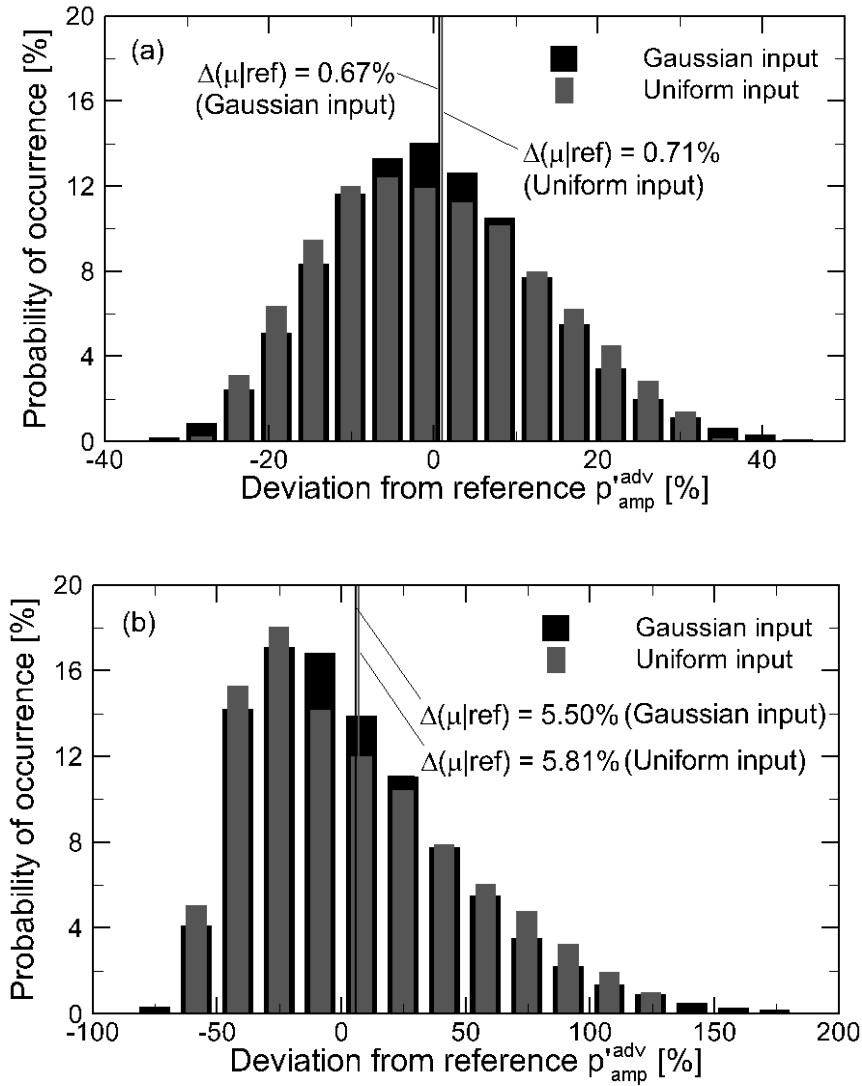


Figure 5.7: Comparison of probability histograms representing the amplitude of acoustic pressure pulse for Gaussian and uniform input uncertainty; (a) $COV = 10\%$, (b) $COV = 30\%$.

point. It noted that only the significant uncertainties are visualised in this illustration. It can be seen that a standard deviation of 4.23 Pa corresponds to the positive peak of BVI pulse, whilst 5.15 Pa uncertainty is recorded at the negative peak. The resulting non-deterministic amplitude of acoustic pressure is very close to the $p_{amp}^{adv} = 9.9117 \text{ Pa}$ calculated in Section 5.3, which confirms the suitability of the employed descriptor as objective of the UQ process.

Fig. 5.3 b) presents the non-deterministic prediction of full time-history of a single BVI acoustic pressure pulse for the case of uniform input distributions with $COV=30\%$. The mean acoustic pressure distribution is similar to the deterministic, although slightly over-predicted, with peak local differences reaching 2.1%. The width of uncertainty bars follows same trends as in the Gaussian $COV=10\%$ case, however, the maximum uncertainties reach 14.67 Pa and 15.55 Pa at the positive and negative peaks of the BVI pulse, respectively. The distribution of uncertainty in the non-deterministic prediction of acoustic pressure is essentially reflected on the spectral analysis in the frequency range where BVI noise is dominant. Fig. 5.4 a) and Fig. 5.4 b) illustrate the mean FFT with superimposed uncertainty bars corresponding to the $COV=10\%$ and $COV=30\%$ cases, respectively. The non-deterministic mean values are identical across the entire frequency range, which confirms the immunity of mean SPL to vortex core input uncertainties. For the case of Gaussian input with $COV=10\%$, relatively small levels of uncertainty are observed across the entire frequency range, with standard deviations in the order of 1 dB. This highlights the robustness of free-wake/FW-H approaches in predicting BVI noise when experimental data referring to tip-vortex core growth are available. However, in the case of absent data, uncertainties in the order of 3.5 dB govern the mid-frequency content of BVI signal, primarily attributed to the high standard deviations observed around the peaks of acoustic pressure pulse. It is noted that the low-frequency harmonic noise is less affected by vortex-core-related uncertainty, as standard deviations below 1 dB are associated with the first three BPF harmonics.

The impact of uncertainty on the spectral content of BVI noise on OASPL predictions is illustrated in Fig. 5.6 a), which presents the 6-40 BPF mean ground noise distribution for the BL conditions of the HART II experiment (Ref. [32]). As expected, the mean OASPL distribution is very similar to the deterministic, as presented in Fig. 4.8. The characteristic advancing- and retreating-side BVI lobes are identical, whilst slight deviations can be seen in the region between the two lobes. Consequently, this is the region of highest

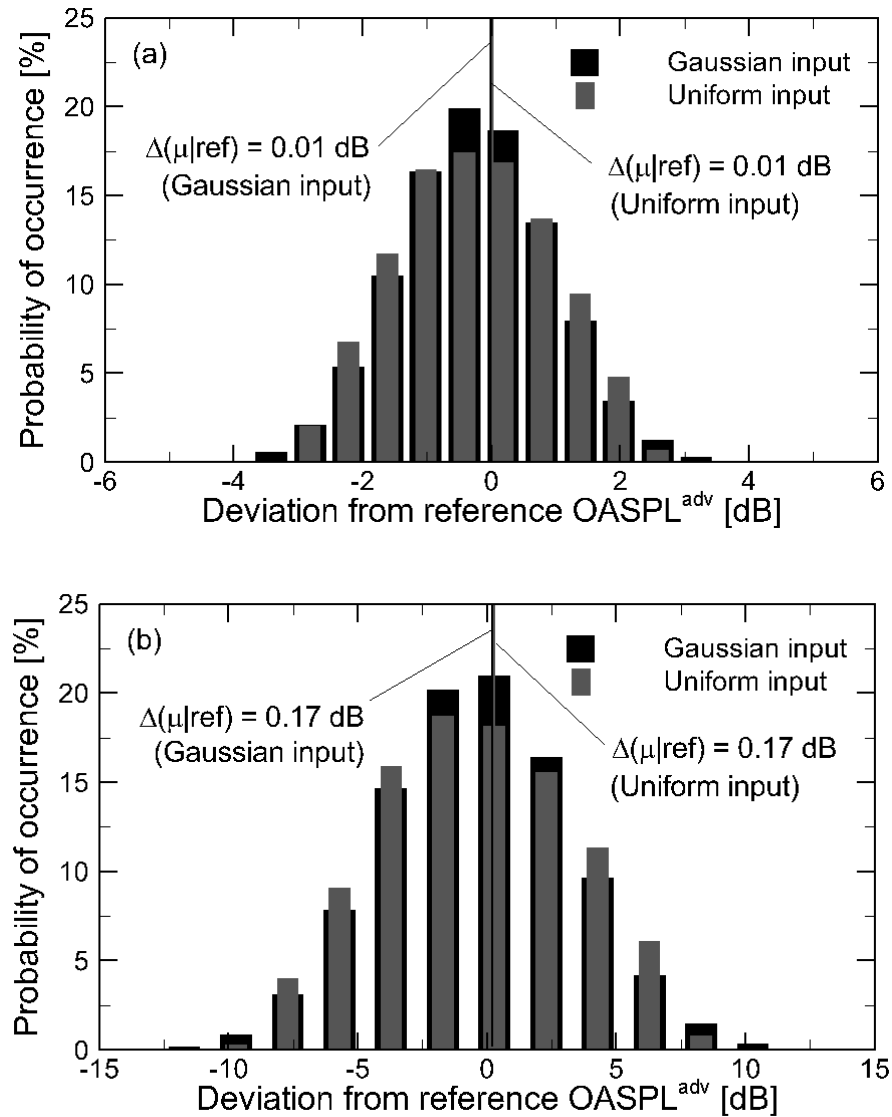


Figure 5.8: Comparison of probability histograms representing OASPL for Gaussian and uniform input uncertainty; (a) $COV = 10\%$, (b) $COV = 30\%$.

uncertainty, with standard deviation reaching values up to 1.4 dB, as shown in Fig. 5.5 b).

Generally, the prediction of ground noise contour is adequately robust in the $COV = 10\%$ case, with values of standard deviations not exceeding 1 dB at most measurement nodes. Identical mean OASPL distribution is obtained in the uniform $COV = 30\%$ case as shown in Fig. 5.6 a). However, the corresponding uncertainty can reach up to 4.6 dB, dictating the importance of non-deterministic

simulation when experimental evidence referring to vortex core growth is not available.

5.6 Fundamental Modelling Guidelines

The previous analysis has shown that the impact of input uncertainty on the robustness of aeroacoustic predictions is dependent upon the shape and COV of input distributions. Fig. 5.7 a) presents the probability histograms of p_{amp}^{adv} , comparing the cases of Gaussian and uniform uncertain input distributions, both

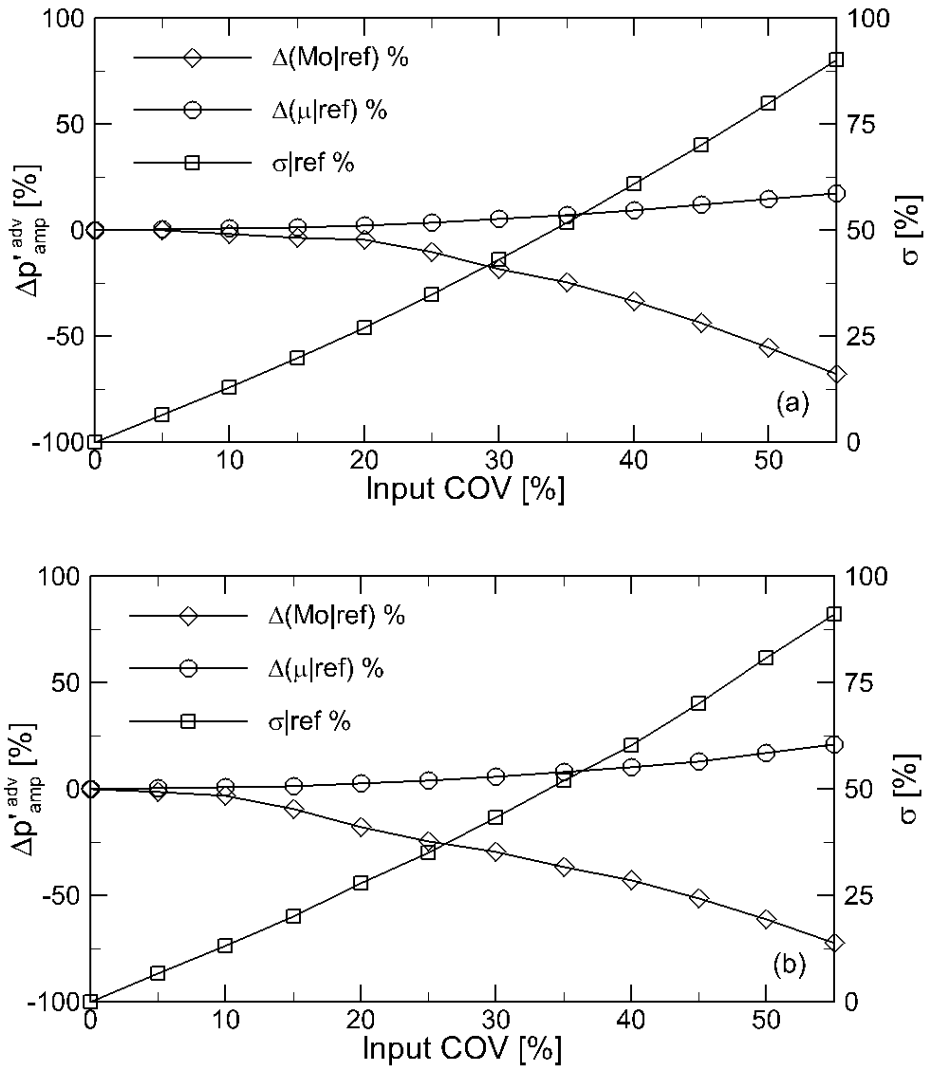


Figure 5.9: Impact of input uncertainty on the prediction of acoustic pressure; (a) Gaussian input, (b) Uniform input.

with $COV = 10\%$. The distributions are expressed as relative deviation from the p_{amp}^{adv} which achieves the best match with experimental data. This value will be used as reference for the entire analysis presented in this section. It is shown that both histograms are fairly symmetrical, with the one corresponding to uniformly distributed inputs being slightly platykurtic (kurtosis lower than 3). Fig. 5.7 b) illustrates the corresponding histograms for $COV = 30\%$. It can be seen that both distributions are positively skewed with modes Mo around -25% of the reference value. It is noted that the mode of a distribution is the value with the highest probability if the uncertain inputs are picked randomly. The above finding depicts the importance of non-deterministic simulation at high values of input uncertainty: a considerable under-prediction of p_{amp}^{adv} will most probably occur if the case is approached deterministically. On the other hand, non-deterministic simulation would result in an over-prediction of only 5.8% relative to the reference p_{amp}^{adv} . $OASPL^{adv}$ is less sensitive to input variations, retaining almost symmetrical histograms in both Gaussian and uniform input cases, as shown in Fig. 5.8 a) and Fig. 5.8 b).

The uncertainty quantification method described in Section 3.3 is repeated for a wide range of input COV and for two different shapes of input distributions, Gaussian and uniform. Fig. 5.9 a) and b) illustrate the uncertainty map referring to p_{amp}^{adv} for input COV ranging from 0% to 55% and for Gaussian and uniform input distributions, respectively. Each point of the map essentially represents the outputs of full UQ based on dedicated PCEs comprising 60 collocation points, each, resulting in a total of 1320 deterministic simulations.

The output predictions are provided as percentage changes relative to the reference of p_{amp}^{adv} , whilst output uncertainty is presented as percentage of the same reference value. It can be seen that output uncertainty is almost linearly correlated with input uncertainty, with a gradient of approximately 58 deg. for both distributions. As regards the relative deviation of mean p_{amp}^{adv} , it is described by a 2nd order polynomial trend, with values generally higher than the

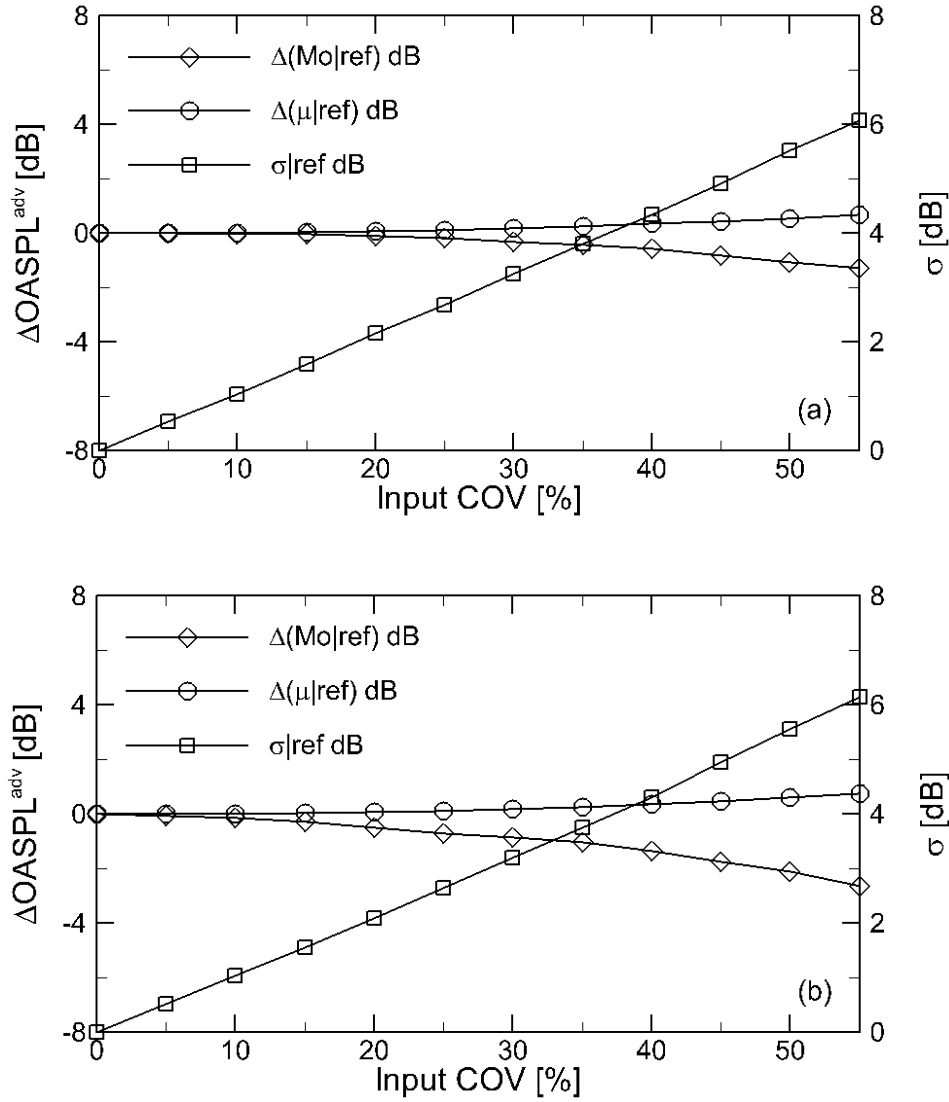


Figure 5.10: Impact of input uncertainty on the prediction of OASPL; (a) Gaussian input, (b) Uniform input.

reference. The relative deviation of the Mo of p_{amp}^{adv} presents a diverging trend, leading to significant under-predictions at COV greater than 15%. This is the point where input uncertainty starts to become important for the accuracy of acoustic pressure predictions. The difference between the curves corresponding to μ and Mo of the acoustic pressure amplitude predictions essentially quantifies the benefits that arise from non-deterministic simulation relative to the most probable outcome of the deterministic execution.

Fig. 5.10 a) and b) present the same analysis for the $OASPL^{adv}$ output objective. The non-deterministic mean-value discrepancies from the reference $OASPL^{adv}$, as well as standard deviation are provided in dB. Generally low deviations of the non-deterministic mean from the reference are observed, with values not exceeding 1 dB even at high levels of input uncertainty. The associated output uncertainty is again linearly correlated with input COV, with a gradient of 6.4 deg. The relative deviation of the Mo of $OASPL^{adv}$ presents again a diverging trend, with under-predictions of approximately 2 dB at excessively high values of input uncertainty.

Summarising, for given requirements in terms of accuracy and robustness, the analyst can resort to the developed uncertainty maps in order to: (a) estimate the anticipated error of deterministic simulation, (b) evaluate the necessity of non-deterministic simulation and assess the expected improvement in accuracy, and (c) quantify the associated uncertainty depending on output objectives and analysis requirements. The above guidelines are applicable for both detailed acoustic pressure prediction and holistic ground noise evaluation associated with helicopter rotors in BVI-dominated flight conditions.

5.7 Discussion

In this chapter a holistic uncertainty analysis has been undertaken, examining the impact of tip-vortex core modelling uncertainty on the aeroacoustic simulation of helicopter rotors in BVI conditions. Two distinct cases have been defined: (a) when experimental data about the geometry of tip-vortex core are available and (b) when experimental data are absent. For both cases, uncertainty quantification has been conducted, based on Polynomial Chaos expansion. The quality of approximation has been assessed through Leave-One-Out Cross-Validation, depicting the predictive accuracy of the Polynomial expansion and its suitability for uncertainty quantification of highly complex BVI noise cases. The output objectives for statistical analysis were metrics representative of unsteady rotor airloads, acoustic pressure history and OASPL noise.

A sensitivity analysis based on Sobol indices has been carried out to identify the relative contribution of each uncertain parameter to the overall response of the system. It was found that the inflow-related parameters, and especially the Squire parameter, have the greatest influence on BVI airloads and noise uncertainty.

It has been shown that in the case of availability of experimental data, the levels of uncertainty in acoustic pressure prediction are 11.78% of the predicted mean value, whilst the associated numerical error for the same evaluation is 0.84%. Similarly, the OASPL uncertainty is in the order of 1 dB, whilst numerical errors reach up to 0.33dB. For both objectives, numerical errors are one order of magnitude lower than the uncertainty in modelling which demonstrates the suitability of free-wake/FW-H approaches for modelling helicopter BVI noise.

When vortex-core-related experimental data are absent, output uncertainty in the order of 3.5 dB can occur for conservative selection of input uncertainty. This highlights the necessity of non-deterministic simulation when correlations with tip-vortex experimental data are not feasible. Furthermore, it has been shown that in the absence of experimental evidence, deterministic approaches could lead to acoustic pressure under-predictions in the order of -25%. For the same case, employment of non-deterministic simulation improves the accuracy of simulation to values corresponding to only 5.8% over-predictions, relative to the validated reference acoustic pressure.

A generalised set of aeroacoustic predictions, expressed with reference to the validated solution which achieves best match with experimental data has been provided. The simulations have been carried out at different levels and distributions of input uncertainty, essentially providing an uncertainty map, which can serve as modelling guidelines for analysts. Specifically, for given requirements in terms of predictive robustness, the analyst can estimate the anticipated error of deterministic simulation, evaluate the necessity and benefit of non-deterministic simulation and quantify the associated levels of uncertainty depending on output objectives and analysis requirements.

6 Results and Discussion; Helicopter Noise Prediction at Mission Level

6.1 Introduction

A mathematical formulation for the prediction of helicopter rotor noise has been presented in Section 3.2. The proposed method was extensively validated in Chapter 4, as regards unsteady airloads, acoustic pressure and ground noise impact. The robustness of the developed model when tip-vortex-related experimental data are available has been assessed in Chapter 5. A validated and reliable strategy for helicopter rotor noise prediction enables the transition to mission-level noise prediction, applying the integrated method proposed in Section 3.4.

An adaptive grid sensitivity analysis is carried out to determine the width of high-resolution region in both Cartesian directions of the moving observer planar grid. The developed adaptive grid is coupled with the on-the-fly source noise prediction method introduced in Sections 3.2 and 3.4. This alleviates the need for pre-stored acoustic datasets and improves noise predictive accuracy, since source noise is calculated at each trajectory waypoint resolving the operational characteristics of each unique flight state. In this way, the transition to complete mission simulation is achieved in a computationally efficient and physics based manner.

The applicability of the integrated method on “real-world” rotorcraft operations is demonstrated for two generic, three-dimensional missions, without the need for pre-stored noise data. Detailed investigation of individual mission waypoints or isolated acoustic observers is provided through the developed method. Additionally, cumulative metrics are proposed for the evaluation of the complete mission acoustic impact. The proposed approach provides insight into

helicopter noise prediction at mission level, elaborating on the coupling of aeroelastic rotor response with rotorcraft flight dynamics and aeroacoustics.

6.2 Adaptive Grid Sensitivity Analysis

The adaptive grid implemented in this study includes a high-resolution region, fixed to the helicopter position and continuously updated during the mission to capture the complex noise patterns at the predominantly affected ground regions. A lower resolution is selected for the rest of the mission area, as illustrated in Fig. 3.5, to reduce the required computational resources. The four operating parameters that predominantly affect the acoustic footprint patterns are: the flight altitude h , the rotor advance ratio μ , the flightpath angle γ and the rotor thrust coefficient C_T , as these are directly related to the radiation distance, source velocity, source noise directivity and blade loading, respectively.

A sensitivity analysis is carried out to determine the width as well as the number of observers in the high-resolution grid region. The helicopter employed for this

Table 6.1: Bo 105 main rotor geometrical characteristics.

Parameter	Value	Unit
Blade count	4	blades
Rotor radius	4.91	m
Blade chord	0.27	m
Rotor solidity	0.07	-
Rotational speed	44.4	rad/s
Non-dimensional root cut-out distance	0.21	-
Blade linear twist per span	-8	deg
Precone angle	2.5	deg
Aerofoil	NACA23012	-

study is the Airbus Helicopters Bo 105 helicopter. It is a twin-engine light (TEL) multipurpose helicopter, featuring a 4-bladed hingeless rotor. An extensive validation of performance and flight dynamics of this model is provided in Ref. [66]. Table 6.1 summarises the main design characteristics of the main rotor of the selected helicopter. The acoustic footprints are assessed for a wide range of the four aforementioned governing parameters, as described in Table 6.2. The ground Cartesian coordinates, as well as flight altitude are normalized with the rotor radius R .

Five different values (with corresponding IDs) have been determined for each governing parameter, which leads to 20 total cases examined. At each ID, only the values of the governing parameter of interest are varied keeping the other three parameters at their default values annotated in brackets in Table 6.2. For example, when examining the impact of advance ratio μ the ID=2 corresponds to $\mu = 0.08$, with the rest parameters at default $h/R = 100$, $\gamma = 0$ deg. and $C_T = 0.0044$. Subsequently, the X and Y Cartesian gradients of the OASPL recorded are calculated to determine the sensitivity of noise patterns in the

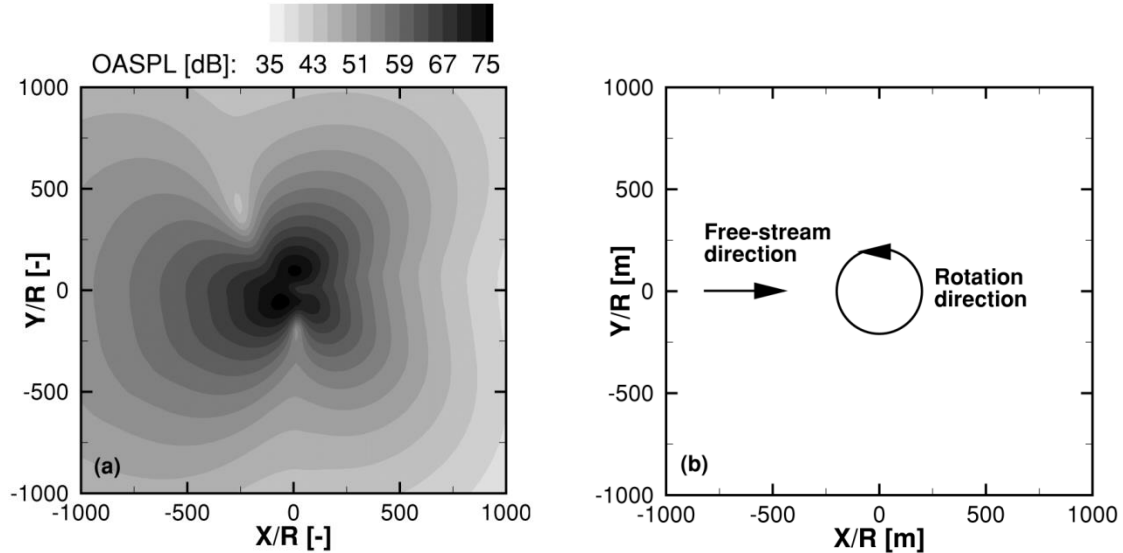


Figure 6.1: Adaptive observer grid sensitivity analysis in level-flight for $h/R = 100$, $\mu = 0.16$ and $C_T = 0.0044$; (a) Ground noise footprint, (b) Free-stream and rotation directions.

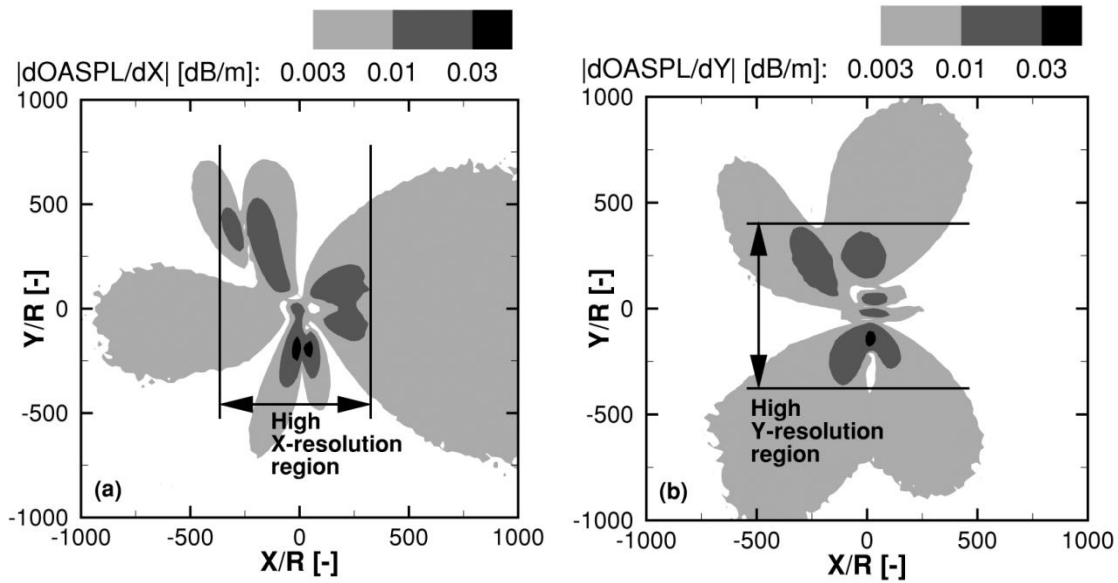


Figure 6.2: Adaptive observer grid sensitivity analysis in level-flight for $h/R=100$, $\mu=0.16$ and $C_T=0.0044$; (a) OASPL absolute X-gradient, (b) OASPL absolute Y-gradient.

aforementioned directions. A fourth-order central finite difference scheme has been employed for the estimation of OASPL gradients.

Fig. 6.1 a) illustrates the ground footprint for the Airbus Helicopters Bo 105 helicopter, cruising at normalized altitude $h/R=100$, with advance ratio $\mu=0.16$ and thrust coefficient $C_T=0.0044$. The helicopter is located at $X/R=0$ m, $Y/R=0$ m and the free-stream direction is towards the positive side of X-axis,

Table 6.2: Cases examined in the observer grid sensitivity analysis.

Parameter	Examined values (IDs 1 to 5)	Unit
Normalised altitude h/R	5, 50, (100), 250, 400	-
Advance ratio μ	0.00, 0.08, (0.16), 0.24, 0.32	-
Flight path angle γ	-7.00, -3.00, (0.00), 3.00, 7.00	deg
Thrust coefficient C_T	0.0030, 0.0038, (0.0044), 0.0050, 0.0058	-

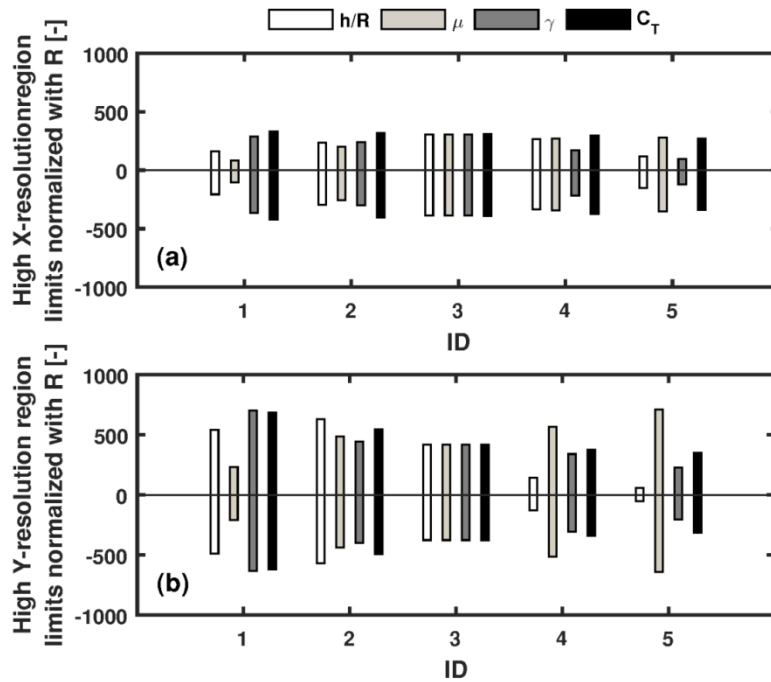


Figure 6.3: Summary of the limits of the high-resolution regions, exported from the observer grid sensitivity analysis; (a) X-direction, (b) Y-direction.

as shown in Fig. 6.1 b). The noise metric adopted in this study is the Overall Sound Pressure Level (OASPL), which is calculated by integrating over the resolved frequencies of the acoustic signal (Ref. [155]). Thickness noise has in-plane directivity so ground noise is generally higher upstream of the helicopter. Loading noise has mainly out-of-plane directivity, which in combination with the higher Mach number values of the advancing side, results in higher loading noise below the right side of the helicopter.

Fig. 6.2 a) and b) depict the absolute value of the X and Y gradients utilizing a three-level logarithmic scale, from 0.003 to 0.03 dB/m. Gradients greater than 0.01 dB/m are observed in ground region extending from $X/R=-325$ to $X/R=286$ in the X-direction, and from $Y/R=-320$ to $Y/R=411$ in the Y-direction. Limited regions with gradients greater than 0.03 dB/m are located inside the aforementioned area. The rest of the mission area is described by OASPL gradients lower than 0.003 dB/m. Similar patterns are observed for the rest of

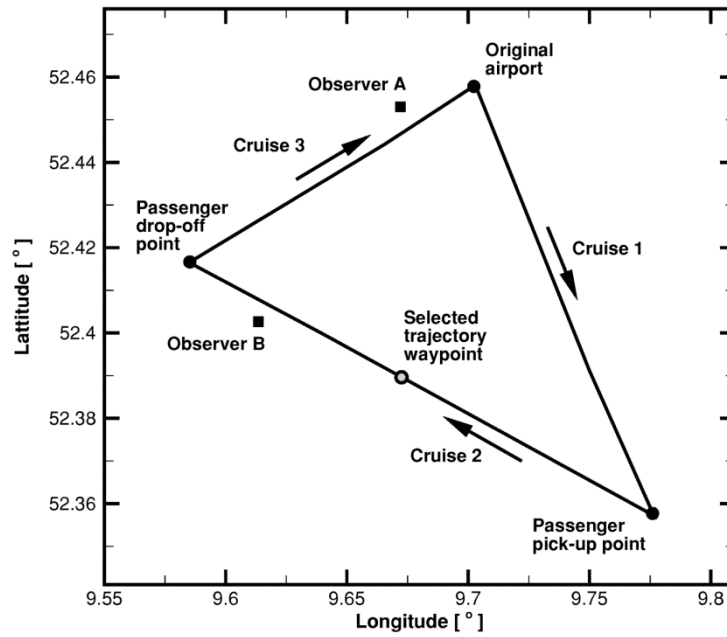


Figure 6.4: PAT scenario geographical definition in WGS 84.

the examined cases, albeit with variations in the location and width of the high-gradient regions.

Fig. 6.3 summarises the results of observer grid sensitivity analysis providing the extent of the high-gradient regions in both X and Y positive/negative directions. Each of the governing operational parameters previously defined is represented by one coloured bar. The IDs 1 to 5 refer to the five different values examined for each governing parameter. At each ID, only the values of the governing parameter of interest are varying; the other three governing parameters are assigned their default value indicated in Table 6.2. It can be concluded that the $X/R=-500$ to $X/R=500$ and the $Y/R=-500$ to $Y/R=500$ ranges adequately determine the limits of the high-resolution grid region across the entire envelope of operating conditions. The observer spacing in both Cartesian directions inside the high-resolution area will be $dX=dY=30$ m, which will lead to a numerical accuracy of approximately 1 dB based on the maximum value recorded for the OASPL gradient. This accuracy is comparable to the 2 dB scatter observed in state-of-the-art experimental noise measurement, as reported in Ref. [32]. A spacing of $dX=dY=180$ m will be adopted outside of the

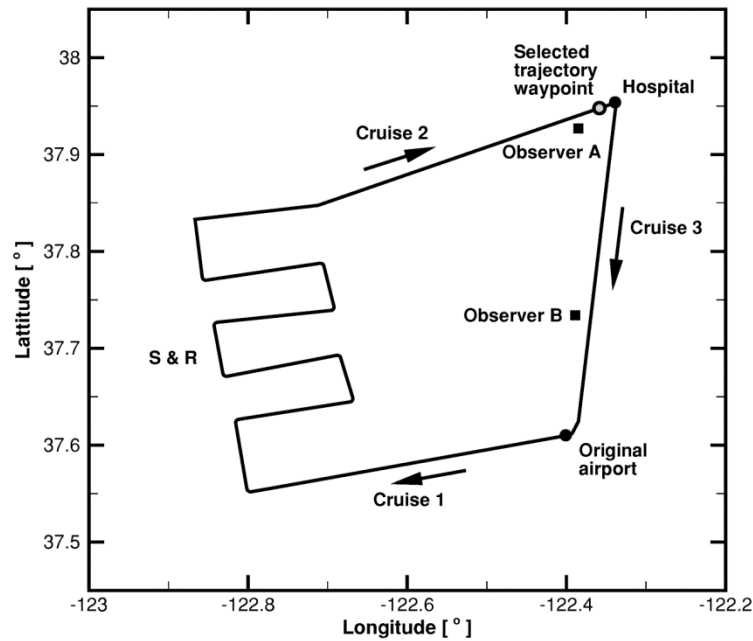


Figure 6.5: SAR scenario geographical definition in WGS 84.

high-resolution area where OASPL gradients are significantly lower. As discussed in Section 3.4, the high-resolution region of the ground grid dynamically adjusts with mission time by “following” the helicopter across its trajectory.

6.3 Mission Scenarios

An integrated methodology has been established for the prediction of source and ground noise, which now allows the application on realistic rotorcraft operations. Two generic, three-dimensional missions representative of modern TEL helicopter operations have been defined: a passenger air-transport mission (PAT) and a search and rescue (SAR) mission. The incorporated operational procedures in terms of geographical location selection, deployed airspeed, altitude, climb/descent rates, and idle times have been defined in collaboration with the European Helicopter Operators Committee (EHOC). It is noted that the effect of change in air density and speed of sound is inherently accounted in the variation of flight altitude. The geographical representation in terms of global coordinates, are illustrated in Fig. 6.4 and Fig. 6.5 for the PAT and SAR case,

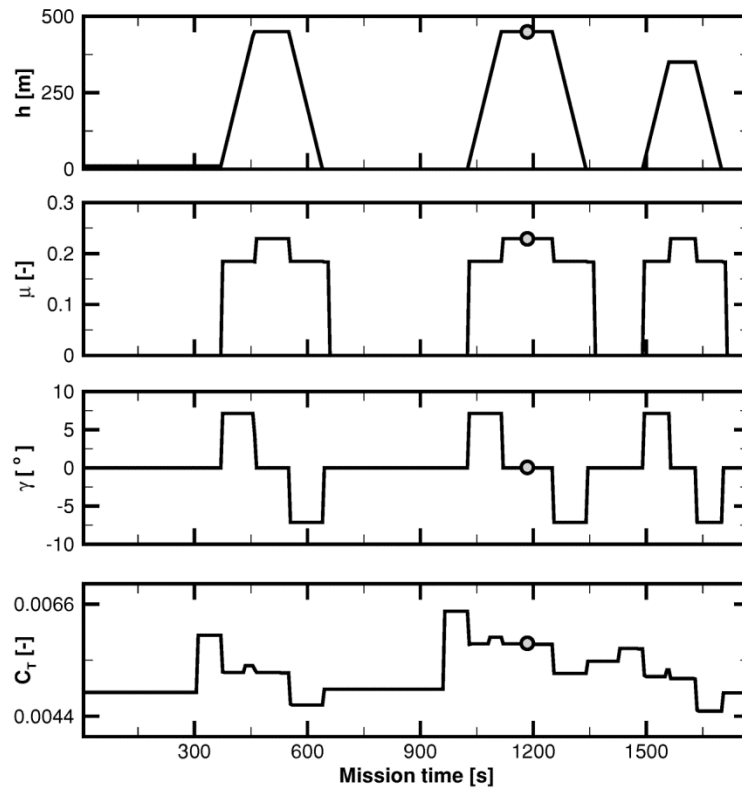


Figure 6.6: PAT scenario time variations of deployed operational altitude advance ratio, flightpath angle and thrust coefficient.

respectively. Individual trajectory waypoints are selected for both scenarios (grey-filled circle markers) to investigate the radiation-sphere and instantaneous ground footprints. Two isolated observers (A and B) are defined for each mission to assess the perceived noise time-variations at representative ground locations.

The PAT mission assumes that the rotorcraft takes-off from Hanover Airport (Germany) and cruises in order to pick up passengers from a designated location. It subsequently cruises towards the Garbsen Hotel to drop them off and then returns back to Hanover Airport. The SAR mission assumes that the helicopter takes off from San Francisco Airport (USA) and then travels towards the Gulf of the Farallones. The helicopter is then assumed to engage in a search pattern to trace and pick up citizens in distress from an assumed naval incident location. After successfully locating the citizens in distress, the

helicopter SAR crew picks them up while hovering over the imaginary incident zone. The citizens are then transported to the San Pablo Hospital, where the helicopter drops them off and subsequently transits back to San Francisco Airport. Climb and descent rates are held fixed at 5 m/s. A time step of 5 s is used for each individual mission segment. During idle operation, the overall helicopter power requirements are assumed to be equal to 10% of maximum contingency engine shaft power. The acoustic impact of the helicopter in idle is neglected.

The time variations of the helicopter operating parameters are represented in Fig. 6.6 and Fig. 6.7, for the PAT and SAR case, respectively. The PAT mission comprises three legs in total and covers a total area of 323 km². The cruise speed within the first two legs is 50 m/s at an altitude of $h = 450$ m, while for the final leg, the cruise speed is 45 m/s at an altitude of $h = 350$ m. The total PAT mission duration is 0.49 hours, extending to a total range of 36.22 kilometres

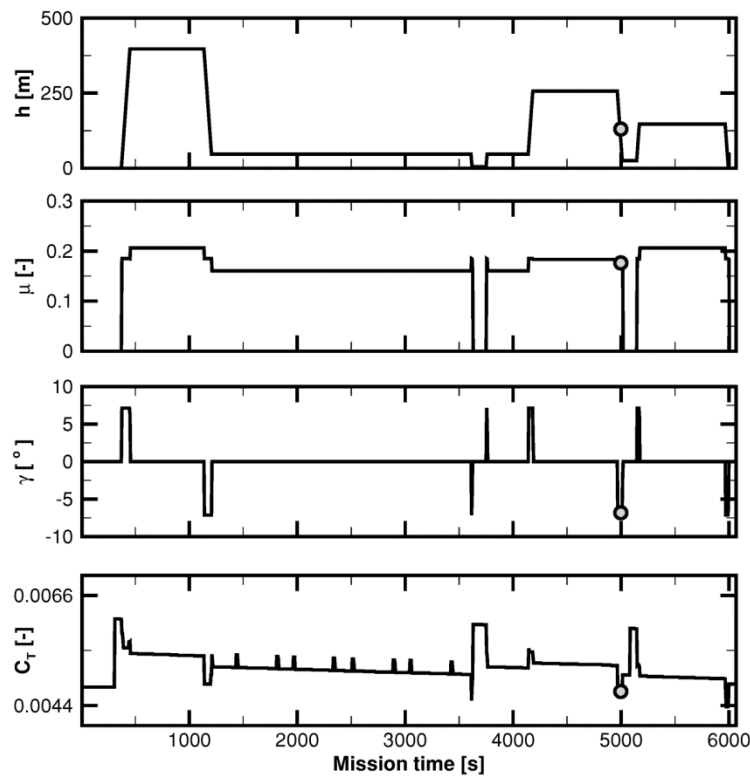


Figure 6.7: SAR scenario time variations of deployed operational altitude, advance ratio, flightpath angle and thrust coefficient.

and consuming 55.19 kg of fuel. As regards the SAR mission, a total area of 3969 km² is examined. The helicopter spends almost half of the mission time engaged in the SAR pattern, where flight speed and altitude are 35 m/s and 50 m, respectively. The total SAR mission duration is 1.68 hours, extending to a total range of 208.45 kilometres and consuming 191.41 kg of fuel.

6.4 Impact of Fuel Burn on Source Noise Generation

The amount of fuel burned during the SAR mission equals approximately to 8.7% of the take-off mass of the helicopter. The AUM reduction due to fuel burn has direct impact on the C_T time history, where a 9.26% reduction is observed

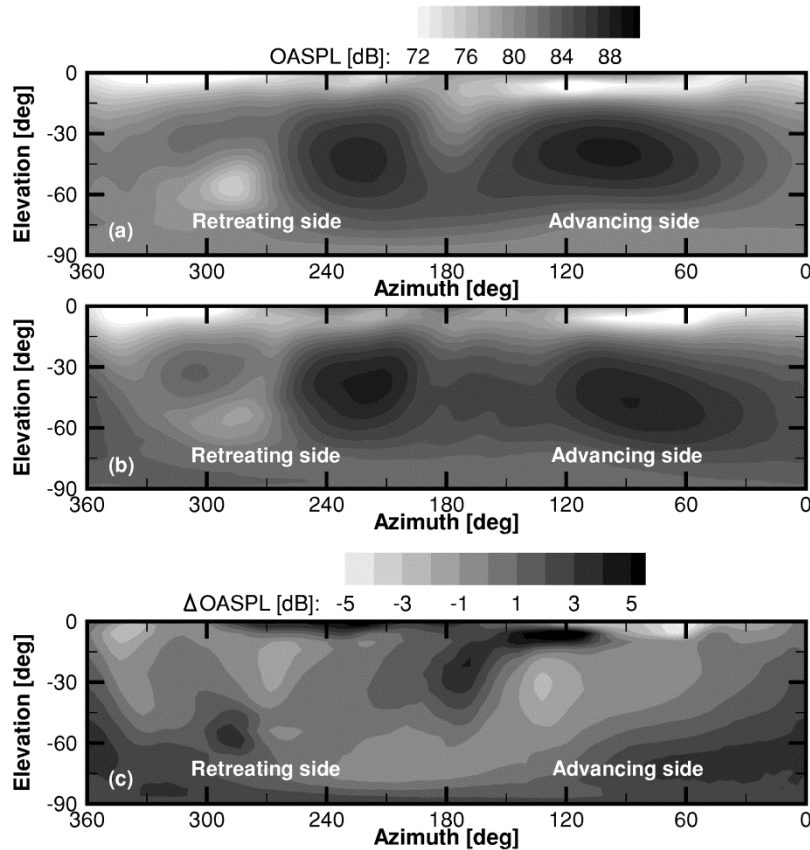


Figure 6.8: Loading noise acoustic spheres for the SAR scenario; (a) beginning of cruise 1: $t = 455$ s, $\mu = 0.206$, $\gamma = 0$ deg., $C_T = 0.0054$, (b) ending of the cruise 3: $t = 5965$ s, $\mu = 0.206$, $\gamma = 0^\circ$, $C_T = 0.0049$, (c) OASPL difference between (a) and (b).

between the beginning of Cruise 1 and end of Cruise 3 sections of the SAR mission. C_T is an integral non-dimensional metric of rotor disc loading; hence its gradual reduction within the mission course due to fuel burn is essentially reflected on the corresponding loading noise acoustic spheres, as depicted in Fig. 6.8.

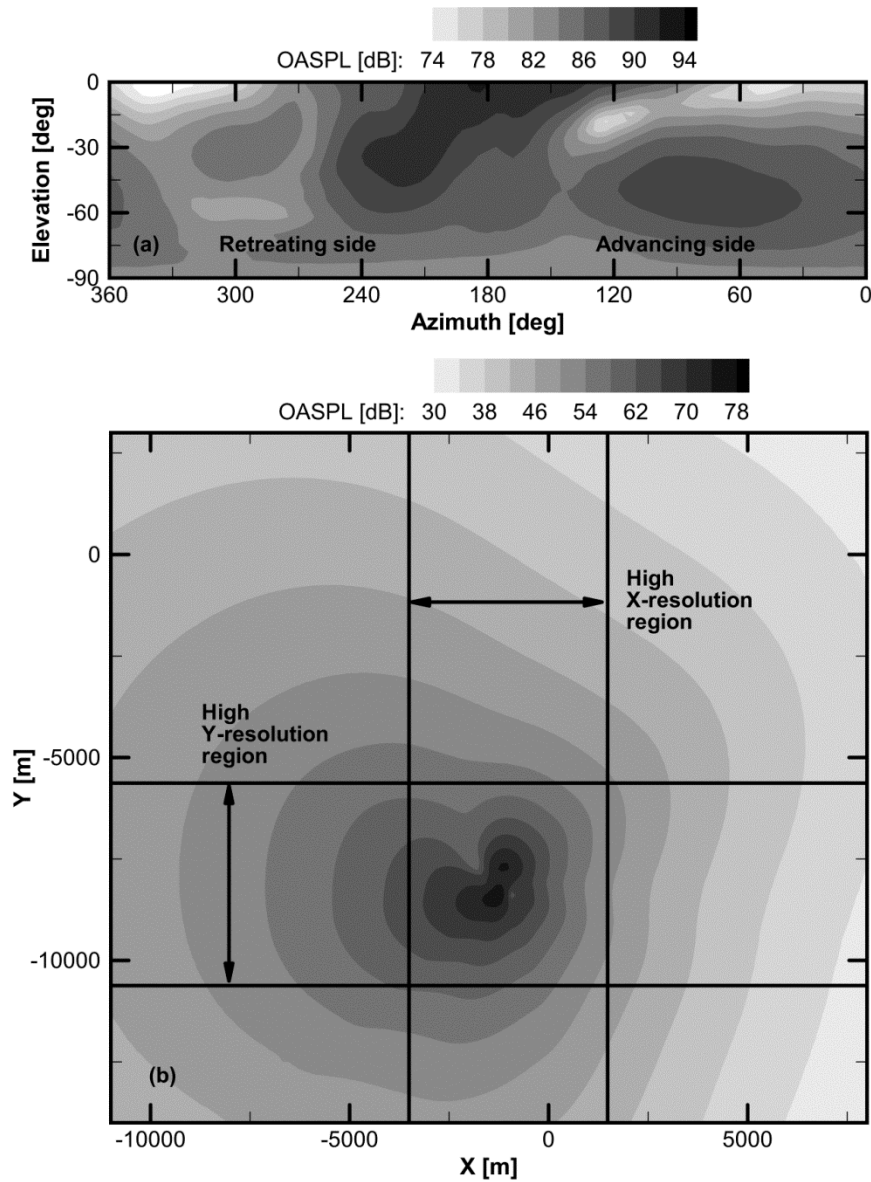


Figure 6.9: Selected trajectory waypoint of the PAT scenario; (a) acoustic sphere, (b) corresponding ground noise footprint at $t = 1180$ s, $\mu = 0.230$, $\gamma = 0$ deg., $C_T = 0.0060$.

A two-dimensional representation of the acoustic sphere is adopted in the context of this work, in a way that the three-dimensional spherical surface is “un-wrapped” on a two-dimensional map. The center of the acoustic sphere is fixed to the aircraft hub. The azimuth angle is measured from the tail following the direction of rotation of the main rotor. The elevation angle is measured from the horizontal plane attached to the rotor hub and is positive upwards. Only the lower noise hemisphere is presented since the upper does not radiate noise to the ground.

Differences of up to ± 5 dB are present on the acoustic spheres between the beginning and the ending of the mission. It is noted that both μ and γ are the same between the compared mission time-instances. The relative deviations in the rotor trim angles and blade flapping angles are less than 3%, leading to negligible directivity variations between the compared states; hence the dominant factor that changes between the two trim states is C_T . The reduction of C_T is attributed to the reduction of fuselage AUM which affects the vertical force requirement for the main rotor. This results in a more negative tip-path-plane angle, increased downwash and reduced angles of attack at the aft region of the rotor disc. This leads to a re-distribution of the elementary thrust across the rotor disc by increasing the loading at the front and significantly reducing at the aft region of the rotor disc. Consequently, the OASPL of the advancing side of the noise hemisphere is reduced, whilst it increases at the retreating side. It becomes clear that the impact of fuel burn is crucial on the prediction of helicopter AUM, C_T and finally, ground noise, especially for long-range missions.

6.5 Time-Domain Analysis

Fig. 6.8 and Fig. 6.9 demonstrate the acoustic spheres (a) and the corresponding ground footprints (b) for the selected trajectory waypoints of the PAT and SAR case, respectively. The high resolution regions in the X and Y direction are also superimposed. Considering the adopted ground grid set-up as described in Section 6.2, a resolution of 30 m is adopted in the high-gradient

region and 180 m in the rest parts of the ground grid in both Cartesian directions.

As a result, a total of 56,538 observers occur for the PAT case, resulting in an 84.24% reduction compared to the uniform observer grid of 30 m resolution. For the SAR scenario, a total of 238,252 acoustic observers have been utilized,

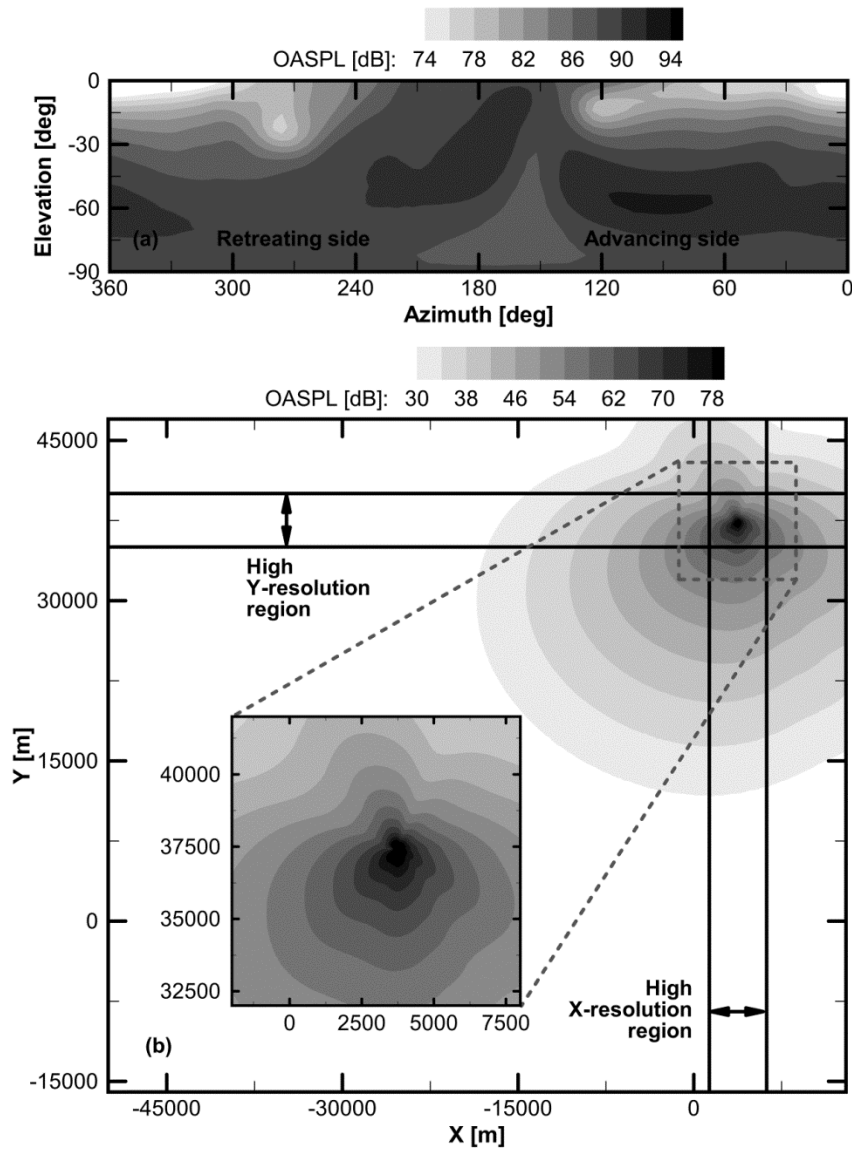


Figure 6.10: Selected trajectory waypoint of the SAR scenario; (a) acoustic sphere, (b) corresponding ground noise footprint at $t = 4990$ s, $\mu = 0.185$, $\gamma = -7.13$ deg., $C_T = 0.0047$.

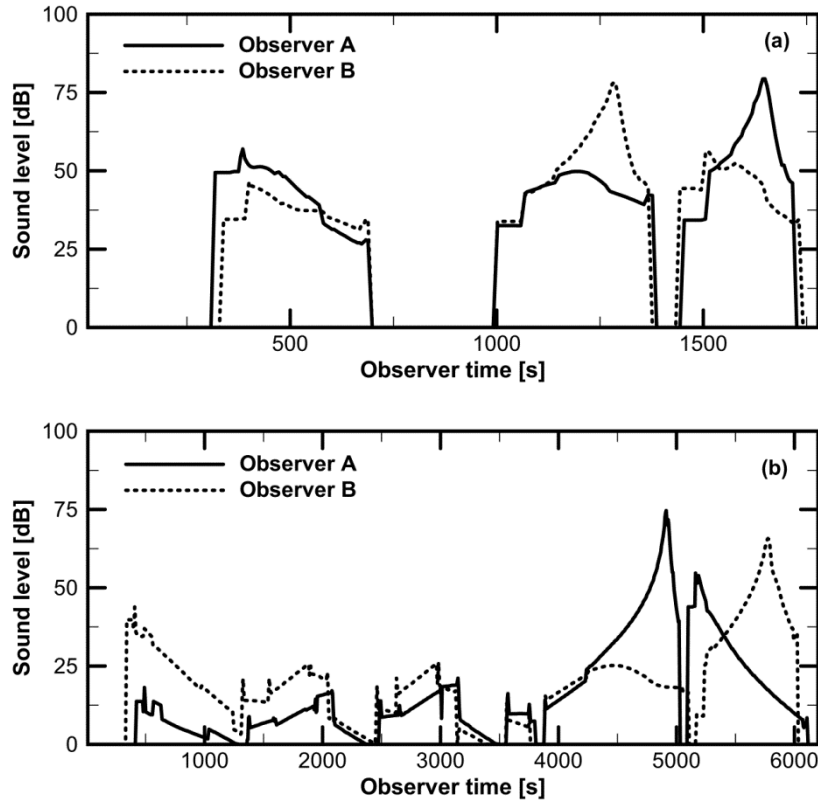


Figure 6.11: Acoustic histories for selected isolated observers A and B for each mission; (a) PAT, (b) SAR.

resulting in a 94.59% reduction in the number of employed observers, relative to the uniform grid configuration. It is noted that the high-resolution region follows the motion of the helicopter; hence a new computational grid is set up at each trajectory waypoint to capture the complex ground noise pattern with the least computational effort.

For the PAT mission, the selected trajectory waypoint corresponds to cruise conditions, at an advance ratio $\mu = 0.23$ and altitude $h = 450$ m. Thickness noise is dominant in the plane and upstream of the rotor, thus a strong peak is observed at $\psi = 180$ deg. and $\beta = 0$ deg. on the acoustic sphere. Significant loading noise is observed at $\psi = 90$ deg. and $\beta = -45$ deg. generated by the aft loaded region of the rotor. For the SAR case a time-instant at altitude $h = 132$ m with the helicopter in descent with $\gamma = -7.13$ deg. at advance ratio $\mu = 0.18$ is

selected. The BVIs generated due to negative disc incidence angle leads to increased out-of-plane loading noise, especially below the advancing side at $\psi = 90 \text{ deg.}$ and $\beta = -60 \text{ deg.}$

The aforementioned source-noise patterns are reflected on the ground footprint contours. Increased noise levels are observed upstream of the helicopter in the PAT selected waypoint, whilst significant noise radiation is observed below the advancing side in the SAR selected snapshot as shown in the magnified area of Fig. 6.10 b). It can also be observed that the affected ground regions extend to kilometre-range distances. The quantification of the corresponding noise exposure times becomes of utmost priority for the mission-level assessment of the acoustic impact on the ground.

Fig. 6.11 presents the observer-time distributions of noise levels for two isolated observers (A and B) for the PAT (a) and the SAR (b) mission. The observer time is calculated using Eq. 3.25 with the aircraft-time (mission-time) serving as source time. The noise travel-time is estimated assuming uniform atmosphere and utilising the radiation distance between the rotor hub and the observer. Negligible rotor noise radiation is assumed during idle periods. For the PAT scenario, generally higher noise levels are observed, because of the shorter range, which keeps the helicopter close to the observers during the entire mission. Additionally, the cruise advance ratio is higher compared to the SAR mission; hence the higher thickness noise levels contribute significantly to the overall noise.

The highest noise levels are recorded when the radiation distance is minimised. However, there is an extended time-period before and after the peaks, where gradually increasing and decreasing, respectively, noise levels are recorded. Some short-duration peaks are present in the SAR case for both the examined observers, which are due to the banked turns which take place during the search and rescue mission task element. The change of orientation results in slight peaks in the C_T time history as seen in Fig. 6.11 b), which contribute to increased loading noise. Moreover, significant thickness noise radiation towards the observers A and B occurs during the orientation changes across the SAR

mission task elements. The relative time-exposure is of importance for the holistic mission-level noise assessment. This indicates that a cumulative metric has to be adopted, that apart from absolute noise levels, it also accounts for noise exposure times.

6.6 Cumulative Metrics of Mission Acoustic Impact

In Section 6.5 it was shown that rotorcraft missions can affect wide ground regions and for extended periods of time. Cumulative metrics are essential for the inclusion of both spatial and temporal effects into the mission acoustic impact assessment. Fig. 12 a) and b) illustrate the sound exposure levels (SEL) for the PAT and SAR missions, respectively. SEL is defined as the energy averaged sound level over a specified period of time or single noise event, with reference duration of 1 second (Ref. [155]):

$$SEL = 10 \log_{10} \left[\frac{1}{T_0} \int_{t_1}^{t_2} 10^{\frac{OASPL(t)}{10}} dt \right], \quad (6.1)$$

where T_0 is the reference duration of 1 sec and $t_2 - t_1$ is the actual duration of the acoustic event. In the context of this study, the SEL is calculated applying continuous time integration for the entire exposure period at each observer position, as followed in Ref. [8]. The PAT scenario has generally higher overall noise levels due to the higher cruise speeds compared to the SAR mission.

It is observed that the ground regions located before the landing points experience the highest SEL. That is due to temporal annoyance effects accounted for in the calculation of the SEL. The in-plane directed thickness noise is focused on the ground upstream of the helicopter during trimmed descent; hence it has a continuous contribution in the upstream ground regions, up until the landing points. Additionally, as shown in Fig. 6.10, significant BVI noise contributions are added to the overall annoyance during descent, which also have temporal contribution to the SEL.

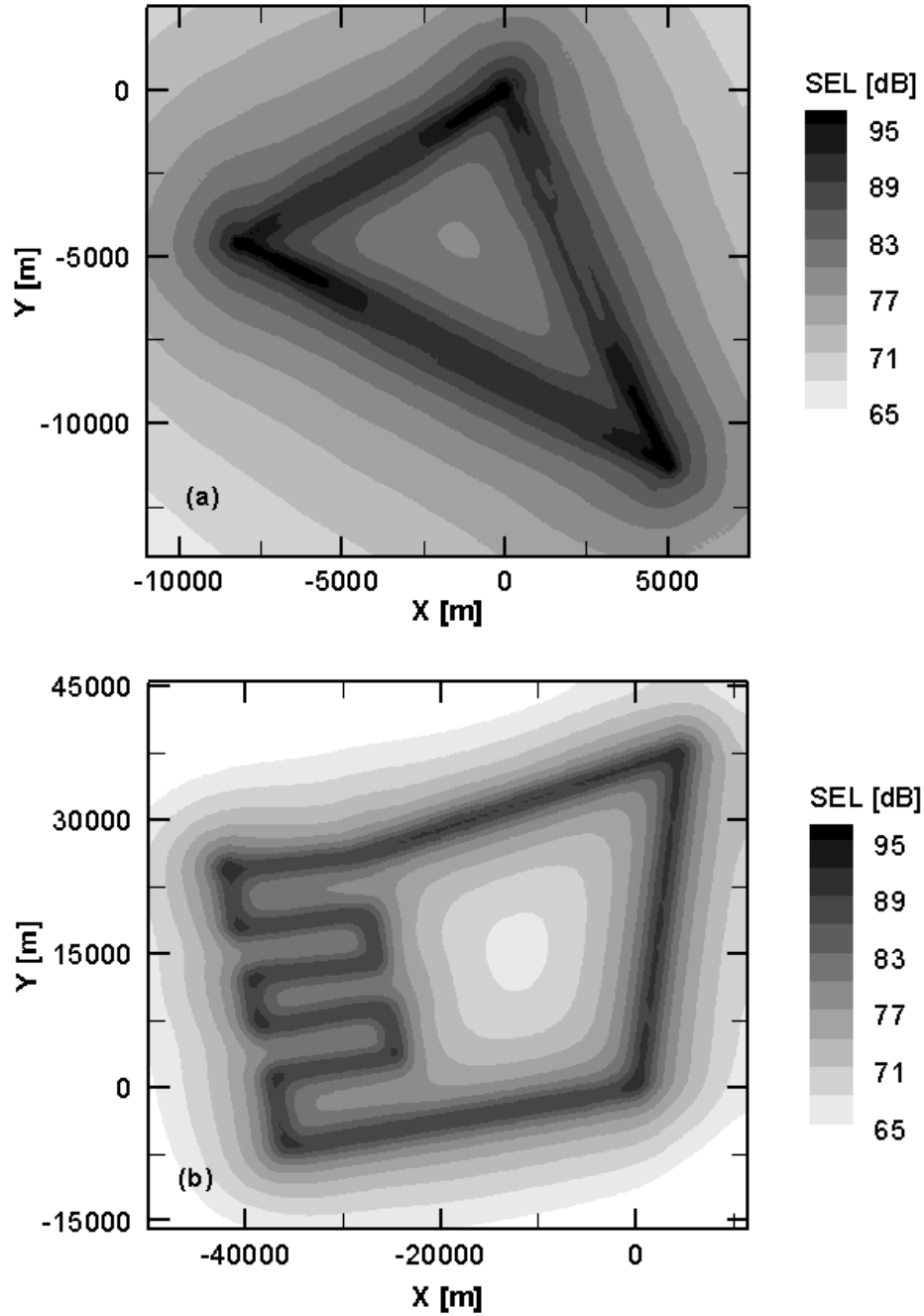


Figure 6.12: Sound exposure levels; (a) PAT, (b) SAR. Uniform input.

6.7 Assessment of Computational Performance

The computational performance of the individual models is assessed to quantify the computational savings introduced by the parallelization and the implementation of trajectory adaptive ground observer grids. The free-wake

Table 6.3: Rotor airloads model computational performance.

No of threads	CPU time [mins]		
	$\Delta\psi=1$ deg	$\Delta\psi=2$ deg	$\Delta\psi=5$ deg
1 thread	167.2	44.5	23.8
8 threads	29.5	7.1	4.3

rotor airloads code and the rotor aeroacoustics code are assessed separately for individual trim states, although the exported outcomes can be extended to mission level. The assessment has been undertaken on a personal computer equipped with 16 gigabytes (GB) of RAM and a central processing unit (CPU) with 4 cores (8 threads) operating at 3.60 GHz.

Table 6.3 presents the elapsed computational times for the free-wake rotor airloads code to trim the rotor for the selected trajectory waypoint of the SAR mission presented in Section 6.5. Computational savings of up to 84.05% are recorded between the single and the 8-threaded execution for the $\Delta\psi=2$ deg. azimuthal resolution case.

Table 6.4 presents a comparison of elapsed computational times for the aeroacoustics module of the framework referring to one single trim point of the PAT mission scenario. The trim point under investigation is the selected

Table 6.4: Rotor aeroacoustics model computational performance.

No of threads/ Grid structure	CPU time [mins]		
	$\Delta\psi=1$ deg	$\Delta\psi=2$ deg	$\Delta\psi=5$ deg
1 thread/Uniform grid	448.3	222.7	91.5
8 threads/Uniform grid	112.4	56.8	22.3
1 thread/Adaptive grid	180.1	89.6	36.2
8 threads/Adaptive grid	45.8	21.5	8.1

trajectory waypoint of the PAT case presented in Section 6.5. Computational savings of up to 76.00% are recorded between the single and the 8-threaded execution for the $\Delta\psi=2$ deg. azimuthal resolution case with adaptive ground grids implemented. The employed number of observers is 56,538, as specified in Section 6.5. It can be seen that the use of the proposed grid adaptation method results in a 62.15% reduction in computing time requirements. This allows the method to be used on-the-fly without the need of pre-stored noise data.

6.8 Discussion

An analytical approach has been developed for the prediction of the acoustic impact of three-dimensional rotorcraft operations. It comprises a mission analysis comprehensive code, an aero-elastic high resolution airloads free wake model and the newly developed rotor aeroacoustics model. A procedure for the generation of acoustic spheres on-the-fly, coupled with trajectory-adaptive ground observer grids has been incorporated into a mission-level simulation environment. The developed methodology has been successfully applied towards the assessment of ground noise impact of two generic missions of a TEL helicopter model, representative of the Airbus Helicopters Bo 105.

The on-the-fly acoustic sphere generation method, which employs computationally efficient mission flight dynamics coupled with high-resolution aero-elastic free wake computations, has been the enabler for the mission-level application of noise prediction. When combined with trajectory-adaptive ground grids, reductions of up to 94.59% have been calculated in the total of employed ground observers, relative to uniform grid configurations for a long range mission. The devised methodology has been successful in handling long-range and long-duration rotorcraft missions. The impact of fuel burn across the mission has been proven crucial for the long-duration missions. A 9.26% reduction in the value of main rotor thrust coefficient has been observed between the beginning and the ending of the mission, which corresponds to differences of up to ± 5 dB on the source loading noise radiation.

The analytical approach implemented in this chapter enables the implementation of physics-based rotor noise prediction in three-dimensional mission simulations, keeping the associated computational overhead at acceptable levels. This renders the tool suitable for industry-standard noise prediction, in the context of mission planning and potentially, operational optimisation. The proposed strategy constitutes a generalised methodology that is independent of pre-stored acoustic data. It is offered in a form that is readily available for application on any designated three-dimensional rotorcraft operation.

7 Results and Discussion; Multi-Disciplinary Optimisation of Morphing Helicopter Rotors

7.1 Introduction

This chapter investigates the potential of simultaneous mitigation of the environmental and acoustic impact of VRS/ABT helicopters in descent. The holistic multi-disciplinary optimisation approach introduced in Section 3.5.2 is applied towards the concurrent minimisation of $EAOASPL$ and NO_x^{rate} . A Design Space Exploration (DSE) approach comprising full-factorial Design of Experiments (DOE) and Kriging-based Response Surface Modelling (RSM) is utilised to approximate the physical behaviour of the system. The predictive accuracy of the derived RSMs is assessed through Leave-One-Out Cross-Validation. The complex system's behaviour is explored to identify trade-offs between $EAOASPL$ and NO_x^{rate} .

The multi-disciplinary optimisation strategy is employed for the derivation of optimised VRS and ABT schedules which are able to minimise noise, NOx or achieve an adequate trade-off between these two parameters. The schedules are provided in a generalised form, covering the entire envelope of descending flight and have the potential to be directly applied towards the reduction of adverse environmental and acoustic impact of VRS/ABT helicopters.

7.2 Cross Validation

The helicopter under investigation is the Airbus Helicopters Bo 105 which has been analysed in Chapter 6, hence no further description is provided. The six-dimensional design space is defined by the following design and operational parameters: (a) variable rotor speed, (b) active blade twist angle, (c) rotorcraft all-up-mass, (d) density altitude, (e) flight speed and (f) rate of climb. The design space is discretised using a non-uniform, full-factorial sampling design

Table 7.1: Design space bounds and values of design and operational parameters to be directly simulated for the DSE process. Reference values annotated in parentheses and bold font.

Parameter	Values directly simulated	Unit
All-up-mass AUM	1600, (2000) , 2400	kg
Density altitude h	0, (250) , 500	m
Flight speed V_x	20, (30) , 40	m/s
Climb rate V_z	0, (-4) , -5	m/s
Rotor speed Ω	75, 80, 85, 90, 95, (100)	%
Blade twist angle θ_{tw}	2.4, 5.2, (8.0) , 10.8, 13.6, 16.40	deg

as presented in Table 7.1. Reference values are defined annotated in parentheses and bold. The three-step integrated procedure of Section 3.5.1 is utilised to obtain the $EAOASPL$ and NO_x^{rate} at each of the 2,916 sample points. The helicopter is trimmed in each and every point of the sample. A fixed azimuthal and polar discretisation step $\Delta\psi = \Delta\zeta = 6$ deg. is used for the noise hemispheres. A grid convergence investigation as described in Section 4.6 has been carried out for the reference operating and design conditions of Table 7.1. The deviation of $EAOASPL$ from the infinitely fine grid case was less than 0.07 dB, therefore the employed mesh is sufficiently accurate for this study.

A Kriging-based interpolation is employed to generate the RSMs for the approximation of system's behaviour, as explained in Section 3.5.1. In the present analysis, the Kriging model is implemented through quadratic regression combined with exponential auto-correlation, which has been shown to provide the most accurate approximations, as reported in Ref. [156].

Before commencing the optimisation process, the predictive accuracy of the derived RSMs is assessed through Leave-One-Out Cross-Validation (LOOCV),

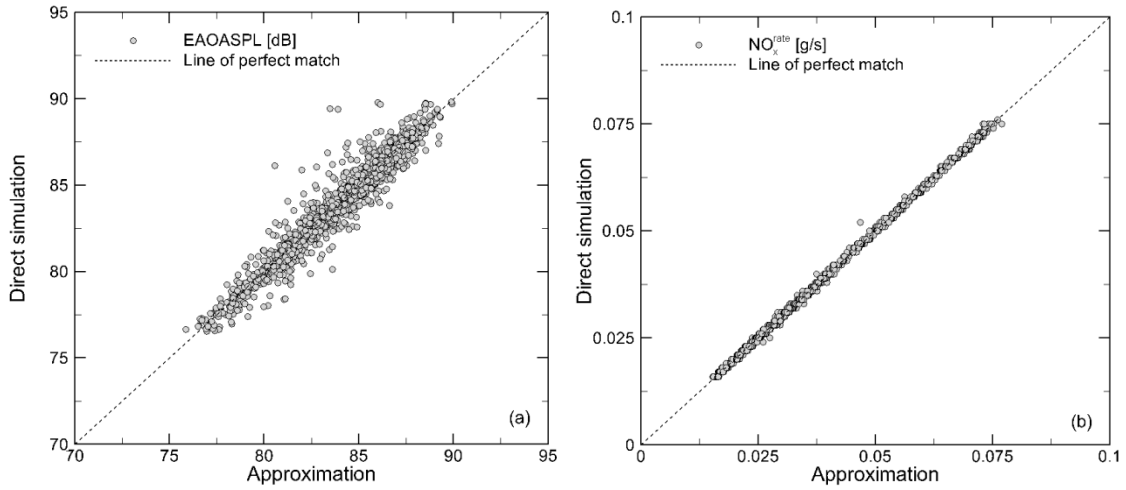


Figure 7.1: Leave-one-out Cross Validation applied to the developed RSM; (a) $EAOASPL$, (b) NO_x^{rate} .

as described in Section 5.2. The quality of the original RSMs is assessed in terms of Pearson's product moment of correlation $N_{Pearson}$ along with the gradient of the associated linear regression line (Ref. [112]).

Fig. 7.1 a) and b) present the LOOCV applied to the $EAOASPL$ and NO_x^{rate} RSMs, respectively. Good agreement is observed between the approximated and directly simulated values of $EAOASPL$ and NO_x^{rate} . It is noted that $EAOASPL$ points present a higher relative spread compared to the corresponding NO_x^{rate} points. This is due to the relatively low range of the $EAOASPL$ values, due to the logarithmic nature of the dB scale. The average absolute deviation between approximated and simulated $EAOASPL$ is 0.44 dB, which is sufficiently small for the range of expected benefits for this concept. Table 7.2 summarises the statistical properties of LOOCV for both objectives. Very good linear correlation is observed for both objectives, with $N_{Pearson}$ and gradients very close to the ideal values. This showcases the ability of the developed RSMs to replicate the physical behaviour of the three-step integrated simulation. Furthermore, the normalised root-mean-square error (NRMSE) of each objective is in the order of 1% which quantified the predictive accuracy of the developed RSMs. Finally,

Table 7.2: Statistical properties of LOOCV for the objectives of interest.

Objective	N_{pearson} [-]	Gradient [deg]	NRMSE [%]	σ [%]
$EAOASPL$	0.996	44.991	0.817	0.818
NO_x^{rate}	0.999	45.004	1.199	1.200

the corresponding standard deviation σ of the calculated error is again in the order of 1% for both variables. This indicates that the RSM error is uniformly distributed throughout the investigated design space.

7.3 Design Space Behaviour

The system's sensitivity to variations of Ω and θ_{tw} is explored at various operating conditions. Relative metrics have been adopted for the quantifications of this work, namely $\Delta NO_x^{\text{rate}}$ which represents the relative change of NO_x^{rate} with respect to the corresponding value at nominal Ω and θ_{tw} , and $\Delta EAOASPL$

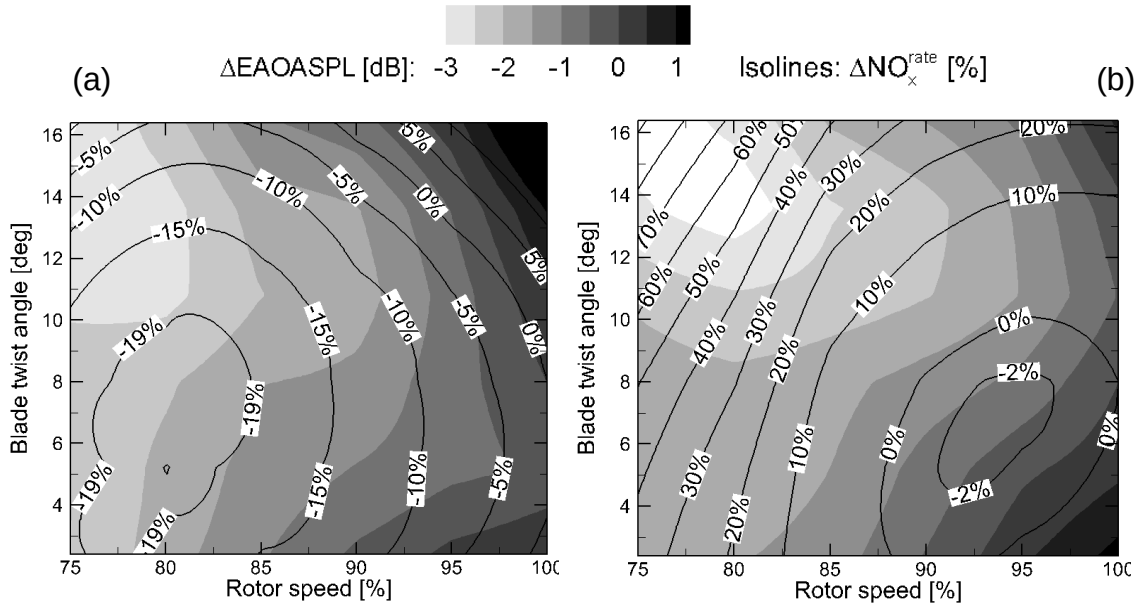


Figure 7.2: Design Space Exploration: impact of all-up-mass (AUM) on variable rotor speed and active blade twist noise and NO_x emissions at $h = 250$ m, $V_x = 30$ m / s, $V_z = -4$ m / s; (a) $AUM = 1600$ kg, (b) $AUM = 2400$ kg.

which represents the difference in dB relative to nominal rotor control settings.

Fig. 7.2 a) shows how noise and NO_x are affected from VRS/ABT at $AUM = 1600 \text{ kg}$. The rest flight conditions retain the reference values of Table 7.1. EAOASPL is minimum at low Ω and high θ_{tw} in these flight conditions.

Reduced Ω yields reduced acoustic radiation Mach numbers $\vec{M} \cdot \vec{r}$, thus a reduced Doppler amplification factor $(1 - \vec{M} \cdot \vec{r})^{-1}$ which, according to the derivations of thickness and loading noise in Formulation 1A of Farassat, mitigates the contribution of the acoustic terms of both noise components.

Variation in θ_{tw} leads to a re-distribution of aerodynamic loading across the rotor disc. Increased θ_{tw} tends to move the peaks of blade loading inboards. As shown in Fig. 4.3, the most severe BVIs occur at azimuthal angles around $\psi = 50 \text{ deg}$ and $\psi = 300 \text{ deg}$, focused at the outboard regions of the blades. This is due to the small-miss distance between trailed tip-vortices and rotor blades in these regions. Inboards re-distribution of loading leads to reduced circulation around the tip and eventually reduced BVI noise.

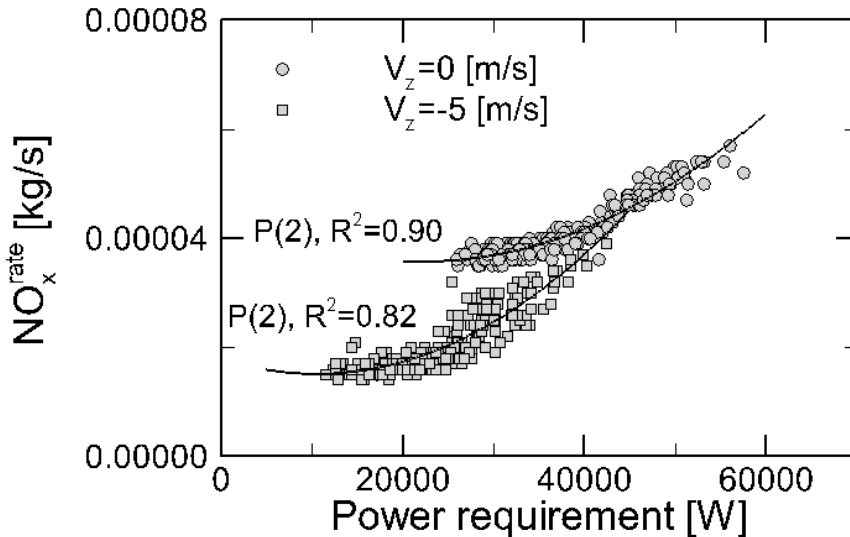


Figure 7.3: Correlation between rotor power requirement and NO_x for $V_z = 0 \text{ m/s}$ and $V_z = -5 \text{ m/s}$ at $AUM = 1600 \text{ kg}$.

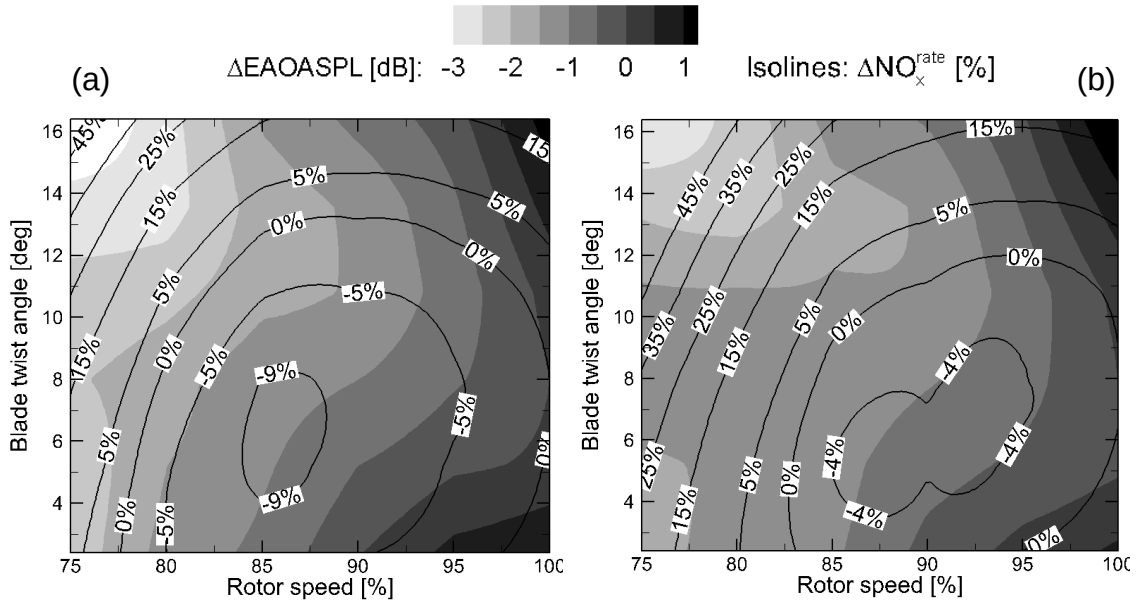


Figure 7.4: Design Space Exploration: impact of density altitude (h) on combined variable rotor speed and active blade twist noise and NO_x emissions at $AUM = 2000 \text{ m}$, $V_x = 30 \text{ m/s}$, $V_z = -4 \text{ m/s}$; (a) $h = 50 \text{ m}$, (b) $h = 450 \text{ m}$.

The $\Delta \text{NO}_x^{\text{rate}}$ isolines are superimposed to show the trade-off between noise and NO_x . NO_x emissions are dependent on the engine operating point which is determined by: (a) rotor aerodynamic performance and (b) engine thermodynamic performance. Fig. 7.3 presents the correlation between rotor power and NO_x . It is shown that families of correlated points are formed using a 2nd order polynomial regression with coefficients of determination $R^2=0.90$ and $R^2=0.82$ for $V_z = 0 \text{ m/s}$ and $V_z = -5 \text{ m/s}$, respectively. These refer to $AUM = 1600 \text{ kg}$, but similar response was found for the rest AUMs. The NO_x emission index rises exponentially with power (Ref. [146]) whilst specific fuel consumption (SFC) drops exponentially with power setting. This trade-off results in an overall balance in the correlation between NO_x and rotor power. The associated amplitude-normalised standard deviations are 12.36% and 25.33%, due to the impact of engine thermodynamic performance. However, the monotonically-ascending trend between the two objectives suggest that NO_x can be used as surrogate for rotor power in the analysis to follow.

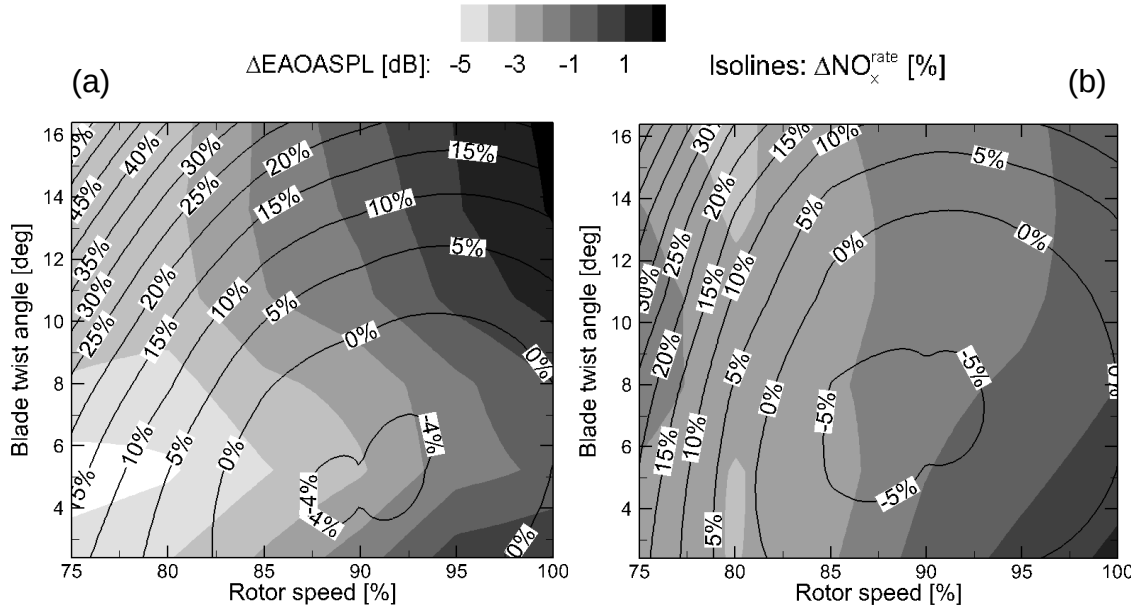


Figure 7.5: Design Space Exploration: impact of flight speed (V_x) on combined variable rotor speed and active blade twist noise and NO_x emissions at $h = 250 \text{ m}$, $AUM = 2000 \text{ kg}$, $V_z = -4 \text{ m/s}$; (a) $V_x = 20 \text{ m/s}$, (b) $V_x = 40 \text{ m/s}$.

In terms of main rotor aerodynamics and performance, variation of Ω affects profile drag due to altered rotor dynamic head, as well as blade local pitch angle, due to the collective and cyclic pitch control variations necessary to restore rotor thrust and meet trim requirements (Ref. [101]). This leads to variations in blade loading (Ref. [112]). Excessive increase in blade loading can negatively affect the aerodynamic performance of rotor blade elements and eventually, increase rotor power requirements.

As regards engine performance, Walsh and Fletcher (Ref. [157]) showed that at a given gas generator (GG) rotational speed, there is a uniquely nominal free-power turbine (FPT) speed that minimises SFC and maximises power generation, as also discussed in Ref. [112]. A reduction in FPT speed relative to the nominal leads to reduced thermal efficiency and increased SFC. Therefore, the overall impact of VRS on fuel burn and NO_x is determined by the trade-off between engine thermodynamic deficit and rotor aerodynamic benefit.

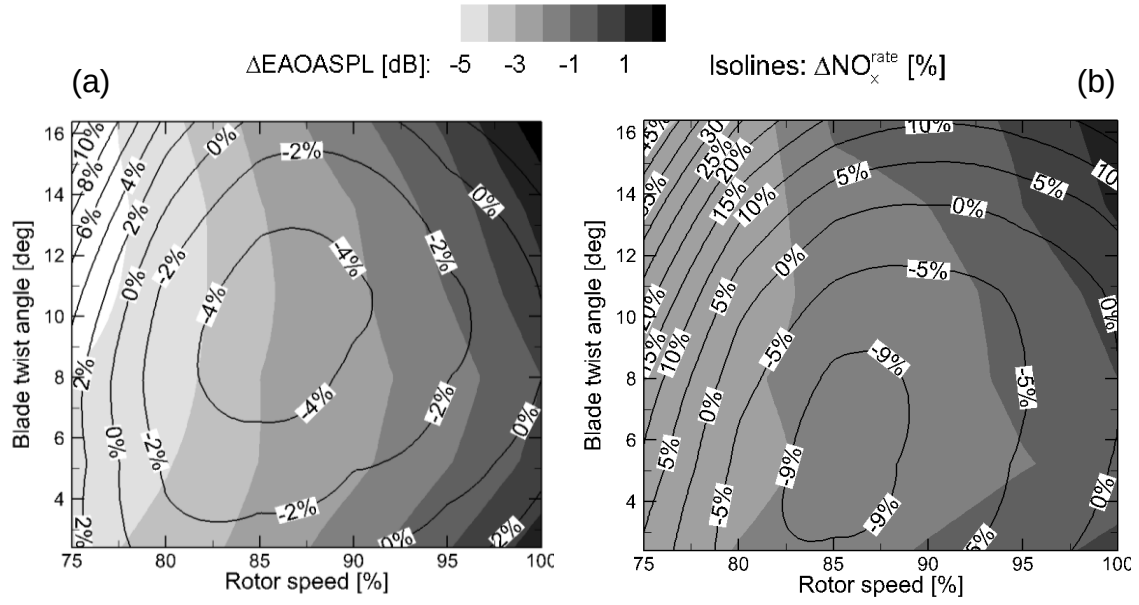


Figure 7.6: Design Space Exploration: impact of climb rate (V_z) on combined variable rotor speed and active blade twist noise and NO_x emissions at $h = 250 \text{ m}$, $V_x = 30 \text{ m/s}$, $AUM = 2000 \text{ kg}$; (a) $V_z = 0 \text{ m/s}$, (b) $V_z = -5 \text{ m/s}$.

Variations in θ_{tw} primarily affect rotor induced power which is dominant in the low-speed region of the flight envelope. At low-speed conditions, an increase in θ_{tw} provides a more uniform inflow through the re-distribution of radial blade circulation, which leads to reduced induced losses.

Fig. 7.2 b) presents the corresponding behaviour of $\Delta\text{EAOASPL}$ and $\Delta\text{NO}_x^{\text{rate}}$ at $AUM = 2400 \text{ kg}$. It can be seen that Ω higher than 90% is required to achieve $\text{NO}_x^{\text{rate}}$ reductions. A lower Ω in this case would result in a very high rotor collective pitch angle to sustain trim, at a cost of overly increased aerodynamic loading and consequently, power requirement and NO_x emissions. At high AUM, reduction in omega results in both aerodynamic and thermodynamic performance deficit, which give a steep gradient in NO_x production rate as Ω reduces. This is in accordance with the findings reported in Refs. [101], [105], [112], which indicate higher performance benefits at “low-and-light” flight conditions. As regards noise, the minimum Ω is required due to Doppler

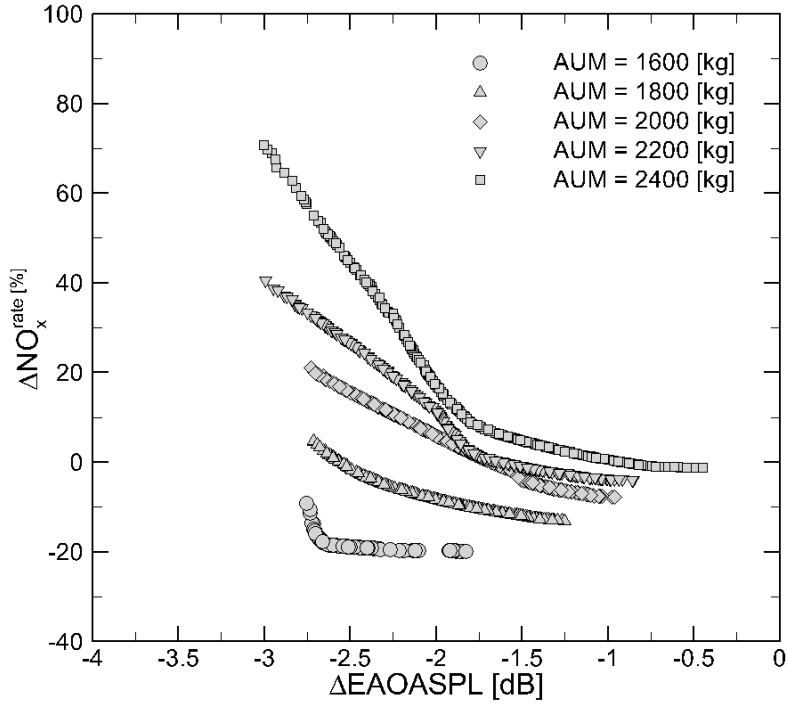


Figure 7.7: Impact of all-up mass (AUM) on Pareto fronts of optimised combined variable rotor speed and active blade twist helicopter rotor at $h = 250 \text{ m}$, $V_x = 30 \text{ m/s}$, $V_z = -4 \text{ m/s}$.

amplification, whilst a higher value of θ_{tw} is required due to the increased blade loading occurred from increased AUM.

Fig. 7.4 a) and b) depict the impact of density altitude h on $\Delta EAOASPL$ and ΔNO_x^{rate} under variations of Ω and θ_{tw} . Due to the relatively low reference flight speed $V_x = 30 \text{ m/s}$, the impact of h is primarily attributed to variations in rotor induced power. Hence, adjustment of blade twist is warranted to minimise induced losses for a given rotor speed. Fig. 7.5 a) and b) discuss the impact of horizontal flight speed (V_x) on $\Delta EAOASPL$ and ΔNO_x^{rate} under variations of Ω and θ_{tw} . It is noted that the rest of flight conditions retain their reference values as in Table 7.1. At $V_x = 20 \text{ m/s}$, freestream upwash is significant relative to V_x which leads to excessive increase in the radially inwards blade loading. A θ_{tw} lower than the nominal is required to re-distribute airloads and reduce noise.

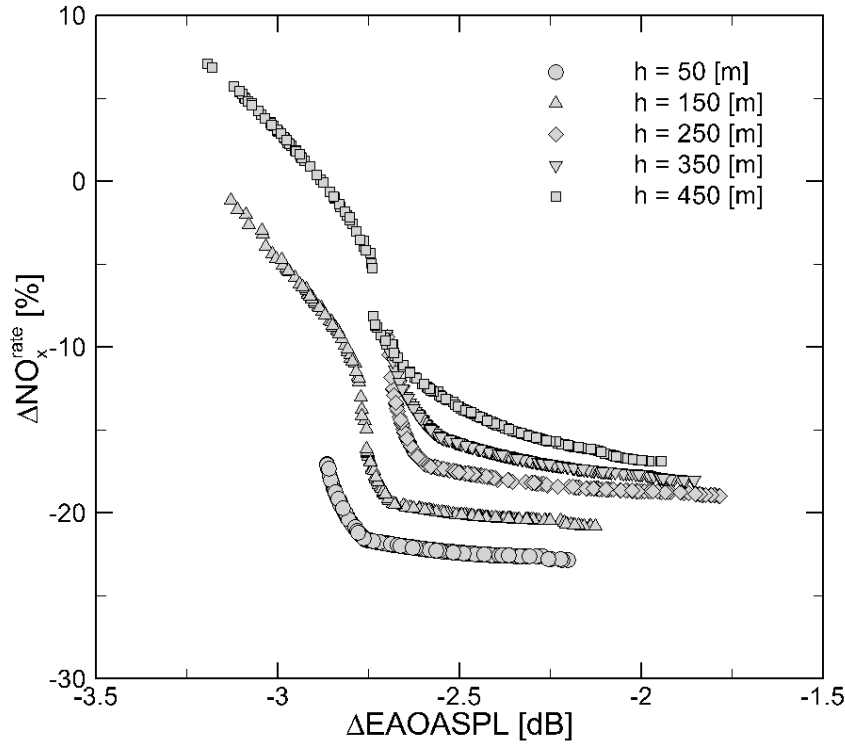


Figure 7.8: Impact of density altitude (h) on the Pareto fronts of optimised combined variable rotor speed and active blade twist helicopter rotor at $AUM = 2000 \text{ kg}$, $V_x = 30 \text{ m/s}$, $V_z = -4 \text{ m/s}$.

At $V_x = 40 \text{ m/s}$ the relative contribution of $V_z = -4 \text{ m/s}$ is lower, hence an increase in θ_{tw} is again required to reduce $EAOASPL$. As regards NO_x^{rate} , no significant deviations are observed between the two states. This is attributed to the physical behaviour of the main rotor power curve which is typically described by a “bucket” shape (Ref. [17]). The examined values of $V_x = 20 \text{ m/s}$ and $V_x = 40 \text{ m/s}$ approximately correspond to the same power requirement, hence the distribution of ΔNO_x^{rate} follows similar patterns in both flight states.

Fig. 7.6 a) and b) present the distributions of $\Delta EAOASPL$ and ΔNO_x^{rate} at climb rates $V_z = 0 \text{ m/s}$ and $V_z = -5 \text{ m/s}$, respectively. In level flight, high θ_{tw} is necessary to achieve the highest $\Delta EAOASPL$, whilst in descent lower θ_{tw} is

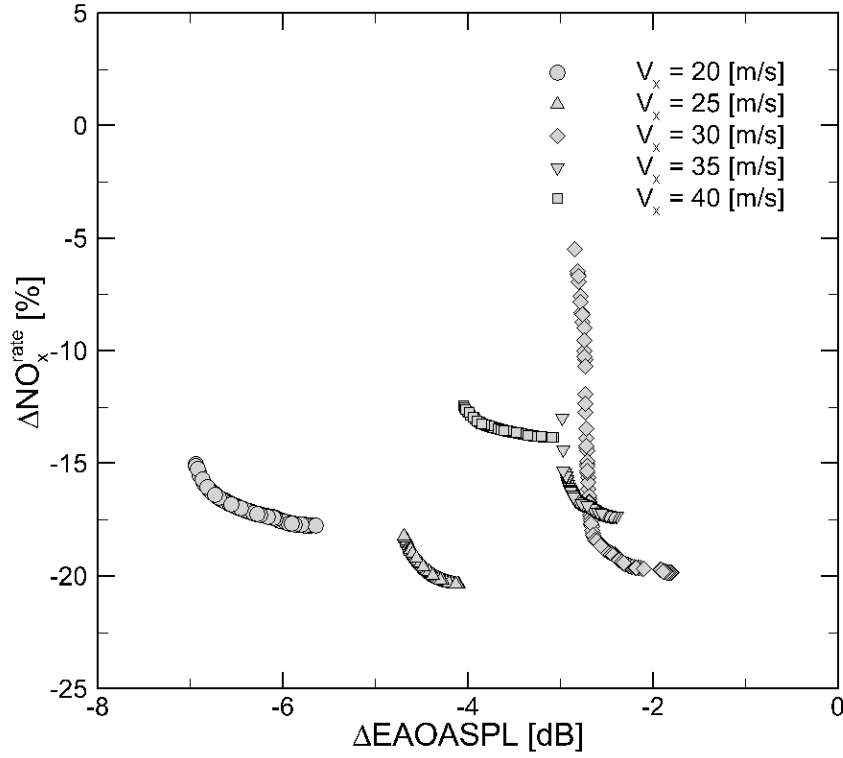


Figure 7.9: Impact of flight speed (V_x) on the Pareto fronts of optimised combined variable rotor speed and active blade twist helicopter rotor at $h = 250 \text{ m}$, $AUM = 2000 \text{ kg}$, $V_z = -4 \text{ m/s}$.

required to compensate for the $V_z = -5 \text{ m/s}$ freestream velocity component. In terms of NO_x^{rate} , higher benefits are achieved in steep descent due to the significantly lower power setting of the engine.

7.4 Trade-Off between Source Noise and NO_x Emissions

The analysis of design space behaviour highlighted the potential of combined VRS/ABT rotor morphing in reducing the $EAOASPL$ emissions and NO_x^{rate} , as well as depicted the strong trade-offs between the aforementioned two objectives. These give space for multi-disciplinary optimisation targeting the concurrent mitigation of acoustic and environmental impact of descending helicopters. The NSGA-II-based optimisation strategy introduced in Section

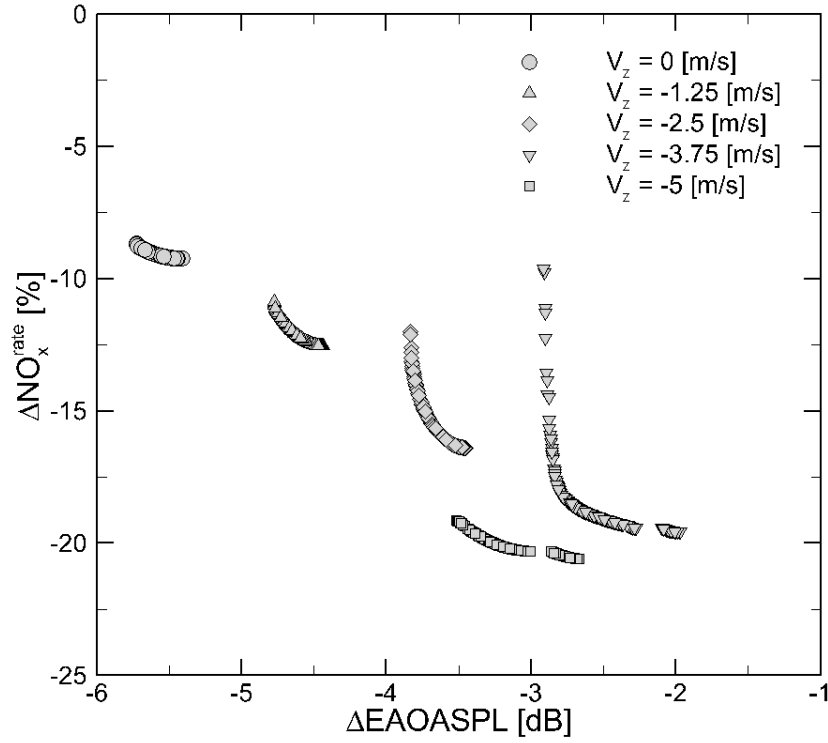


Figure 7.10: Impact of climb rate (V_z) on the Pareto fronts of optimised combined variable rotor speed and active blade twist helicopter rotor at $h = 250 \text{ m}$, $V_x = 30 \text{ m/s}$, $AUM = 2000 \text{ kg}$.

3.5.2 is deployed for the export of Pareto optimal fronts in a parametric manner for a series of operating conditions.

Fig. 7.7 illustrates the Pareto optimal fronts corresponding to five different AUM values, ranging from $AUM = 1600 \text{ kg}$ to $AUM = 2400 \text{ kg}$. The rest of operating parameters have their reference values as presented in Table 7.1. It is observed that the highest noise benefits can be achieved at high AUM. On a highly loaded rotor, the circulation peaks are located closer to the blade tips, compared with a lightly loaded rotor (Ref. [17]). Hence, ABT can more efficiently off-load the tips by moving the circulation peaks inboards. The highest NO_x benefits are attained at lightly-loaded rotor disc conditions, as also reported in Ref. [112]. However, it is observed that no significant $\text{NO}_x^{\text{rate}}$ reductions can be achieved at high AUM. Fig. 7.8 presents the Pareto fronts corresponding to a

series of density altitudes ranging from $h = 50 \text{ m}$ to $h = 450 \text{ m}$. It can be seen that the NO_x optimal solutions are significantly affected by altitude variations, offering up to 22.53% reductions in $\text{NO}_x^{\text{rate}}$. On the other hand, EAOASPL is not considerably altered in the investigated range of density altitude variations.

The impact of horizontal flight speed (V_x) on the trade-off between $\Delta\text{EAOASPL}$ and $\Delta\text{NO}_x^{\text{rate}}$ for VRS/ABT rotors is assessed in Fig. 7.9. It is shown that the highest EAOASPL benefits are achieved in low flight speed for this specific climb rate. This can be attributed to the impact of ABT in this speed regime. Specifically, the combination of $V_x = 20 \text{ m/s}$ with the reference climb rate $V_z = -4 \text{ m/s}$ leads to a considerable increase in radially inboards airloads due to impact of the vertical free-stream velocity component. ABT can effectively mitigate these effects by reducing θ_{tw} to values lower than the nominal.

As regards NO_x , relatively “flat” Pareto fronts are obtained for most V_x settings, due to the shape of the power curve in this flight speed regime. As shown in Ref. [146], the variation of rotor power requirements as function of flight speed follows a “bucket” shape, with zero gradient between $V_x = 20 \text{ m/s}$ and $V_x = 30 \text{ m/s}$ for this specific helicopter. This is reflected on the sensitivity of $\Delta\text{NO}_x^{\text{rate}}$ to V_x variations undertaken in this investigation. However, a considerably wide range in potential NO_x improvements, as well as non-monotonicity with respect to V_x , is observed at $V_x = 30 \text{ m/s}$. Considering the reference climb rate $V_z = -4 \text{ m/s}$ and the 3 deg. longitudinal tilt of the Bo 105 rotor shaft, an equivalent “wind tunnel” shaft angle of $\alpha_s = -4.59 \text{ deg}$ would occur. This is the angle which is most sensitive to BVI phenomena (Ref. [32]) hence the variations of Ω and θ_{tw} have a strong impact on tip-vortex formation and consequently, rotor performance and NO_x emissions.

Similar trends are shown in Fig. 7.10, which presents the impact of V_z on the trade-off between $\Delta\text{EAOASPL}$ and $\Delta\text{NO}_x^{\text{rate}}$ for VRS/ABT helicopters. A wide

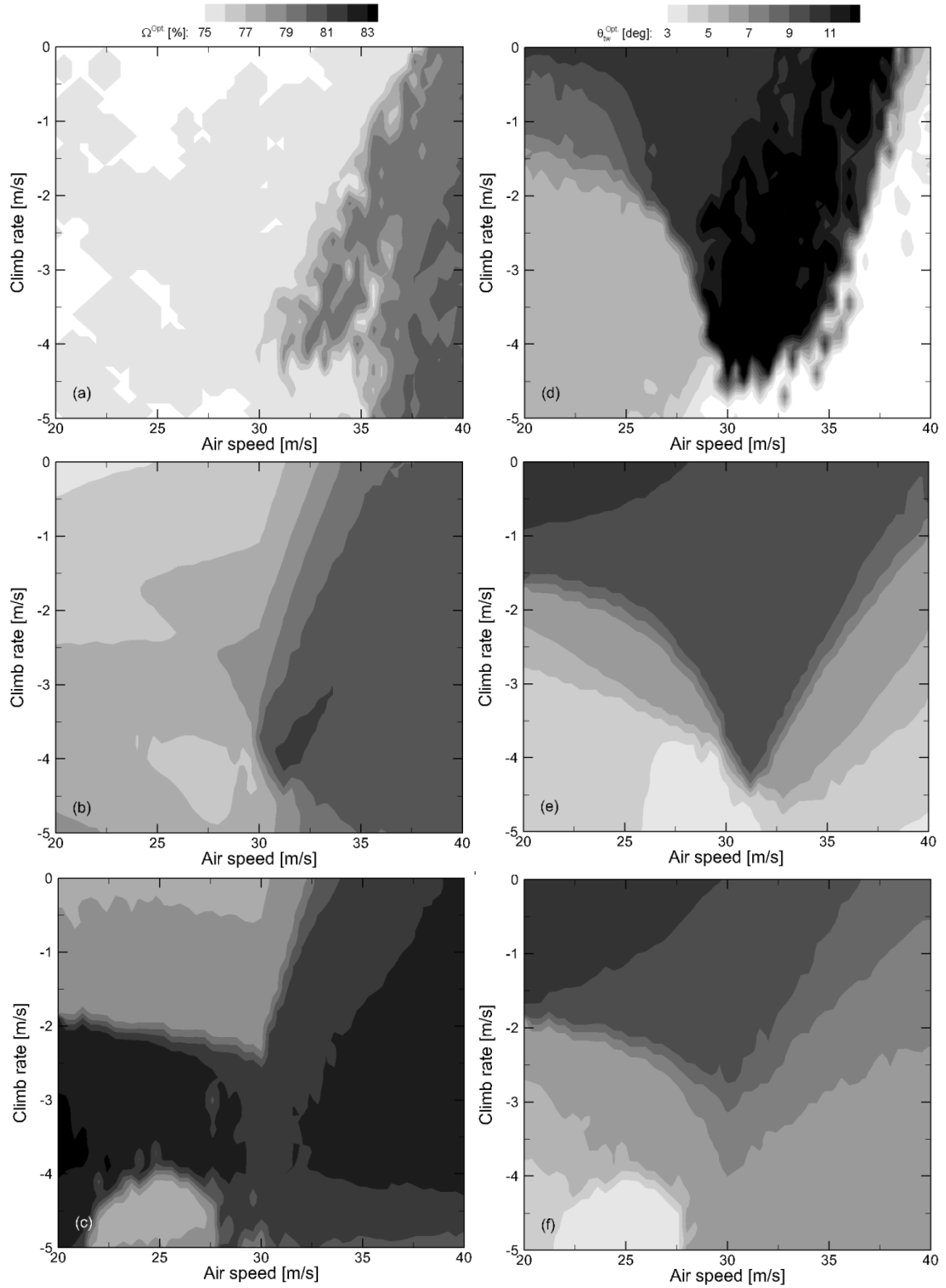


Figure 7.11: Optimised rotor control schedules for $AUM = 1600 \text{ kg}$: Optimal rotor speed for (a) Noise, (b) trade-off, (c) NO_x ; Optimal blade twist angle for (d) Noise, (e) trade-off, (f) NO_x .

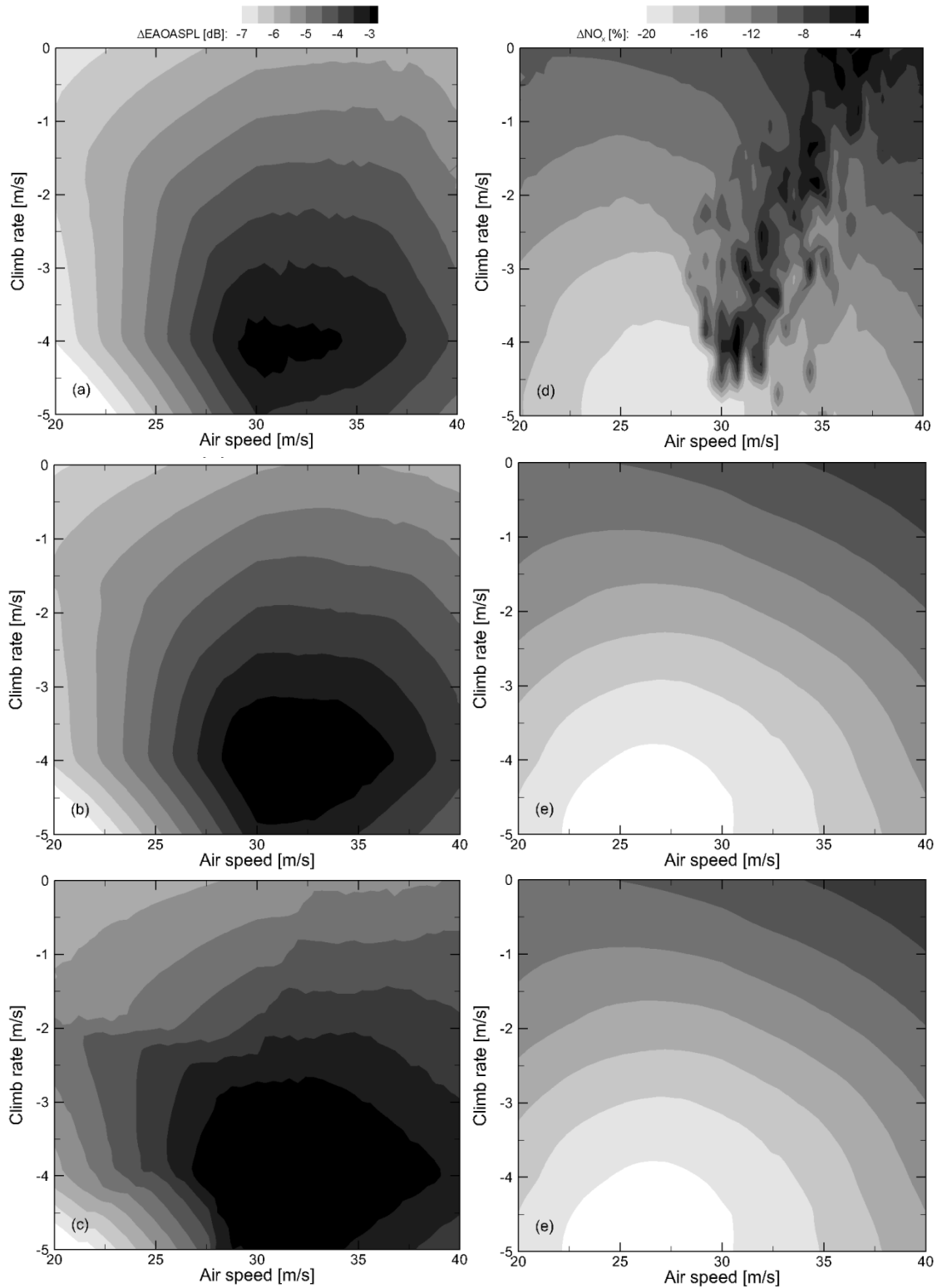


Figure 7.12: Acoustic and environmental benefits for $AUM = 1600 \text{ kg}$: Noise benefits through optimising for (a) Noise, (b) trade-off, (c) NO_x; NO_x benefits through optimising for (d) Noise, (e) trade-off, (f) NO_x.

range of NO_x^{rate} reductions is observed at $V_z = -3.75 \text{ m/s}$ which replicates the strong BVI conditions of the HART II experiment (Ref. [32]). The highest ΔNO_x^{rate} benefits are obtained at $V_z = -5 \text{ m/s}$ which represents the most lightly loaded condition for the rotor in the examined flight envelope. The highest $\Delta EAOASPL$ reductions are achieved at level-flight ($V_z = 0 \text{ m/s}$). In low-speed level-flight, the main rotor vortical wake encounters strong roll-up leading to the formation of a pair of intense trailing vortices let from the sides of the rotor, similar to the wing tip-vortices generated by fixed-wing aircraft (Re. [17]). The concentration of strong vortical structures at the two sides of the rotor results in the formation of two strong loading cells at rotor azimuth angles $\psi = 80 \text{ deg}$ and $\psi = 280 \text{ deg}$ and located in the blade tip region. These constitute significant sources of loading noise, which is effectively mitigated by ABT, through moving the circulation radial peaks towards the inboard rotor direction, which finally alters the tip-vortex circulation and roll-up structure.

7.5 Optimised Rotor Control Schedules

Having established a methodology for the generation of Pareto optimal fronts for $\Delta EAOASPL$ and ΔNO_x^{rate} , the onus of designer is to effectively select one or more specific points from the Pareto fronts based on design intent. In this work, three points will be selected, representing: (a) minimum noise emissions, (b) trade-off between noise and NO_x emissions, and (c) minimum NO_x emissions. The Pareto point which corresponds to minimum noise emissions is essentially the point that achieves minimum (most negative) $\Delta EAOASPL$. Similarly, the minimum NO_x point is the Pareto point which achieves minimum (most negative) ΔNO_x^{rate} . The minimum distance selection method (Ref. [158]) is utilised to determine the Knee point of the Pareto front which will serve as trade-off point for the multi-disciplinary optimisation. Specifically, a Utopia-optimal point is determined as the combination of minimum noise and minimum NO_x points. This is a fictitious optimal point which would achieve idealised minimisation of both objectives, concurrently. The Pareto trade-off point is then determined as the point which has the minimum Euclidean distance from the

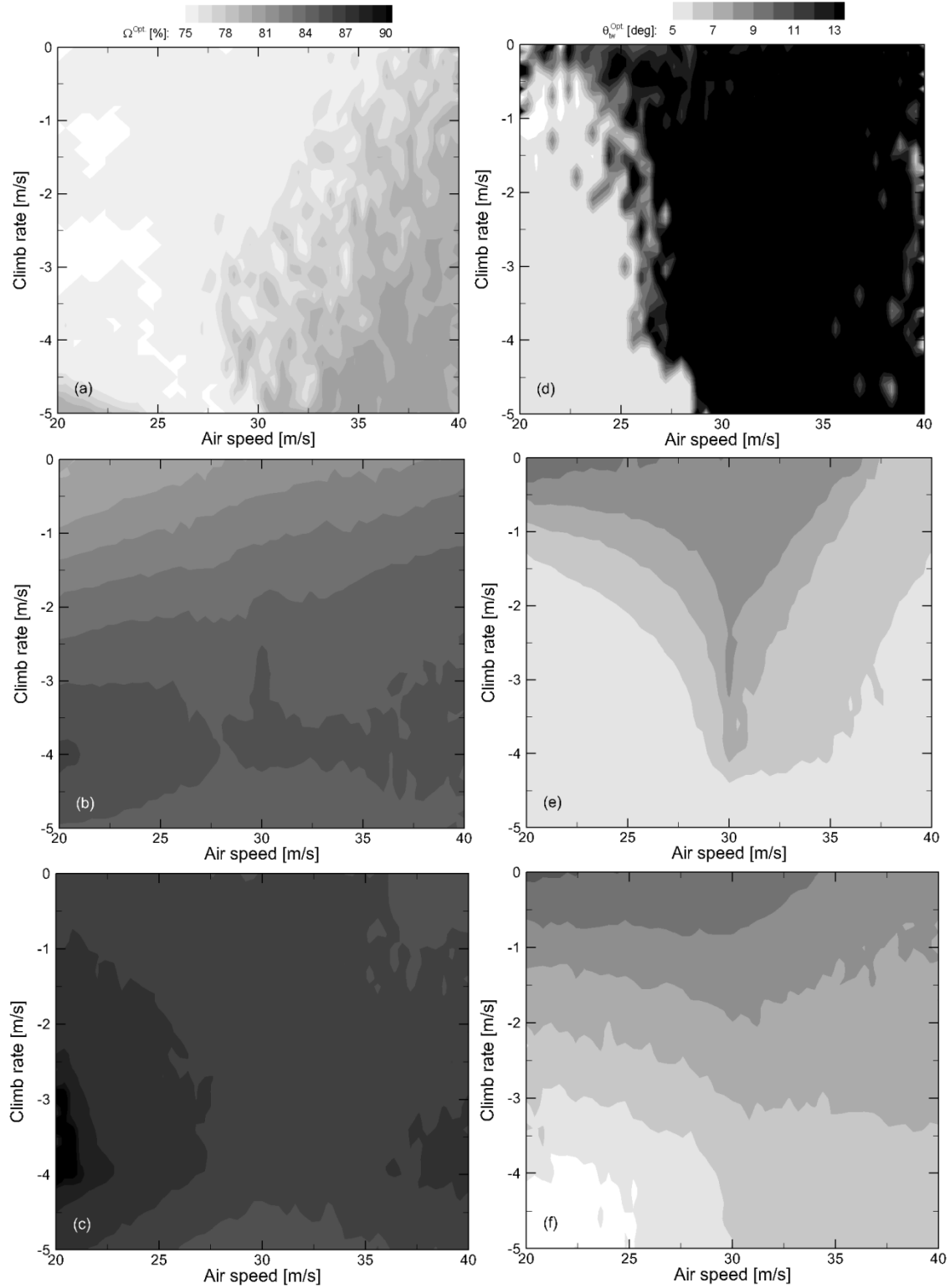


Figure 7.13: Optimised rotor control schedules for $AUM = 2000 \text{ kg}$: Optimal rotor speed for (a) Noise, (b) trade-off, (c) NO_x ; Optimal blade twist angle for (d) Noise, (e) trade-off, (f) NO_x .

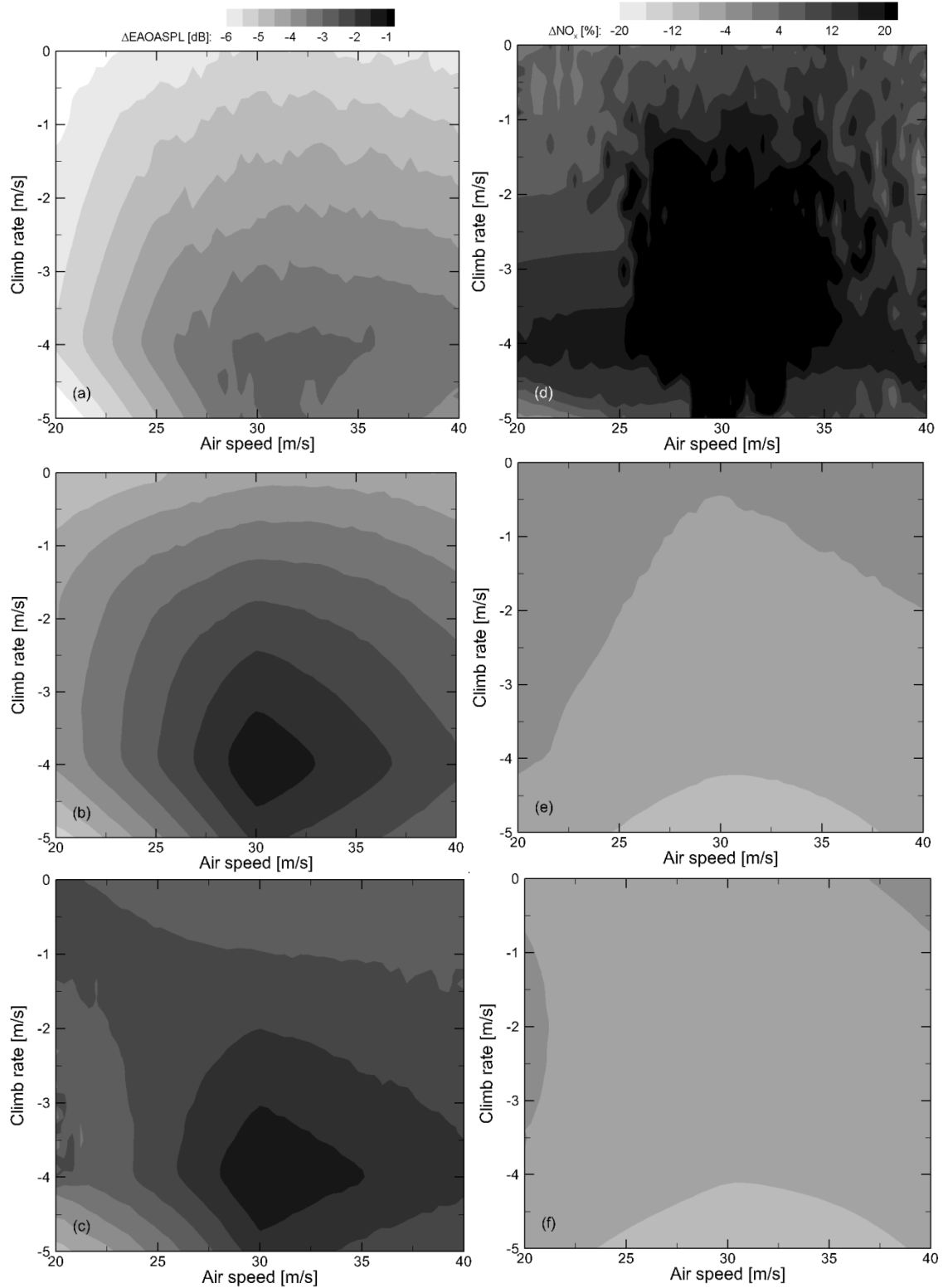


Figure 7.14: Acoustic and environmental benefits for $AUM = 2000 \text{ kg}$: Noise benefits through optimising for (a) Noise, (b) trade-off, (c) NO_x ; NO_x benefits through optimising for (d) Noise, (e) trade-off, (f) NO_x .

Utopia-optimal point. It is noted that for the calculation of Euclidean distance of each Pareto point from the Utopia-optimal point, the front is previously normalised (%) in both dimensions.

In this work, the aforementioned optimal selection strategy has been generalised to export optimised VRS/ABT control schedules for noise minimisation, NO_x minimisation and a trade-off between these two objectives. Fig. 7.11 a)-f) presents the optimised rotor control schedules exported using the optimisation method presented in Section 3.5.2. The schedules refer to $AUM = 1600 \text{ kg}$ at a reference density altitude $h = 250 \text{ m}$. The provided schedules cover the entire horizontal speed (V_x) and climb rate (V_z) range of the descending flight envelope of TEL helicopters. For each and every point of the schedules, a dedicated multi-disciplinary optimisation is carried out, generating a unique Pareto front. Subsequently, the minimum noise, trade-off and minimum NO_x points are derived.

In terms of noise optimisation, it can be seen that the optimal rotor speed Ω^{Opt} is generally lower at low V_x with values below 80% to benefit from the Doppler amplification factor effects. NO_x optimisation requires higher Ω^{Opt} , mitigating adverse effects associated with extremely low rotor dynamic head. As regards, optimal blade twist angle θ_{tw}^{Opt} , generally high blade twist angles (above 11 deg) are preferred to minimise noise in the regions of flight envelope where the aerodynamic behaviour of the main rotor is dominated by tip-vortices. During steep descent though, θ_{tw}^{Opt} is lower than the nominal θ_{tw} to compensate the adverse impact of V_z on blade airloads. In terms of NO_x minimisation, high θ_{tw}^{Opt} is required in the low-speed region where induced power dominates, whilst θ_{tw}^{Opt} values close to the nominal are required in the rest of descent flight envelope. The optimised Ω and θ_{tw} schedules which achieve the best trade-off between noise and NO_x essentially facilitate optimal control patterns from both disciplines.

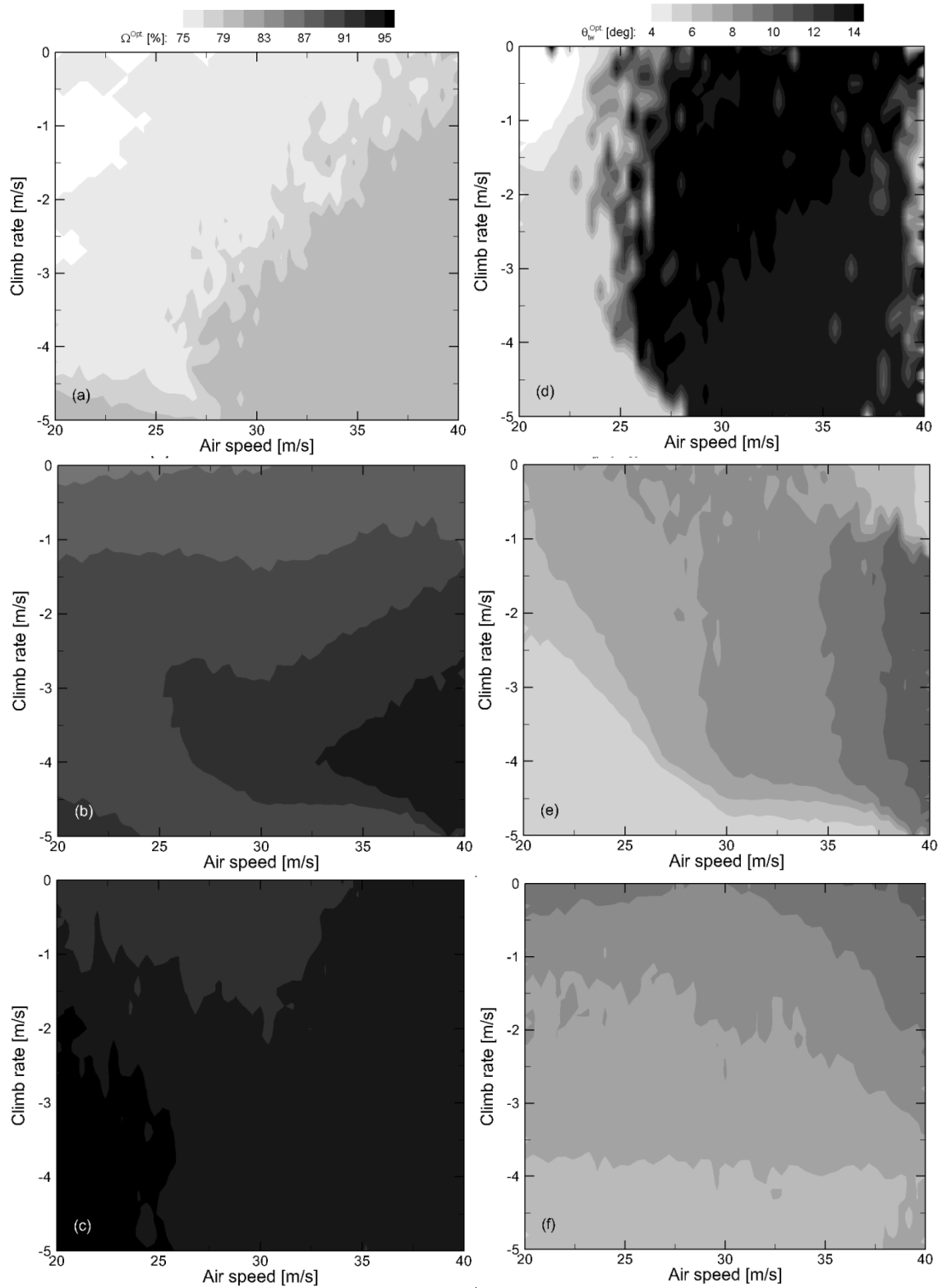


Figure 7.15 Optimised rotor control schedules for $AUM = 2400 \text{ kg}$: Optimal rotor speed for (a) Noise, (b) trade-off, (c) NO_x ; Optimal blade twist angle for (d) Noise, (e) trade-off, (f) NO_x .

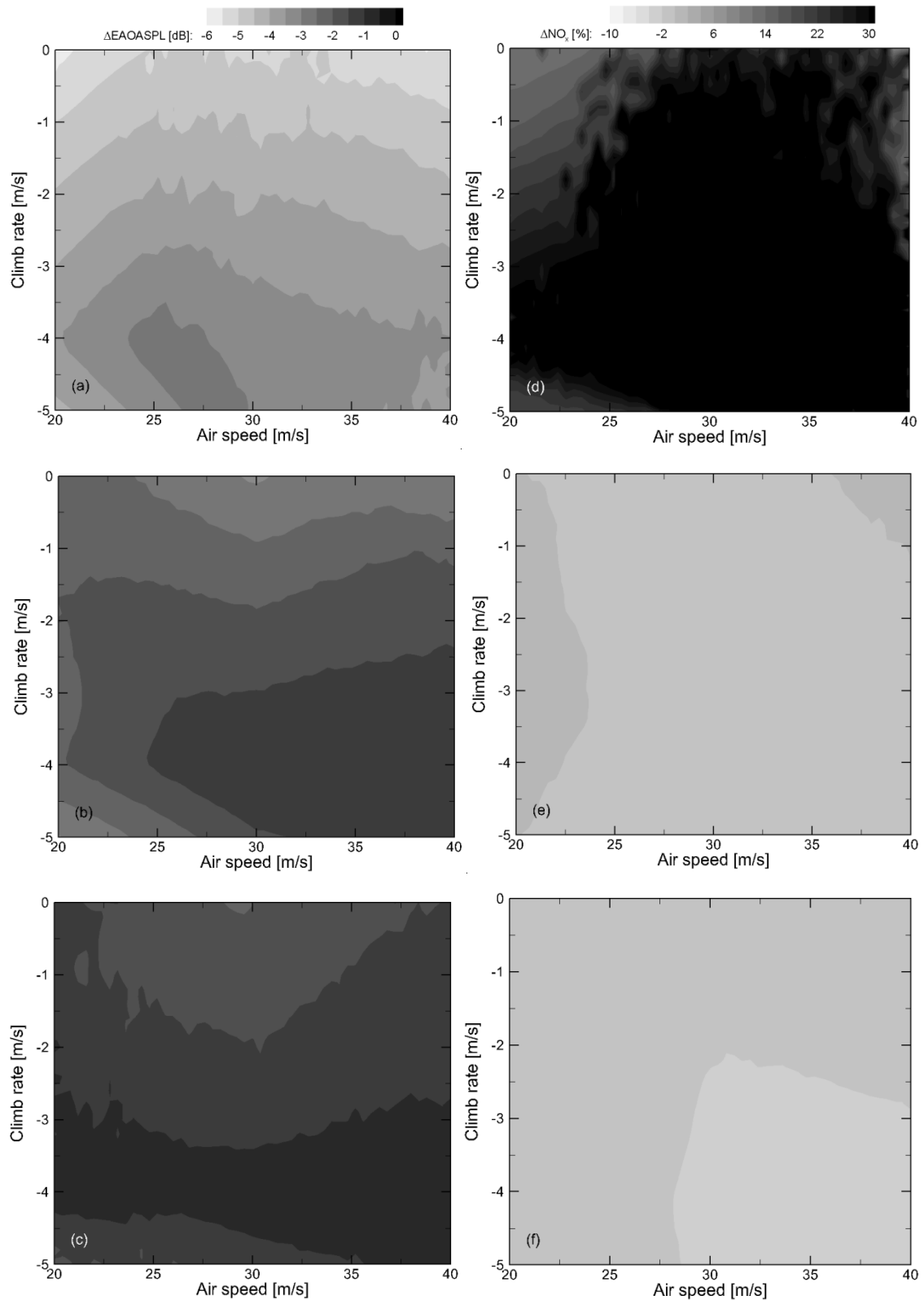


Figure 7.16: Acoustic and environmental benefits for $AUM = 2400 \text{ kg}$: Noise benefits through optimising for (a) Noise, (b) trade-off, (c) NO_x; NO_x benefits through optimising for (d) Noise, (e) trade-off, (f) NO_x.

Fig. 7.12 a)-f) quantifies the $EAOASPL$ and NO_x^{rate} benefits attainable by the derived control schedules at $AUM = 1600 \text{ kg}$. The highest NO_x benefits are expected in lightly-loaded rotor conditions, hence at low V_x and V_z . In terms of $\Delta EAOASPL$, significant improvements are observed in steep descent due to the effectiveness of ABT in counteracting the increase in inboards airloads generated by the strong vertical free-stream velocity component. Additionally, considerable noise reduction is achieved in level flight due to the mitigation of Doppler amplification effects through VRS, combined with reduction of vortex roll-up effects achieved through blade tip off-loading due to ABT. The smallest, but still significant, noise reductions are located in the BVI-dominated region of the flight envelope, around $V_x = 32.5 \text{ m/s}$ and $V_z = -4 \text{ m/s}$, which dictates that the combined VRS/ABT rotor morphing cannot entirely diminish the impact of intensive BVIs. Nevertheless, the proposed concept is capable of achieving overall noise reductions in the range of 2.31 to 8.30 dB, without low harmonic noise penalties reported in alternative rotor morphing concepts as in Ref. [116]. Additionally, the accrued $EAOASPL$ benefits come with concurrent NO_x^{rate} reductions ranging from -6.42% to -21.32% when the trade-off schedules are applied.

Fig. 7.13 a)-f) presents the Ω^{Opt} and θ_{tw}^{Opt} schedules targeting at minimum noise, trade-off and NO_x emissions, respectively, for $AUM = 2000 \text{ kg}$. The observed patterns are similar to the $AUM = 1600 \text{ kg}$ case, albeit suggesting higher Ω^{Opt} and θ_{tw}^{Opt} settings to cater for the increased disc loading of the rotor. The corresponding $EAOASPL$ and NO_x^{rate} benefits for $AUM = 2000 \text{ kg}$ are provided in Fig. 7.14 a)-f). The noise-optimised schedules offer noise reductions up to -6.62 dB, however, this comes at a NO_x increase cost, even up to 60%. The NO_x optimised schedules provide NO_x benefits up to $\Delta NO_x^{rate} = -10.29\%$ along with noise reductions between 1.07 to 4.70 dB. The trade-off schedules, offer noise benefits of up to $\Delta EAOASPL = -5.36 \text{ dB}$ accompanied by environmental benefits of up to $\Delta NO_x^{rate} = -9.91 \%$.

Fig. 7.15 a)-f) presents the Ω^{Opt} and θ_{tw}^{Opt} schedules targeting at minimum noise, trade-off and NO_x emissions, respectively, for $AUM = 2400 \text{ kg}$. Generally higher Ω^{Opt} and θ_{tw}^{Opt} is required compared to the $AUM = 1600 \text{ kg}$ and $AUM = 2000 \text{ kg}$ cases, with Ω^{Opt} reaching values up to 97.03% to minimise NO_x^{rate} in low-speed steep descent. For noise minimisation, θ_{tw}^{Opt} greater than 14 deg. are necessary to mitigate $EAOASPL$ in level-flight conditions, primarily attributed to the impact of strong vortex roll-up. Fig. 7.16 a)-f) reflects the obtained $EAOASPL$ and NO_x^{rate} reductions attainable by the application of proposed schedules. For the trade-off case, it is observed that the accrued improvements are smaller at the highest AUM , with $\Delta EAOASPL$ ranging between -1.10 to -3.08 dB. In terms of NO_x, the highest attainable benefit for the trade-off case is $\Delta NO_x^{rate} = -1.90\%$, with negligible reductions or even slight increases of up to 0.79% at the greatest part of the descent flight envelope.

7.6 Discussion

The potential of concurrent noise and NO_x emissions mitigation for helicopters in descending flight conditions has been investigated. A Design Space Exploration approach has been implemented based on full-factorial Design of Experiments and Kriging-based Response Surface Modelling (RSM). The predictive accuracy of RSMs has been assessed through Leave-One-Out Cross-Validation, obtaining RMS errors of the order of 1%. Having established their sufficient predictive accuracy, the derived RSMs have been utilised towards identifying the physical behaviour of the highly non-linear model. Strong trade-offs have been observed between noise and NO_x across the entire range of descending flight envelope, which dictated the importance of multi-disciplinary optimisation targeting the concurrent minimisation of both objectives.

The NSGA-II genetic algorithm has been employed for the optimisation, providing Pareto-optimal fronts with optimal solutions. A mathematical procedure has been proposed, which apart from noise-optimal and NO_x-optimal

Pareto points, identifies a point which achieves an adequate trade-off between the two examined objectives. The developed approach has been deployed towards exporting generalised and holistic schedules of variable rotor speed and active blade twist that achieve concurrent reduction in both noise and NO_x rate across the full range of descending flight speeds and climb rates and for a series of different all-up-mass options. It was observed that, at low rotorcraft masses, noise reductions between 2.31 and 8.30 dB can be achieved, combined with NO_x rate reductions in the range of 6.42% to 21.32%. These figures are lower at high rotorcraft masses, reaching noise benefits of 1.10-3.08 dB along with NO_x rate deltas of -1.90% to 0.79%.

Summarising, the multi-disciplinary optimisation of combined variable rotor speed and active blade twist concepts is capable of achieving simultaneous mitigation of adverse acoustic and environmental impact of light helicopters in descent.

8 Results and Discussion; Aeroacoustics of Optimised Morphing Helicopter Rotors

8.1 Introduction

This chapter investigates the aeroacoustics of optimised VRS/ABT helicopter rotors across a realistic descent trajectory. A reference descent trajectory is defined based on European Union Aviation Safety Agency (EASA) certification standards. The rotor speed and blade twist values are optimised at each waypoint of the trajectory, targeting the concurrent mitigation of adverse noise and NO_x emissions effects, as shown in Section 3.5. It is noted that during the certification process of a helicopter, the main rotor must operate at nominal rotorspeed. Hence, the optimisation undertaken in this work is not addressed at improving the certification capability of new rotorcraft. However, this flight path serves as a realistic reference to explore the potential of mitigating the adverse noise impact of descending helicopters directly at the source, i.e. the vicinity of the main rotor. This can be considered as a direct contribution to the measures of the “Balanced Approach” (Ref. [6]) proposed by ICAO for the management of aircraft noise around airports.

An in-depth examination of rotor-wake aerodynamics is conducted, showcasing the impact of combined rotor speed and blade-twist optimisation on tip-vortex geometry, wake roll-up and potential BVI locations. The effect of optimised VRS/ABT concept on unsteady airloads and corresponding noise hemispheres is investigated at individual points of the trajectory. A detailed, time-domain analysis of the acoustic signal obtained by BVI-dominated microphones is provided to highlight the effectiveness of the concept in mitigating BVI noise without considerable low-harmonic noise penalties. The cumulative ground noise impact is assessed via the trajectory analysis method presented in Section 3.4. Finally, along with source and ground noise impact benefits,

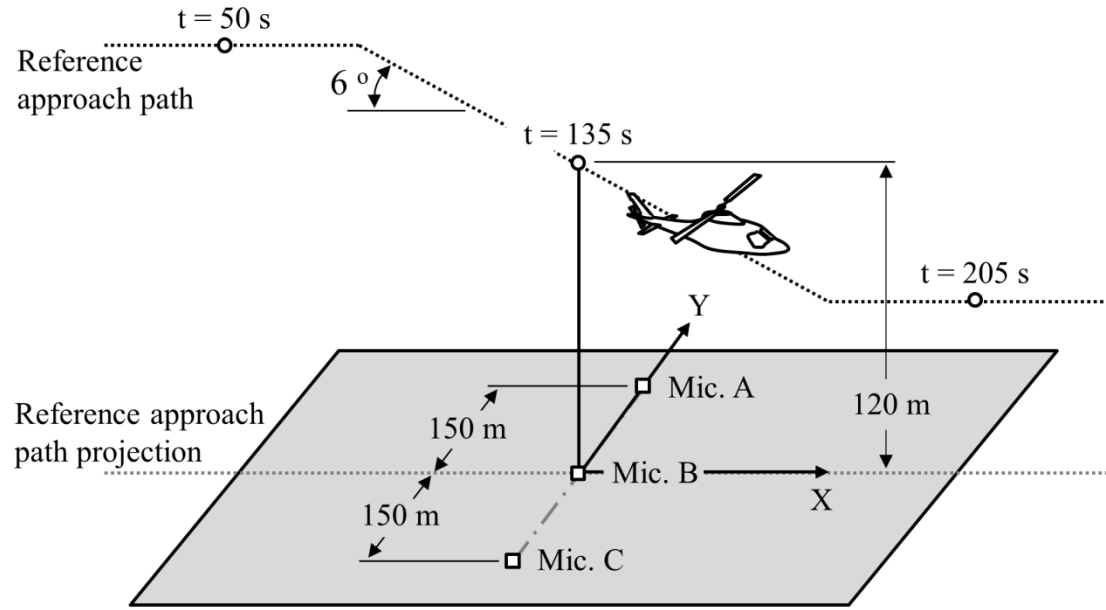


Figure 8.1: Approach path definition, selected control points and microphone locations, reproduced after Ref. [159].

simultaneous NO_x emissions reductions are achieved throughout the derived trajectory.

8.2 Trajectory Definition

The descending flight path for helicopter noise certification approved by the EASA and specified in Annex 16, Volume I of ICAO (Ref. [159]) is replicated for the assessments of this work. A reference flight path angle of -6° is prescribed and three microphones (A, B and C) are located at specific coordinates on the ground level, as depicted in Fig. 8.1. Microphone measurements are taken at the specific time instant when the helicopter flies at altitude $h = 120 \text{ m}$ above Microphone B. Side-line microphones A and C are mounted at lateral distance of 150 m from Microphone B in the port and starboard direction, respectively. According to Ref. [3], the helicopter should be descending at a reference climb rate equal to the certified best rate of descent speed. In this study, the operating conditions are adapted to reflect a BVI-dominated descent. To that end, a horizontal flight speed $V_x = 40 \text{ m/s}$ along

with a climb rate $V_z = -4.2 \text{ m/s}$ are selected for the descending part of the flight path. For continuity of the trajectory, a level flight section with $V_x = 40 \text{ m/s}$ is added prior to the beginning of descent and a low-altitude/low-speed level-flight section with $V_x = 20 \text{ m/s}$ is added after the end of the descending flight segment. Three representative trajectory waypoints at time instances $t = 50 \text{ s}$, $t = 135 \text{ s}$ and $t = 205 \text{ s}$ are selected for analysis of aerodynamic and aeroacoustic phenomena in the vicinity of the rotor, corresponding to each one of the three segments of the trajectory. The resulting time-variations in density altitude, flight speed and flight path angle are provided in Fig. 8.2 a)-c), respectively.

The multi-disciplinary optimisation process described in Section 3.5.2 is applied at each waypoint of the derived trajectory to obtain optimised VRS/ABT settings targeting the concurrent reduction of $EAOASPL$ and NO_x^{rate} . The minimum

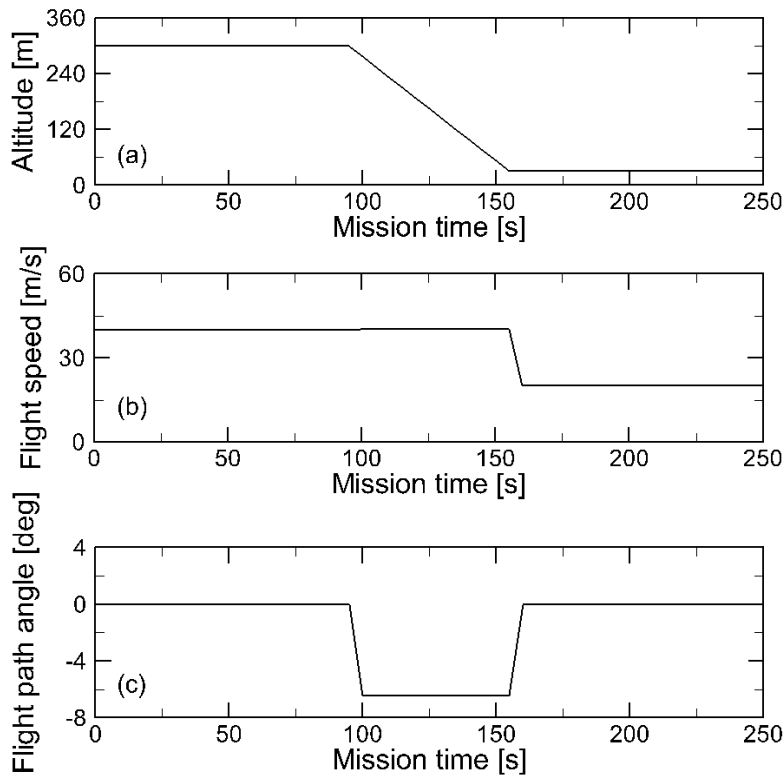


Figure 8.2: Trajectory profile; (a) altitude, (b) flight speed, (c) flight path angle as functions of mission time.

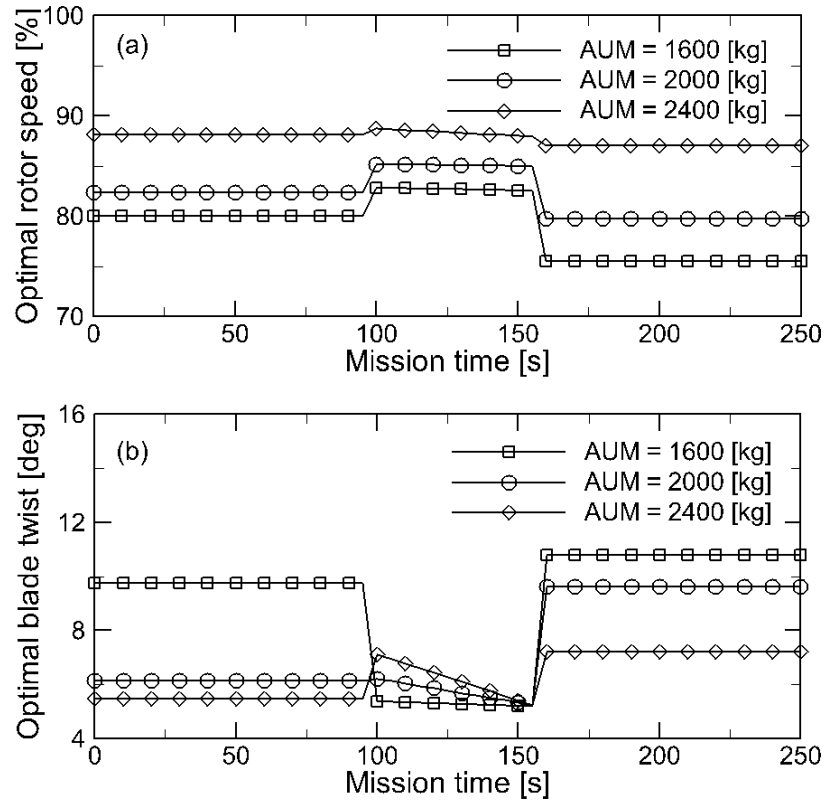


Figure 8.3: Optimised rotor control profiles; (a) rotor speed, (b) blade twist angle as functions of mission time for different all-up-mass (AUM) options.

distance selection method, introduced in Section 7.5 is employed to define the best trade-off point among the full set of Pareto-optimal points of the multi-disciplinary optimisation process. Fig. 8.3 a) illustrates the optimised rotor speed settings across the trajectory for different AUM options. Generally higher Ω^{Opt} is required at higher AUM to compensate for the increased rotor disc loading through increasing rotor dynamic head. The optimum Ω during descent is higher compared to the rest segments. The freestream upwash exerted in descent, yields the formation of a stalled region, a driving region and a driven region (moving radially outwards across the rotor disc). Increase in Ω leads to re-adjustment of radial variation of dynamic head and a (favourable) reduction of near-tip incidence. The reduction of the extent of stalled and driven regions in this case is beneficial for both NO_x and noise.

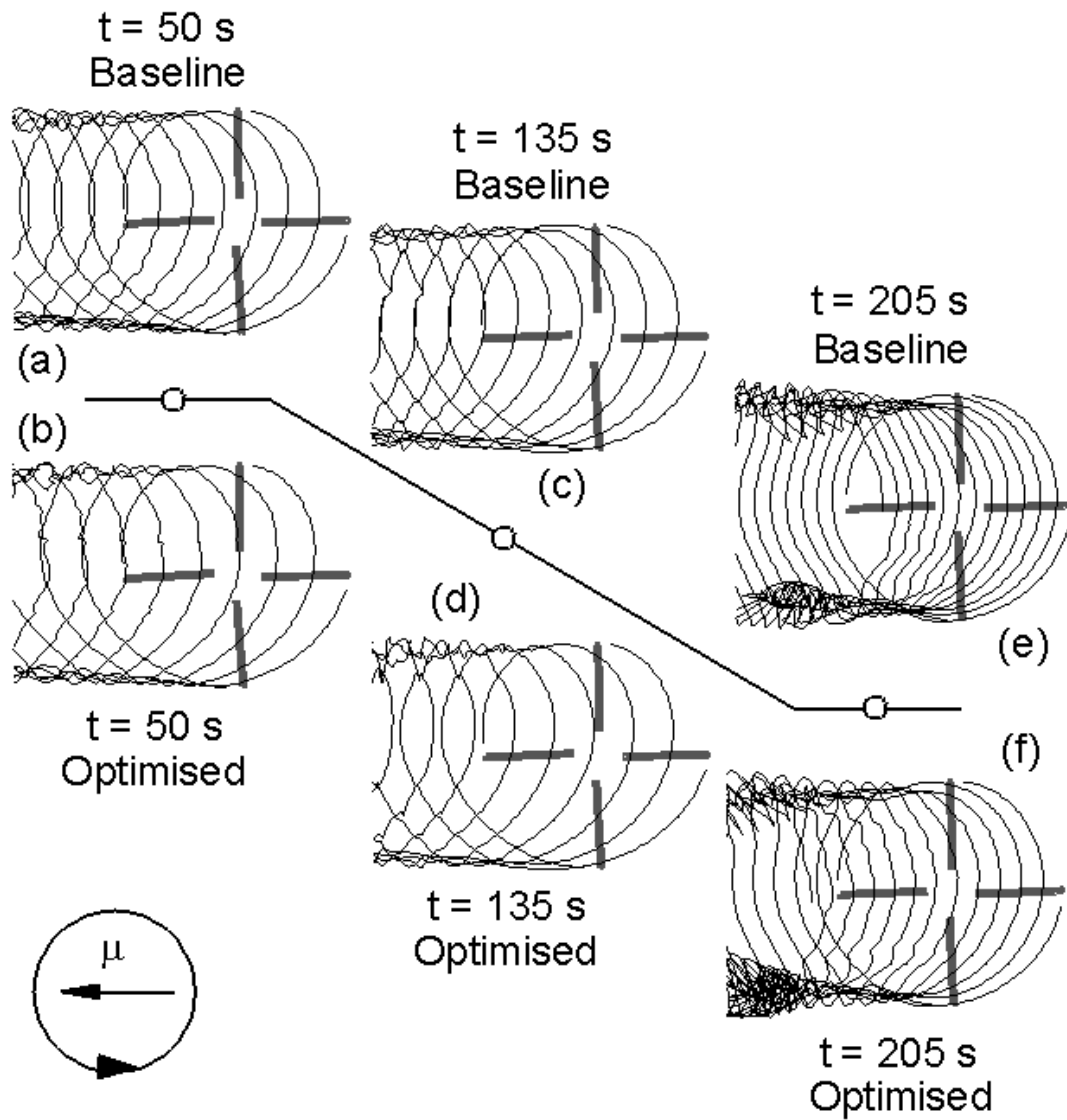


Figure 8.4: Evolution of main rotor wake across the approach trajectory for $AUM = 1600 \text{ kg}$; (a) baseline rotor control at $t = 50 \text{ s}$ with $V_x = 40 \text{ m/s}$ and $V_z = 0 \text{ m/s}$, (b) optimised rotor control at $t = 50 \text{ s}$ with $V_x = 40 \text{ m/s}$ and $V_z = 0 \text{ m/s}$, (c) baseline rotor control at $t = 135 \text{ s}$ with $V_x = 40 \text{ m/s}$ and $V_z = -4.2 \text{ m/s}$, (d) optimised rotor control at $t = 135 \text{ s}$ with $V_x = 40 \text{ m/s}$ and $V_z = -4.2 \text{ m/s}$, (e) baseline rotor control at $t = 205 \text{ s}$ with $V_x = 20 \text{ m/s}$ and $V_z = 0 \text{ m/s}$, (f) optimised rotor control at $t = 205 \text{ s}$ with $V_x = 20 \text{ m/s}$ and $V_z = 0 \text{ m/s}$.

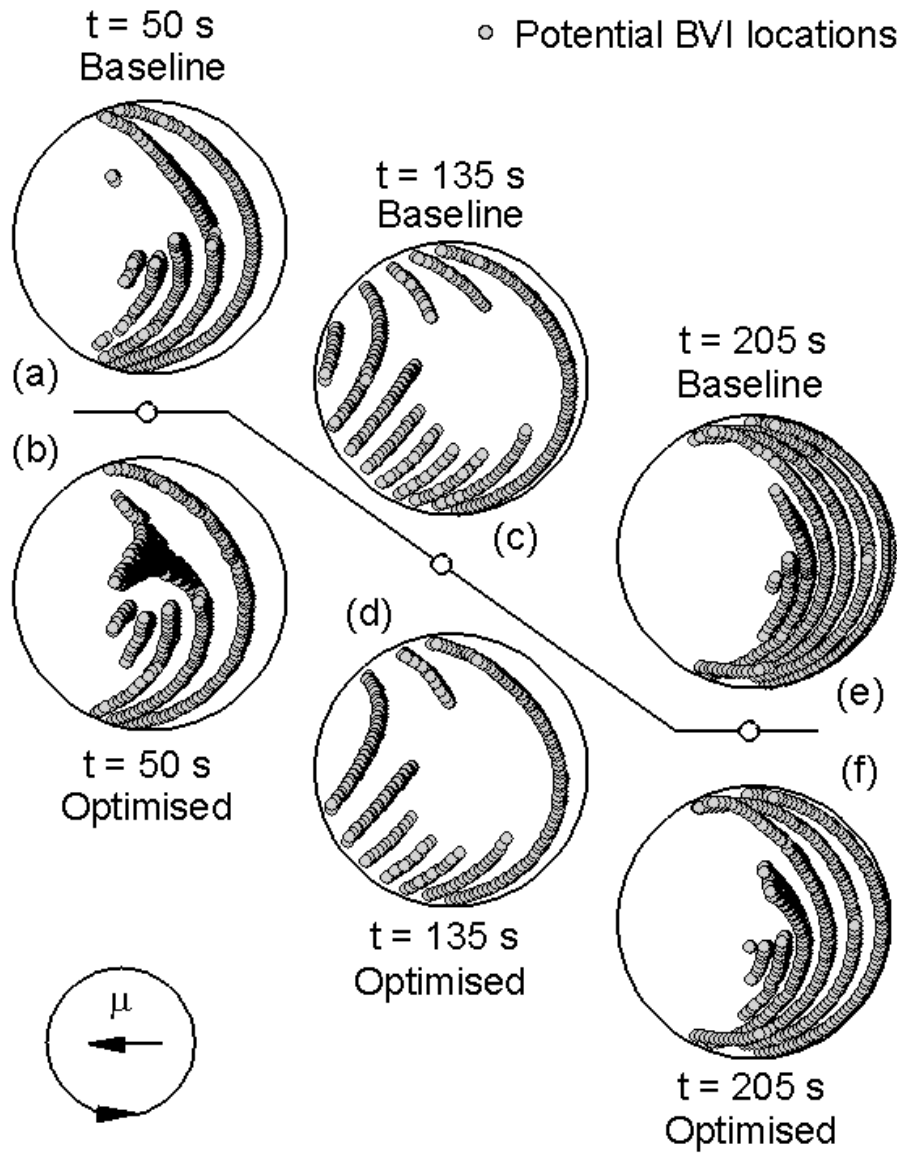


Figure 8.5: Evolution of potential BVI locations on the main rotor disc for $AUM = 1600 \text{ kg}$; (a) baseline rotor control at $t = 50 \text{ s}$ with $V_x = 40 \text{ m/s}$ and $V_z = 0 \text{ m/s}$, (b) optimised rotor control at $t = 50 \text{ s}$ with $V_x = 40 \text{ m/s}$ and $V_z = 0 \text{ m/s}$, (c) baseline rotor control at $t = 135 \text{ s}$ with $V_x = 40 \text{ m/s}$ and $V_z = -4.2 \text{ m/s}$, (d) optimised rotor control at $t = 135 \text{ s}$ with $V_x = 40 \text{ m/s}$ and $V_z = -4.2 \text{ m/s}$, (e) baseline rotor control at $t = 205 \text{ s}$ with $V_x = 20 \text{ m/s}$ and $V_z = 0 \text{ m/s}$, (f) optimised rotor control at $t = 205 \text{ s}$ with $V_x = 20 \text{ m/s}$ and $V_z = 0 \text{ m/s}$.

Fig. 8.3 b) presents the corresponding θ_{tw}^{Opt} settings across the trajectory to achieve the best trade-off between $\Delta EAOASPL$ and ΔNO_x^{rate} . At low AUM , high θ_{tw}^{Opt} is required in level flight, primarily to off-load the blade tips and reduce the impact of tip-vortex circulation on noise generation. At higher AUM though, θ_{tw} lower than the nominal is required in level-flight, to satisfy the generally lower θ_{tw} requirements in terms of NO_x^{rate} reduction, as discussed in Section 7.3. In descent, the considerable increase in upwash is compromised through reduced θ_{tw}^{Opt} , towards obtaining the best trade-off between $EAOASPL$ and NO_x^{rate} . The effect of altitude variation in descent is reflected on the gradual reduction of θ_{tw}^{Opt} as h decreases. This is primarily due to the reduced disc loading at low h which affects both $EAOASPL$ and NO_x^{rate} . In low-speed level flight, considerably high θ_{tw}^{Opt} is required to mitigate the impact of strong tip-vortex roll-up on noise generation.

8.3 Blade-Vortex Interactions

A thorough analysis of the aerodynamic and aeroacoustic effects in the vicinity of the rotor across the derived trajectory can shed light on the complex inter-coupled phenomena associated with VRS/ABT optimisation. The $AUM = 1600 \text{ kg}$ case will be analysed, where higher environmental and acoustic benefits are anticipated, to showcase the best potential of the investigated rotor morphing technology. Fig. 8.4 a) and b) illustrate the tip-vortex geometries for baseline and optimised Ω and θ_{tw} , respectively, at the selected time-instant $t = 50 \text{ s}$. It can be seen that due to reduced Ω , the magnitude of local velocity vector at each vortex filament is decreased which results in less potential intersections between rotor blades and tip-vortices. This has also been discussed by Leishman (Ref. [17]), indicating that advancing-side BVIs can be nearly parallel at specific flight conditions, which is a potential

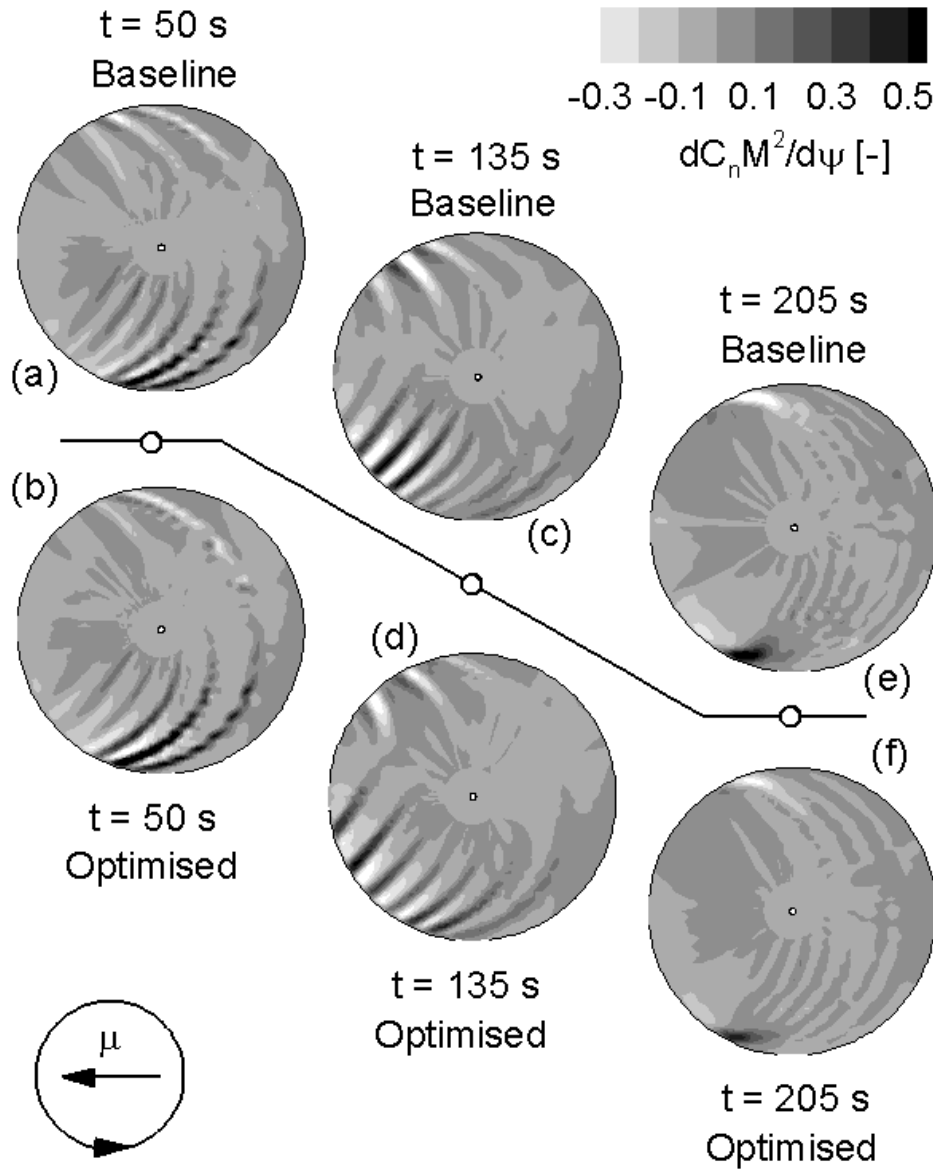


Figure 8.6: Evolution of time-derivative of normal force coefficient for $AUM = 1600 \text{ kg}$; (a) baseline rotor control at $t = 50 \text{ s}$ with $V_x = 40 \text{ m/s}$ and $V_z = 0 \text{ m/s}$, (b) optimised rotor control at $t = 50 \text{ s}$ with $V_x = 40 \text{ m/s}$ and $V_z = 0 \text{ m/s}$, (c) baseline rotor control at $t = 135 \text{ s}$ with $V_x = 40 \text{ m/s}$ and $V_z = -4.2 \text{ m/s}$, (d) optimised rotor control at $t = 135 \text{ s}$ with $V_x = 40 \text{ m/s}$ and $V_z = -4.2 \text{ m/s}$, (e) baseline rotor control at $t = 205 \text{ s}$ with $V_x = 20 \text{ m/s}$ and $V_z = 0 \text{ m/s}$, (f) optimised rotor control at $t = 205 \text{ s}$ with $V_x = 20 \text{ m/s}$ and $V_z = 0 \text{ m/s}$.

source of intensive noise. Similar response is observed in Fig. 8.4 c) and d), where the descending flight conditions at $t = 135\text{ s}$ promote BVI phenomena. It is noted that the temporal shift resulting from reduced Ω essentially leads to reduced frequencies of the BVI events. Low frequency noise is generally less annoying for the human ear; however it is less easily attenuated with distance.

Fig. 8.4 e) and f) present the corresponding tip-vortex topologies at low-speed level-flight at $t = 205\text{ s}$ with baseline and optimised Ω and θ_{tw} , respectively. Tip-vortex roll-up is considerably more pronounced compared to the previous flight states. It can be seen that increased θ_{tw} leads to reduced-strength roll up at the retreating side, primarily due to tip off-loading arising from the radial inboard shift of bound circulation. At the advancing side, reduced strength of the roll-up in the close vicinity of the rotor is observed, although increased concentration of vortex structures is observed in the downstream direction.

The locations of tip-vortices relative to the elastic rotor blades essentially affect the possibility of BVI events across the rotor tip-path-plane. Fig. 8.5 depicts the locus of potential BVI events, calculated using the spherical detection method discussed in Ref. [34]. The illustrated trajectory waypoints correspond to the selected time-instances analysed above. It can be seen that the combined VRS/ABT optimisation leads to reduced number of potential BVI locations at all investigated flight states. In Fig. 8.5 b) it is shown that the mitigation of potential BVI locations in the blade tip region comes at a cost of increased-concentration potential BVI events close to the rotor hub at the retreating side. This is attributed to the impact of high θ_{tw}^{Opt} at this *AUM* option. However, it needs to be noted that the inboard regions of the rotor disc encompass lower magnitudes of radiation Mach number which mitigate the impact of Doppler amplification on noise generation.

Even though the number of interactions may decrease, it is their intensity that determines the potential magnitude and directivity of the associated BVI noise (Ref. [17]). The intensity of BVI aerodynamic events is quantified via the time-derivative of normal force coefficient across the rotor disc, as discussed in

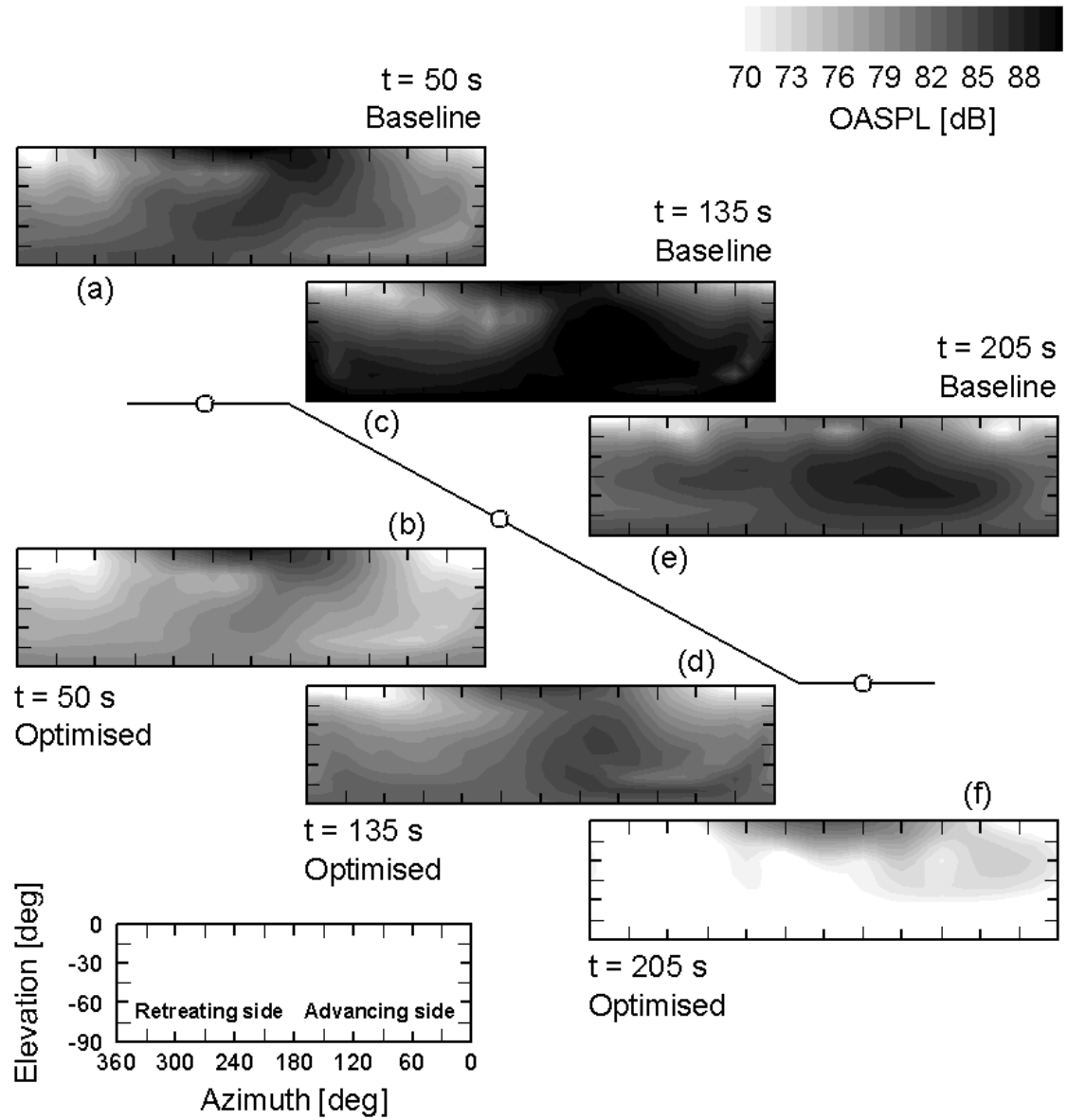


Figure 8.7: Evolution of main rotor noise hemispheres for $AUM = 1600 \text{ kg}$; (a) baseline rotor control at $t = 50 \text{ s}$ with $V_x = 40 \text{ m/s}$ and $V_z = 0 \text{ m/s}$, (b) optimised rotor control at $t = 50 \text{ s}$ with $V_x = 40 \text{ m/s}$ and $V_z = 0 \text{ m/s}$, (c) baseline rotor control at $t = 135 \text{ s}$ with $V_x = 40 \text{ m/s}$ and $V_z = -4.2 \text{ m/s}$, (d) optimised rotor control at $t = 135 \text{ s}$ with $V_x = 40 \text{ m/s}$ and $V_z = -4.2 \text{ m/s}$, (e) baseline rotor control at $t = 205 \text{ s}$ with $V_x = 20 \text{ m/s}$ and $V_z = 0 \text{ m/s}$, (f) optimised rotor control at $t = 205 \text{ s}$ with $V_x = 20 \text{ m/s}$ and $V_z = 0 \text{ m/s}$.

Section 4.3. Figure 8.6 presents the impact of VRS/ABT optimisation on the intensity of unsteady rotor airloads. In Fig. 8.6 b) it can be seen that high value of θ_{tw}^{Opt} , necessary to achieve trim at the reduced Ω^{Opt} , does not allow any significant reductions in the time-derivative of $C_n M^2$. However, it is evident that the frequency of BVI events is reduced after the optimisation, and considerable noise mitigation is expected from reduced Doppler amplification, as well. Similar benefits are observed in Fig. 8.6 d): the most significant reductions of the magnitude of $C_n M^2$ time-derivative can be achieved at the advancing side, although complete alleviation of loading peaks seems not attainable.

Fig 8.6 e) shows the impact of strong vortex roll-up on the formation of high-loading cells in the tip region at both port and starboard directions. These are attributed to the concentration of high-circulation vortical structures in the close vicinity of the blade tips near the aforementioned locations. Fig. 8.6 f) depicts the effectiveness of increased θ_{tw}^{Opt} in off-loading the blade tips and reducing the impact of vortex roll-up on unsteady airloads and consequently, noise.

The distribution of unsteady airloads across the rotor disc is effectively reflected on the corresponding noise hemispheres at the investigated trajectory waypoints. Fig 8.7 b) illustrates the impact of reduced Ω on thickness noise, which has its maximum directivity in the upstream direction in the plane of the rotor. Additionally, it is shown that even without reducing the magnitude of unsteady airloads, considerable loading noise mitigation can be achieved through the reduction of the impact of Doppler amplification on acoustic source terms. Similarly, in strong BVI conditions as in Fig. 8.7 c) and d), the combination of reduced magnitude of radiation Mach number with reduced frequency of intense loading peaks, can lead to considerable reductions of source noise, even at the BVI-dominated regions of the noise hemisphere at $\psi = 90 \text{ deg}$ and $\beta = -45 \text{ deg}$. Finally, the intensive loading noise generated by the high-loading cells at the sides of the rotor in low-speed cruise is efficiently alleviated by the combined VRS/ABT optimisation, as shown in Fig. 8.7 e) and f).

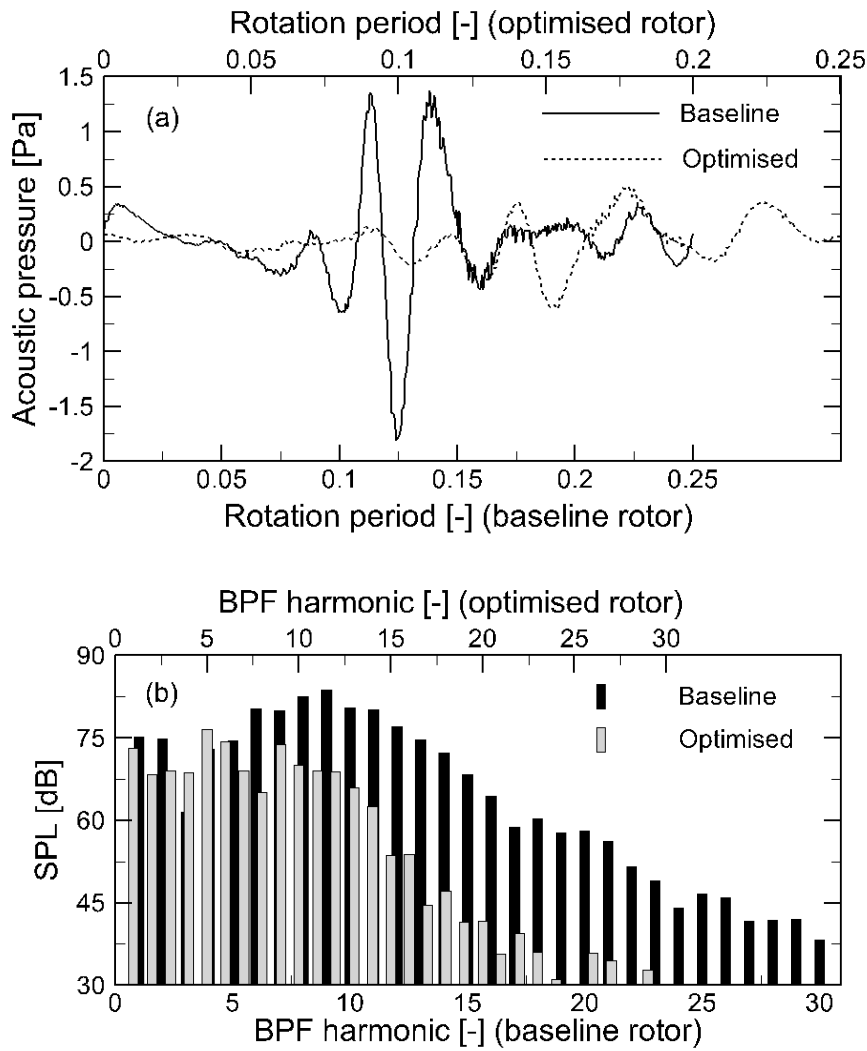


Figure 8.8: Comparison of baseline and optimised noise at microphone C, around $t = 135$ s ; (a) acoustic pressure pulse, (b) spectral analysis.

Fig. 8.8 presents a detailed analysis on the impact of combined VRS/ABT optimisation on mitigating low-harmonic and BVI noise. The analysis refers to the time instant $t = 135$ s when the helicopter is located directly above Microphone B at altitude $h = 120$ m. Fig. 8.8 a) illustrates the acoustic pressure signal at Microphone C, dominated by advancing-side BVI noise, for baseline and optimised rotor VRS/ABT control settings. Since the acoustic signal is periodic with four identical pulses, it is decided to isolate the signal of one pulse only for better clarity. Time axes are both normalised with the duration of a full rotor rotation of each case. The bottom horizontal axis corresponds to the

baseline acoustic signal and the top to the optimised one. Axes scaling corresponds to the relative duration of baseline and optimised pulses. It can be seen that the optimised VRS/ABT control efficiently mitigates the amplitude of BVI-related peaks in the acoustic signal. The corresponding spectral analysis of both signals is represented in a similar manner in Fig. 8.8 b). A clear reduction in the order of 8 dB is calculated for the 6-30 BPF harmonics, whilst the associated low-frequency noise penalty is limited in the order of 1 dB within the first 5 BPF harmonics.

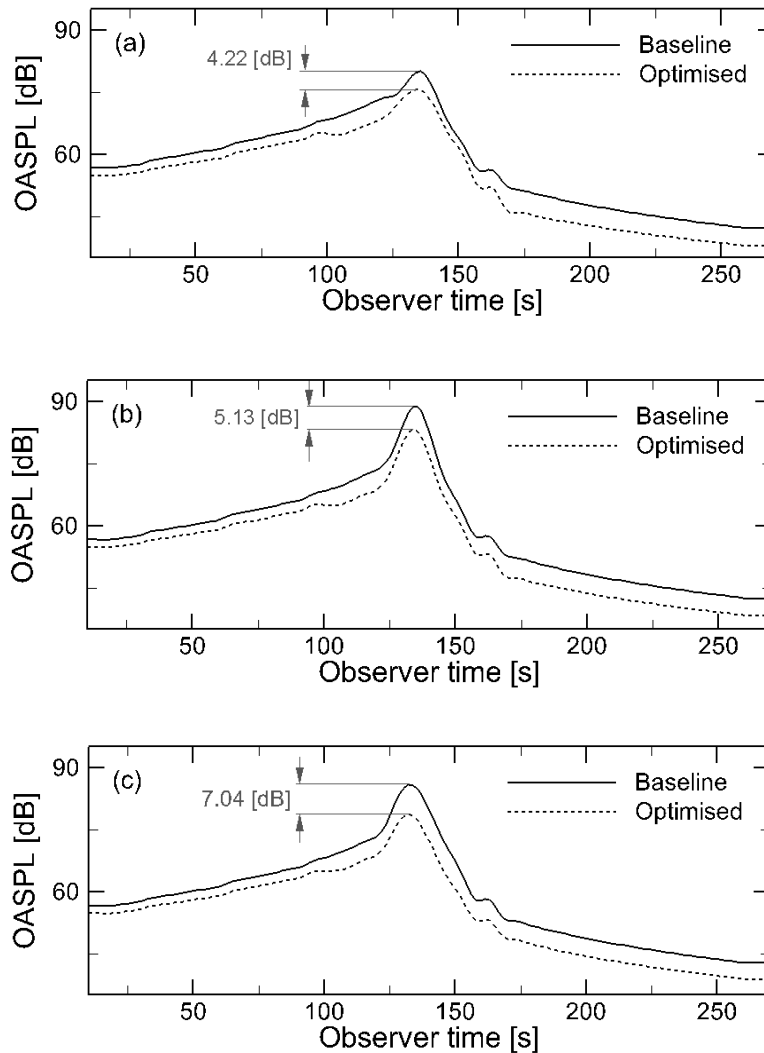


Figure 8.9: Comparison of baseline and optimised rotor control in terms of ground noise at three selected microphones; (a) microphone A, (b) microphone B, (c) microphone C.

8.4 Time-Domain Analysis

Figure 8.9 a)-c) present the full time-history of noise signal at Microphones A-C, respectively. Specifically for Microphone C, it can be seen that the overall noise is 7.04 dB lower when VRS/ABT optimisation is employed compared to the nominal Ω and θ_{tw} settings. Similar improvements are observed in Microphones A and B, albeit with slightly smaller magnitude, which can be attributed to the behaviour of $C_n M^2$ time-derivative, presented in Fig. 8.6 d).

Fig. 8.10 a) and b) illustrate the accrued benefits in terms of $\Delta EAOASPL$ and ΔNO_x^{rate} which are the objectives of the multi-disciplinary optimisation carried out in this work. Generally lower $EAOASPL$ and NO_x^{rate} benefits arise from the

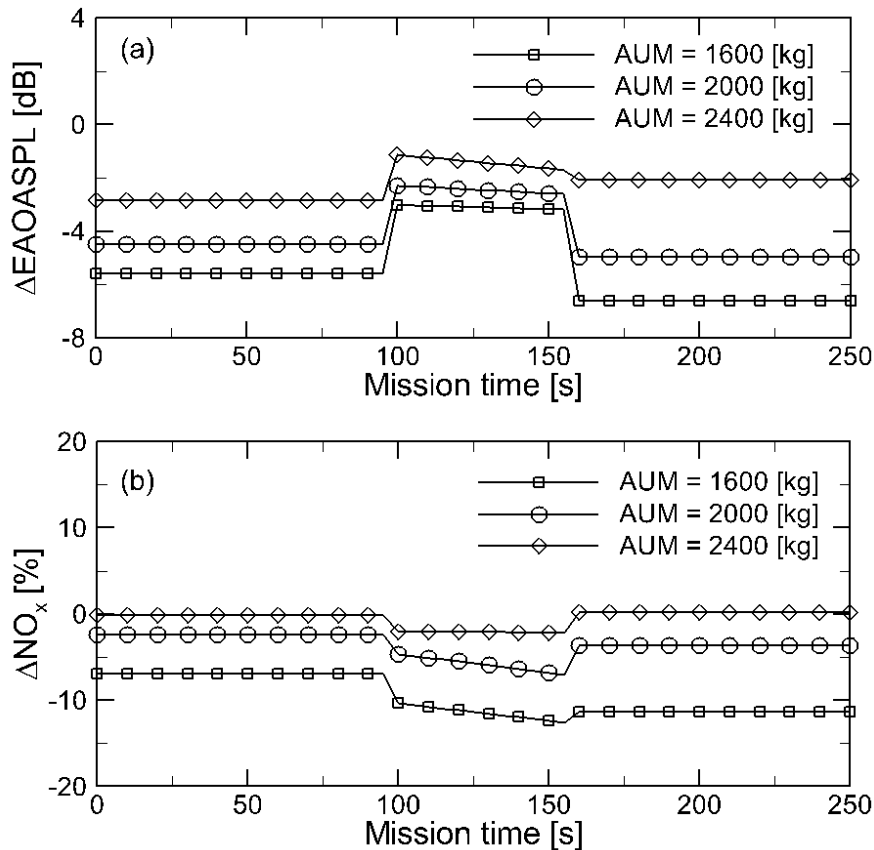


Figure 8.10: Noise and pollutant emissions benefits for three different AUM, arising from optimised rotor control as function of mission time; (a) Noise benefits, (b) NO_x benefits.

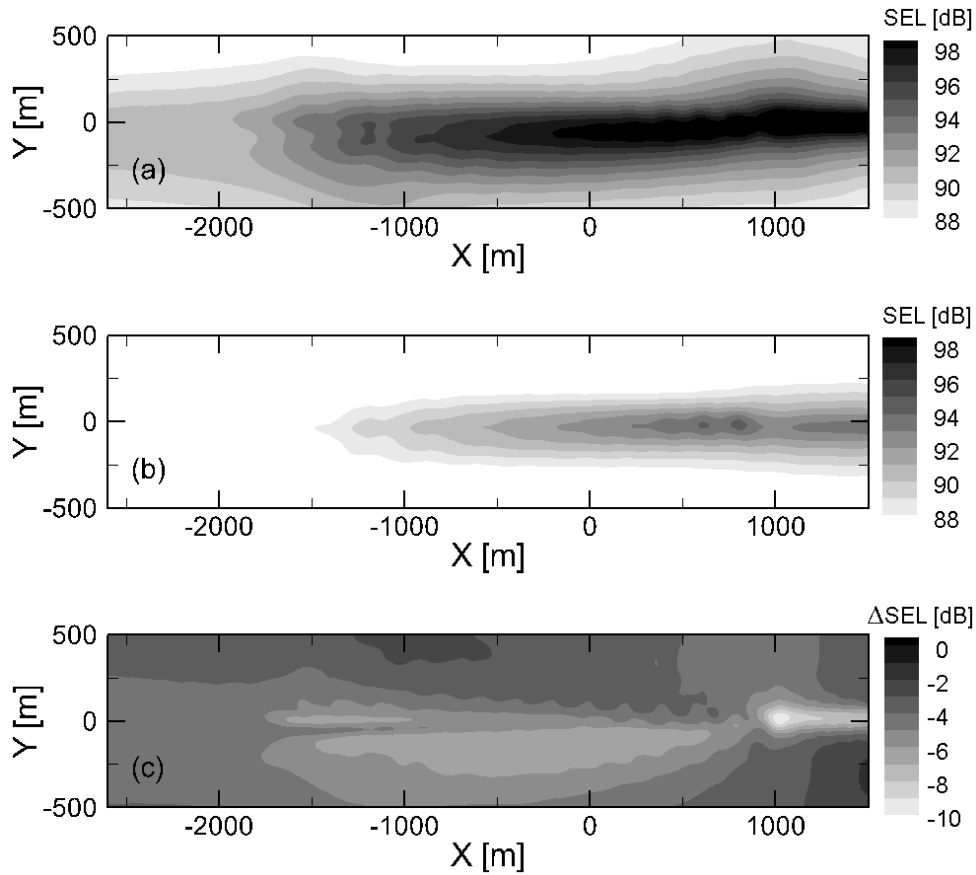


Figure 8.11: Sound Exposure Levels (SEL) of the approach trajectory for $AUM = 1600 \text{ kg}$; (a) baseline, (b) optimised, (c) difference (b)-(a).

optimisation at high AUM . As regards NO_x^{rate} , it has been shown that the combined VRS/ABT concept is less efficient in high disc loading conditions as discussed in Ref. [112]. Due to the multi-disciplinary nature of the undertaken optimisation, the determination of best trade-off point at high AUM conditions is unavoidably favouring NO_x^{rate} reductions to achieve a sufficient compromise with $\Delta EAOASPL$.

Source noise improvements of 6.60 dB are attained during the third segment of the trajectory where ABT effectively mitigates the impact of strong vortex roll-up at the sides of the rotor by off-loading the blade tip-region. The lowest $\Delta EAOASPL$ is achieved in descent, due to the impact of BVI phenomena which cannot be entirely diminished by VRS/ABT. NO_x^{rate} improvements of up to

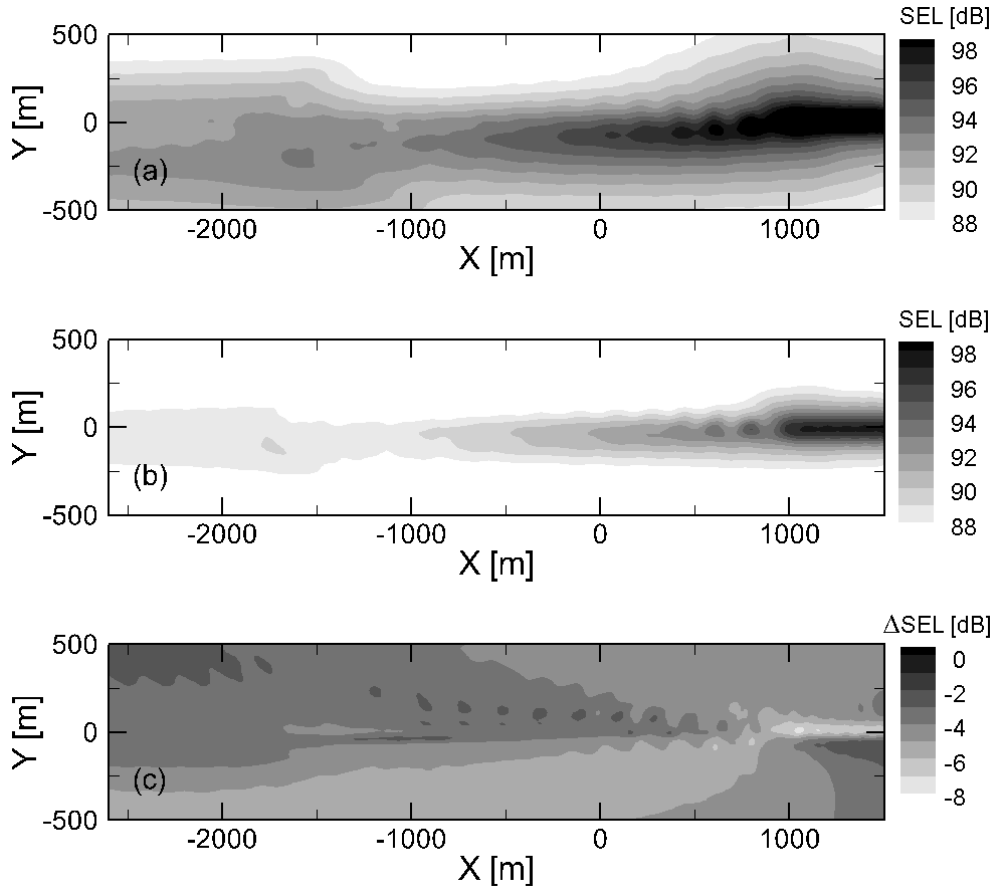


Figure 8.12: Sound Exposure Levels (SEL) of the approach trajectory for $AUM = 2000 \text{ kg}$; (a) baseline, (b) optimised, (c) difference (b)-(a).

11.4% are observed during descent and low speed cruise, which represent the flight states with the minimum disc loading along the derived trajectory. The slight slope observed in the distribution of $\Delta \text{EAOASPL}$ and $\Delta \text{NO}_x^{\text{rate}}$ during descent reflects the impact of altitude variation, as discussed in Section 8.2.

8.5 Ground Noise Impact

The overall impact of optimised VRS/ABT rotor control on ground noise footprint is quantified using the Sound Exposure Level (SEL) metric introduced in Section 6.6. Fig. 8.11 a) and b) illustrate the resulting SEL of the baseline and the optimised VRS/ABT control schedule derived from the multi-disciplinary optimisation. SEL is generally higher towards the end of the trajectory due to

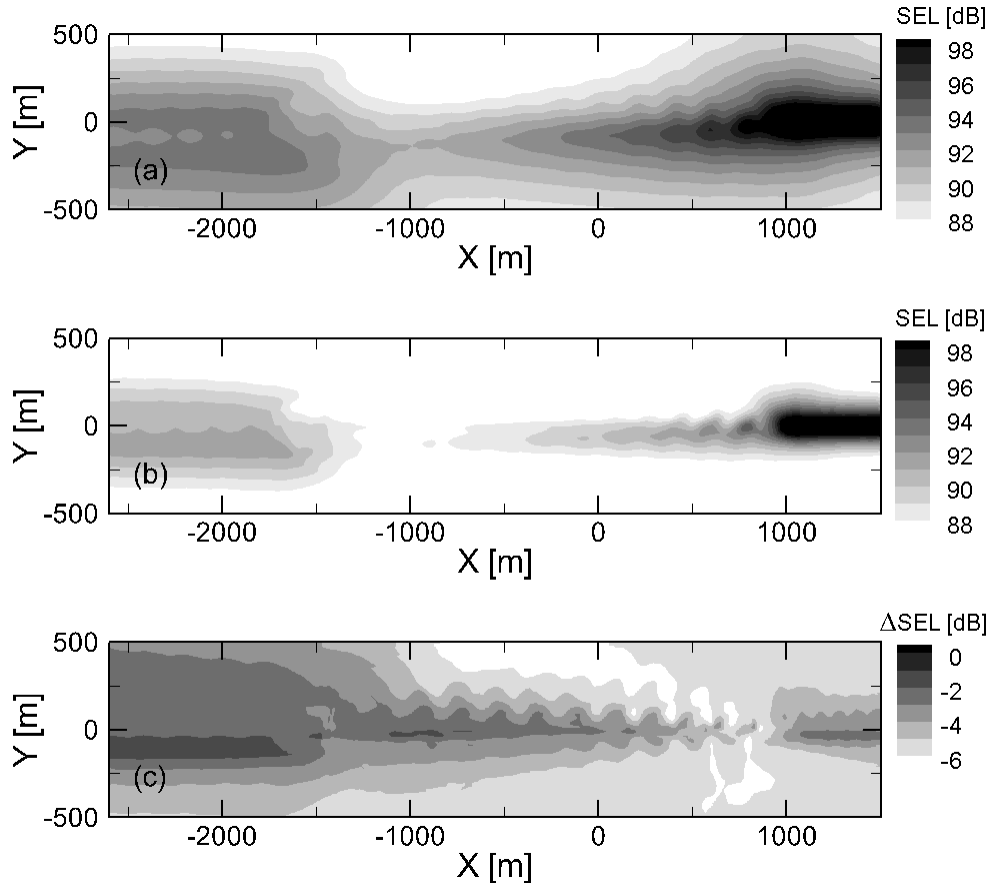


Figure 8.13: Sound Exposure Levels (SEL) of the approach trajectory for $AUM = 2400 \text{ kg}$; (a) baseline, (b) optimised, (c) difference (b)-(a).

the accumulated time exposure of this region in noise emitted by the rotor during the entire trajectory duration. It is evident that SEL is considerably reduced in the optimised case. Fig. 8.11 c) quantifies the spatial distribution of noise deltas across the investigated ground region. SEL improvements up to 10.74 dB are calculated at $AUM = 1600 \text{ kg}$ conditions. Fig. 8.12 a)-c) depicts the corresponding SEL and accrued noise benefits for $AUM = 2000 \text{ kg}$. Similar trends are observed, although with noise reductions reaching up to 8.06 dB. Finally, as shown in Fig. 8.13 c), noise benefits range between 1.14 to 6.70 dB for $AUM = 2400 \text{ kg}$. This is attributed to the reduced effectiveness of ABT towards mitigating the adverse aerodynamic effects of strong vortex roll-up encompassed by the highly loaded rotor disc during the third section of the derived trajectory.

8.6 Discussion

This chapter investigated the effectiveness of optimised VRS/ABT helicopter rotor morphing in noise abatement descent procedures, targeting both noise and NO_x reductions across a designated trajectory. Emphasis has been given to aerodynamic and aeroacoustic phenomena associated with the optimisation of active rotor control. A realistic descent trajectory has been defined based on EASA noise certification standards with appropriately placed ground microphones.

The impact of VRS/ABT optimisation on main rotor wake geometry was explored. Reduced rotor speed effectively leads to a temporal shift of wake vortex filaments, reducing the possibility of BVIs in moderate speed cruising or descending flight. Increased blade twist off-loads the blade tips by re-distributing the peak circulation towards the blade inboard region, consequently mitigating the adverse effects of strong tip-vortex roll-up in low-speed cruise. It has been shown that the first-order effects of the investigated concept are associated with the mitigation of the impact of Doppler amplification on source noise terms, whilst second-order effects are associated with reductions in the time-derivative of unsteady airloads. Overall, the acoustic source terms are considerably affected as shown in the corresponding noise hemispheres across the trajectory: both low-frequency thickness noise in moderate-speed cruise and low-frequency loading noise in low-speed cruise are effectively reduced. Additionally, high-frequency aerodynamic loading terms are also mitigated at the BVI-dominated regions of the noise hemispheres in descent.

It has been shown that source noise reductions of up to 6.6 dB can be achieved with concurrent NO_x benefits of up to 11%. Additionally, the time-domain analysis of the acoustic signal of a BVI-dominated ground microphone has dictated that BVI noise reductions up to 8 dB can be achieved, with only 1 dB low-harmonic noise penalty. Overall, SEL ground noise reductions of approximately up to 10 dB, 8 dB and 6 dB can be achieved for lightly, medium and highly loaded helicopters, respectively.

9 Summary and Conclusions

9.1 Context Overview

This chapter summarises the major findings obtained from this research work to assess the level of success in terms of addressing the objectives set to achieve the overall scope. Furthermore, the contribution to knowledge is discussed and suggestions for future work are provided.

The scope of this work essentially comprises two fundamental requirements identified in Chapter 1. Specifically:

- Generic and robust methodology for modelling helicopter aeroacoustics, able to resolve individual design variations and applicable on three-dimensional mission simulation.
- Multi-disciplinary optimisation of novel morphing helicopter rotor technologies targeting the simultaneous mitigation of environmental and acoustic impact at both aircraft and operational level.

The first requirement constitutes the development of ICARUS, a tool with proven robustness, which can effectively upgrade the capabilities of rotorcraft research community in terms of quantifying the acoustic impact of rotorcraft at fully-integrated operational level, and incorporating novel modelling approaches aiming to improve the state-of-the-art. The second requirement targets at the demonstration of the applicability of the developed approach towards maximising the potential impact of a novel helicopter concept. The obtained results effectively add contribution towards achieving the challenging environmental targets set for a sustainable rotorcraft aviation.

9.2 Achievement of Set Objectives

A set of five individual objectives has been defined in Chapter 1 to address the scope of this work. Each objective essentially constitutes a specific research topic related to rotorcraft aeroacoustics. Chapter 2 has provided a critical review

of the state-of-the-art with respect to each research topic. Suitable engineering approaches have been selected and associated gaps in the knowledge have been identified. Chapter 3 has described the individual methods developed or employed towards accomplishing the individual objectives. Each objective represents a specific work-package, analysed in Chapters 4-8. Hence, the quantification of success in achieving the overall scope will be split among the five set objectives. Therefore, for each of the defined objectives, the approach followed as well as conclusive evaluation of the accomplishments is provided.

9.2.1 Helicopter rotor source noise prediction

Objective: *Development and validation of an analytical procedure for the evaluation of helicopter rotor aeroacoustics resolving rotorcraft flight dynamics, blade design, three-dimensional kinematics of elastic rotor blades and unsteady rotor aerodynamic loading.*

Approach: A physics-based method for helicopter rotor noise prediction in the time domain has been developed. An existing free-wake aeroelastic rotor model has been employed to predict unsteady rotor airloads and elastic blade motion. A rotor noise prediction methodology has been developed, implementing mathematical formulations for the integral solution of FW-H equation. Formulations for the prediction of thickness and loading noise (including BVI) in the time-domain have been employed. A semi-empirical model for the rapid estimation of broadband noise contribution has been utilised. Compact expressions of the acoustic formulations have been employed to reduce computational time and enable direct coupling with the lifting-line-based approach adopted in the free-wake aerodynamics model.

Conclusions: An extensive validation study has been carried out to assess the predictive accuracy of the developed method. The experimental results of HART II campaign have been used as benchmark, trimmed at acoustically complex operating conditions representative of low-speed descent where BVI noise dominates. It has been shown that the developed method is capable of predicting acoustic pressure histories at BVI-dominated microphones with accuracy of approximately 6% relative to measured data for the first 30 BPF

harmonics and only within a fraction of a second. In terms of ground noise, the average deviation between predicted and measured data has been between 3 and 4 dB, a value comparable to the 2 dB scatter reported in the experimental noise results.

9.2.2 Uncertainty quantification in rotorcraft aeroacoustics

Objective: *Implementation of a computationally-efficient uncertainty quantification method to assess the robustness of the developed noise prediction approach and to provide fundamental modelling guidelines for non-deterministic aeroacoustic simulation in the absence of reliable experimental data to calibrate the wake models.*

Approach: A computationally efficient method for non-deterministic simulation has been deployed for the quantification of uncertainty in aeroacoustics due to wake tip-vortex core modelling uncertainty. The method has been implemented through non-intrusive Polynomial Chaos expansion based on least-squares regression. A sensitivity analysis method utilising Sobol indices has been employed to quantify the impact of each source on the overall system uncertainty. Two distinct cases have been analysed: (a) the case where experimental data regarding the geometry of tip-vortex core are available, and (b) the case of not available experimental data for calibration of tip-vortex core models. Additionally, a holistic investigation has been conducted to assess the impact of tip-vortex uncertainty on aeroacoustic simulation.

Conclusions: It was shown that in the case of availability of experimental data, the levels of simulation uncertainty are in the order of 1 dB, a value which is lower than the experimental uncertainty of 2 dB. This depicts the suitability of free-wake/FW-H approaches as robust tools for modelling helicopter BVI noise. When vortex-core-related experimental data are absent, output uncertainty in the order of 3.5 dB can occur for conservative levels of input uncertainty. This highlights the necessity of non-deterministic simulation when correlations with tip-vortex experimental data are not feasible. The inflow-related parameters, and especially the Squire parameter, have shown to the greatest influence on BVI airloads and noise uncertainty. A holistic uncertainty map has been

developed, which can serve as modelling guidelines for non-deterministic aeroacoustic analysis. Specifically, for given requirements in terms of predictive robustness, the analyst can quantify the anticipated deviation between deterministic and non-deterministic simulation, determine the associated levels of uncertainty and also evaluate the necessity of non-deterministic simulation depending on output objectives and analysis requirements.

9.2.3 Helicopter noise prediction at mission level

Objective: *Introduction of a novel, mission simulation approach based on the integration of existing flight dynamics and aerodynamics models with the developed aeroacoustics approach, capable of simulating realistic three-dimensional helicopter operations.*

Approach: An integrated numerical procedure has been established for the prediction of rotor acoustic impact across complete three-dimensional rotorcraft operations. The flight dynamics code HECTOR has been coupled with the free-wake airloads model and the developed aeroacoustic model to predict rotor source noise on-the-fly, without the need of pre-stored acoustic data. A far-field noise propagation module has been developed to account for atmospheric absorption and spreading losses. A trajectory-adaptive ground observer grid has been introduced to reduce computational requirements and enable mission-level simulation capability. A helicopter model, representative of the twin-engine light Airbus Helicopters Bo 105, has been used for the mission assessments. The applicability of the method has been demonstrated a passenger transport and a search and rescue mission.

Conclusions: The on-the-fly acoustic sphere generation method, which employs computationally efficient mission flight dynamics coupled with high-resolution aero-elastic free wake computations, has been the enabler for the mission-level application of noise prediction. When combined with trajectory-adaptive ground grids, reductions of up to 94.59% have been calculated in the total of employed ground observers, relative to uniform grid configurations for a long range mission. The devised methodology has been successful in handling long-range and long-duration rotorcraft missions. The impact of fuel burn across

the mission has been proven crucial for the long-duration missions. A 9.26% reduction in the value of main rotor thrust coefficient has been observed between the beginning and the ending of the mission, which corresponds to differences of up to ± 5 dB on the source loading noise radiation.

9.2.4 Optimisation of novel morphing rotor helicopter rotors

Objective: *Incorporation of the developed noise prediction method into an elaborate multi-disciplinary optimisation approach, undertaken at aircraft level, targeting the concurrent reduction of pollutant and noise emissions of novel morphing-rotor helicopter configurations.*

Approach: A method for the multi-disciplinary optimisation of novel morphing rotor helicopters has been developed. The novel rotor incorporates combined variable rotor speed and active blade twist. A design space exploration strategy has been implemented based on full-factorial sampling to identify the interaction between the multiple noise generation mechanisms across the complete investigated design space. The system's response has been approximated through Kriging-based interpolation. A multi-objective optimisation strategy based on the NSGA-II genetic algorithm has been employed for the identification of optimal solutions for minimum noise and NO_x at any given operating condition of the descent flight envelope. The optimal solutions have been expressed as Pareto fronts, which allow the direct investigation of the trade-off between noise and NO_x emissions of helicopters in descent.

Conclusions: The predictive accuracy of RSMs has been assessed through Leave-One-Out Cross-Validation, obtaining RMS errors of the order of 1%. Strong trade-offs have been observed between noise and NO_x across the entire range of descending flight envelope, which dictates the importance of multi-disciplinary optimisation targeting the concurrent minimisation of both objectives. The developed optimisation method has been deployed towards exporting generalised and holistic schedules of variable rotor speed and active blade twist that achieve concurrent reduction in both noise and NO_x rate across the full range of descending flight speeds and climb rates and for a series of different all-up-mass options. It has been shown that, at low rotorcraft masses,

noise between 2.31 and 8.30 dB can be achieved, combined with NO_x rate reductions in the range of 6.42% to 21.32%. At high rotorcraft masses, noise benefits of 1.10-3.08 dB have been achieved along with NO_x rate deltas between -1.90% and 0.79%.

9.2.5 Aeroacoustic analysis of optimised morphing helicopter rotors

Objective: *Assessment of the aerodynamics and aeroacoustics of optimised morphing helicopter rotors at operational level, with respect to realistic trajectories in the vicinity of airports.*

Approach: A reference descent trajectory has been defined based on EASA certification standards. The rotor speed and blade twist values have been optimised at each waypoint of the trajectory, using the multi-disciplinary optimisation method developed for the concurrent mitigation of adverse noise and NO_x emissions effects. An in-depth investigation of rotor aerodynamics and aeroacoustics across the trajectory has been carried out employing the integrated noise prediction method developed in this work. The impact of combined rotor speed and blade-twist optimisation on tip-vortex geometry, wake roll-up and potential BVI locations has been assessed. The effect of optimised VRS/ABT concept on unsteady airloads and corresponding noise hemispheres has been investigated at individual points of the trajectory. A detailed, time-domain analysis of the acoustic signal obtained by BVI-dominated microphones has been provided to highlight the effectiveness of the concept in mitigating BVI noise. The cumulative ground noise impact has been quantified via the trajectory analysis method developed in this work.

Conclusions: It has been observed that reduced rotor speed effectively leads to a temporal shift of wake vortex filaments, reducing the possibility of BVIs in intermediate speed cruising or descending flight. Increased blade twist off-loads the blade tips by re-distributing the peak circulation towards the blade inboard direction, consequently mitigating the adverse effects of strong tip-vortex roll-up in low-speed cruise. It has been found that the first-order effects of the investigated concept are associated with the mitigation of the impact of Doppler amplification of source noise terms, whilst second-order effects are associated

with reductions in the time-derivative of unsteady airloads. Overall, the acoustic source terms are considerably affected as showed in the corresponding noise hemispheres across the trajectory. The time-domain analysis of the acoustic signal of a BVI-dominated ground microphone has dictated that BVI noise reductions up to 8 dB can be achieved, with only a 1 dB low-harmonic noise penalty. Overall, SEL ground noise reductions of approximately up to 10 dB, 8 dB and 6 dB have been achieved for lightly, medium and highly loaded helicopter, respectively. The multi-disciplinary optimisation presented in this work, has been capable of achieving simultaneous NO_x reductions of approximately up to 11% in “low and light” descent conditions.

9.3 Contribution to Knowledge

It can be concluded that the objectives set to achieve the overall scope of this work have been thoroughly addressed. The successful accomplishment of defined objectives has primarily been achieved through the development of “ICARUS” (Integral Computational AeRoacoustics Unified Software) for advanced rotorcraft aeroacoustic analysis. ICARUS has been delivered in a fully productionised layout to the Centre for Propulsion Engineering of Cranfield University. It is currently utilised in a number of MSc and PhD research projects across the Centre, therefore developing new knowledge and continuously contributing to the field of rotorcraft aeroacoustics. Parts of the outcomes obtained from ICARUS simulations have found place in Research Proposals related to rotorcraft research, submitted by Cranfield University.

The scientific contribution arising from the achievement of defined objectives should be assessed individually for each objective. This will provide a desensitised metric of novelty and significance of the accrued outcomes towards (a) improving the state-of-the-art in helicopter rotor aeroacoustic simulation and (b) contributing to the knowledge regarding methods and strategies to achieve the demanding environmental goals for sustainable rotorcraft aviation. Specifically:

- A generic methodology, comprising a series of fundamental mathematical formulations for the prediction of helicopter rotor noise in the time domain

has been developed. The extensively validated methodology has been offered in a form that is readily available for application on any helicopter rotor model and at any designated flight conditions. Additionally, the method is compatible with any blade aerodynamics theory applicable in the time domain.

- A computationally efficient strategy for uncertainty quantification in helicopter aeroacoustic simulation has been demonstrated, applicable to complex BVI noise investigations. Insight has been provided on the robustness of free-wake/FW-H methods for BVI noise prediction, under tip-vortex core modelling input uncertainty. Furthermore, a set of fundamental modelling guidelines has been provided, which can be directly used by analysts to pre-estimate the anticipated uncertainty in aeroacoustic simulation based on the robustness of available input data. These guidelines pave the way for the transition from traditional deterministic approaches to non-deterministic simulation of rotorcraft aeroacoustics in the absence of robust experimental input data for tip-vortex core model calibration.
- A novel approach to helicopter mission simulation has been introduced, incorporating trajectory-adaptive ground observer grids coupled with on-the-fly prediction of rotor source noise. The method essentially alleviates the need of pre-stored acoustic data and enables the physics-based prediction of helicopter noise resolving the unique flight dynamics-aerodynamics-aeroelasticity-aeroacoustics of each mission flight state. The impact of fuel burn on rotor source noise generation has been demonstrated, therefore improving the state-of-the art in treating long-range rotorcraft operations. The generic method is applicable on any designated three-dimensional mission scenario.
- A generalised schedule of optimised variable rotor speed and active blade twist control has been developed. The schedule offers simultaneous

reduction of helicopter pollutant and acoustic emissions across a wide range of BVI-dominated operating conditions. The obtained outcomes essentially constitute new findings referring to technologies for sustainable rotorcraft aviation and reduced environmental impact.

- New insight has been provided into the aerodynamics and aeroacoustics of optimised variable rotor speed/active blade twist helicopter rotors, with respect to a realistic EASA-standard descent trajectory. This has been the first investigation of its kind with respect to the novel concept of combined variable rotor speed and active blade. The obtained results significantly contribute towards identifying methods for noise reduction at source. This is effectively the first measure of the Balanced Approach for noise management in the vicinity of airports, proposed by ICAO.

Furthermore, scientific contribution has arisen from the Technology Evaluator activities of the Clean Sky Green Rotorcraft (GRC) Integrated Technology Demonstrator (ITD) European Research Programme. Specifically, the impact of novel helicopter technologies proposed by Clean Sky has been quantified in terms of fuel burn, pollutant and noise emissions at mission, operational and air-transport system (ATS) level. This can be considered as direct contribution to knowledge regarding methods and strategies to achieve the FlightPath 2050 environmental goals set by ACARE.

The contribution to knowledge achieved through the scientific outcomes obtained during the period of this PhD research, has been disseminated in twelve publications, specifically: eight international conference papers, one published high-impact journal article and three more articles to be submitted to high-impact scientific journals.

9.4 Future Work

The developed method constitutes a generic and modular tool for rotor noise prediction. Further validation studies, involving different helicopter configurations would demonstrate the generality of the developed toolkit and

increase the level of trust to the capabilities of the software. Alternative noise metrics could also be tested such as Perceived Noise Levels (PNL) or perception-weighted metrics.

ICARUS could effectively be employed in a fast-rotorcraft comprehensive simulation framework to predict noise emitted by tiltrotor aircraft and compound rotorcraft with minimal modifications to the source code. This will require specific adaptations to the free-wake aerodynamics code in order to facilitate rotor-rotor and rotor-airframe aerodynamic interactions.

The deployed source noise prediction mathematical formulations fundamentally model thickness and loading noise in the time domain using compact expressions of Farassat's Formulation 1A. Recently-developed dual-compact thickness noise formulations would improve thickness noise predictive accuracy, especially for near-field observers located in the plane of the rotor.

The employed non-deterministic simulation approach using Polynomial Chaos expansion could be incorporated into the developed mission simulation numerical procedure for non-deterministic noise prediction at operational level. Principally, this could be initiated for the implemented EASA descent trajectory, targeting at quantifying the uncertainty associated to helicopter virtual certification.

The promising results obtained from the multi-disciplinary optimisation of variable rotor speed/active blade twist rotors in terms of pollutant emissions and acoustic impact could be enriched by adding vibrations in the list of considered objectives. This could be implemented via expanding the existing response surface models and repeating the multi-disciplinary optimisation process.

The transition to robust and multi-disciplinary optimisation of novel rotorcraft technologies at aircraft or mission level could gradually be accomplished through coupling of all the methods developed in this work. Ultimately, optimised novel rotorcraft operations could be obtained, achieving simultaneous mitigation of adverse environmental, acoustic and operational effects, whilst

concurrently accounting for the inherent uncertainty associated with rotorcraft simulation and operation.

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Appendix A: Author's Publications

This appendix includes references to all the journal articles and conference publications produced in the period of this PhD research.

Journal Articles

- Vouros S., Goulos I. and Pachidis V., 2019, "Integrated Methodology for the Prediction of Helicopter Rotor Noise at Mission Level", *Aerospace Science and Technology*, Vol. 89, June 2019, pp. 136-149, ELSEVIER.
- Vouros S., Goulos I. Scullion C., Nalianda D. and Pachidis V., 2019, "Impact of Tip-Vortex Modelling Uncertainty on Helicopter Rotor BVI Noise Prediction", *Article to be submitted to the Journal of the American Helicopter Society*.
- Polyzos N., Vouros S., Goulos I. and Pachidis V., 2019, "Multi-Disciplinary Optimization of Variable Rotor Speed/Active Blade Twist Helicopter Rotors", *Article to be submitted to Aerospace Science and Technology*.
- Vouros S., Polyzos N., Goulos I. and Pachidis V., 2019, "Impact of Optimized Variable Rotor Speed/Active Blade Twist Control on Helicopter Blade-Vortex Interaction Noise", *Article to be submitted to the Journal of Fluids and Structures*.

Conference Proceedings

- Vouros S., Chasoglou A., Efstathiadis T. and Kalfas A., 2016, "Effects of Rotor-Speed-Ratio and Crosswind Inlet Distortion on Off-Design Performance of Contra-Rotating Propelling Unit", ASME Turbo Expo 2016, Seoul, South Korea. *Cycle Innovations Committee Best Paper Award*.
- Stevens J., Rademaker E., Scullion C., Vouros S., Nalianda D., Pachidis V., van Oosten N., Misté G. and Benini E., 2019, "Design Evaluation and

Performance Assessment of Rotorcraft Technology by 2050", European Rotorcraft Forum 2019, Warsaw, Poland.

- Scullion C., Vouros S., Nalianda D., Pachidis V., Junior A., Misté G., Benini E., Stevens J., Rademaker E. and van Oosten N., 2019, "Environmental and Mobility Benefits of Coaxial Compound Rotorcraft Configurations", European Aviation & Space Network International Conference 2019, Athens, Greece.
- Nalianda D., Vouros S., Scullion C., Pachidis V., Junior A., Misté G., Benini E., Stevens J., Rademaker E. and van Oosten N., "Design Evaluation and Performance Assessment of Fast Rotorcraft Concepts in the 2050 Timeframe", Aerospace Europe Conference 2020, Bordeaux, France.
- Vouros S., Goulos I., Scullion C., Nalianda D. and Pachidis V., 2019, "Impact of Aerodynamic Modelling Uncertainty on Helicopter Rotor Aeroacoustic Analysis", European Rotorcraft Forum 2019, Warsaw, Poland.
- Cappuzzo F., Broca O., Roumeliotis I., Vouros S. and Scullion C., 2020, "Conceptual Design of a Parallel Hybrid-Electric Version of the ATR 42-500: From System Architecture to System Simulation", Abstract accepted by ASME Turbo Expo 2020, London, UK.
- Roumeliotis I., Arena F., Yize L., Vouros S., Pachidis V., Unlu D., Toure, Broca O., 2020, "Performance and Emission Assessment of Thermo-Electric Power Plant for Rotorcraft Propulsion", Abstract accepted by ASME Turbo Expo 2020, London, UK.
- Scullion C., Vouros S., Goulos I., Nalianda D. and Pachidis V., 2020, "Optimal Control of a Compound Rotorcraft for Engine Performance Enhancement". Abstract accepted by ASME Turbo Expo 2020, London, UK.