AEROCOUSTIC MEASUREMENTS
OF AIRFRAME COMPONENTS
IN AN OPEN-JET, A HARD-WALL AND A HYBRID WIND TUNNEL TEST SECTION

MARTINUS P.J. SANDERS
AEROACOUSTIC MEASUREMENTS OF AIRFRAME COMPONENTS:
IN AN OPEN-JET, A HARD-WALL AND A HYBRID WIND TUNNEL TEST SECTION

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AEROACOUSTIC MEASUREMENTS OF AIRFRAME COMPONENTS:
IN AN OPEN-JET, A HARD-WALL AND A HYBRID WIND TUNNEL TEST SECTION

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Abstract

Aircraft noise is a major source of environmental noise that negatively affects human health. Stakeholders and policymakers are nowadays increasingly confronted with protests from residents exposed to high aircraft noise levels, forcing them to change their noise policies. A solution to the aircraft noise problem is the further development and validation of more accurate noise prediction tools and noise reduction technologies. These tools and technologies are crucial to aircraft manufacturers because they help reduce uncertainties arising during the design process of a new aircraft. Wind tunnel measurements play an essential role in the development of noise prediction tools and noise reduction technologies because they provide valuable physics-based data in a controlled environment in a cost-efficient way. This data is then be used to develop or validate new (semi-)analytical noise prediction models or extend existing semi-empirical noise prediction models for new or even disruptive aircraft designs. Moreover, computational aeroacoustic (CAA) simulation tools still rely heavily on wind tunnel data for validation.

Aeroacoustic measurements in wind tunnels nowadays commonly use the microphone phased array technique for the localization and quantification of sound sources. However, this type of measurement is subject to considerable uncertainty. This is mainly due to a lack of: (1) an understanding of how the wind tunnel affects the aeroacoustic measurements, (2) a general framework to correct aeroacoustic measurement to a standard free-field condition and (3) an identification and reduction of systematic errors. The work in this thesis aims at understanding and reducing the uncertainty in wind tunnel testing, thereby facilitating the development of more accurate noise prediction tools and noise reduction technologies.

A literature review of the theoretical principles and basic concepts of aerodynamic noise is first presented, and airframe noise prediction models are described. Aeroacoustic measurements in wind tunnels are discussed, and a review of comparability studies found
in the literature is given. The aeroacoustic facility and experimental methods used are described comprehensively. A significant part of the experimental setups and methods used were newly developed as part of this PhD project and are therefore discussed in detail. An open-jet, a hard-wall, and a hybrid (Kevlar-walled) test section were designed and are described. A detailed description of the airfoil models and developed instrumentation is also given. The microphone phased array technique and beamforming algorithms are discussed, and a benchmark validation is presented with data from an array benchmark database. Finally, a general framework to correct single microphone and microphone array data to a standard free-field condition is presented.

A study is presented comparing trailing edge noise measurements in an open-jet, a hard wall, and a hybrid wind tunnel test section. This type of comparative study, involving three different test section configurations, is unique in the current literature. For this study, a NACA-0012, a NACA-0018, a NACA-63018, and a DU97W300 airfoil model were used. The DU97W300 airfoil model validated basic 2D aerodynamic corrections in each test section configuration. The NACA-0012, NACA-0018, and NACA-63018 were used to perform acoustic measurements at low angles of attack. Sound source statistics were measured with unsteady wall pressure measurements near the trailing edge of the airfoils. It was found that the unsteady wall pressure statistics show a negligible difference between the different test section types. Absolute noise levels were measured and compared using microphone phased arrays in a frequency range between 500 Hz and 5 kHz. The averaged maximum deviation of integrated far-field noise levels was ±3 dB, with differences typically lower. The hard-wall test section results show a more significant variation at 500 Hz, which is likely caused by a significantly degraded performance of the microphone phased array used in this test section configuration. Relative noise levels were obtained by measuring the noise reduction from trailing-edge serrations, a noise reduction device. The averaged maximum deviation did not exceed 3 dB, with differences typically yielding comparable noise levels within ±1-3 dB. These results show that the experimental hardware and acoustic corrections framework used will yield comparable noise levels for microphone phased array measurements in different wind tunnel configurations.

The effect of sweep angle on slat noise characteristics from a common high-lift research model, the 30P30N, was investigated. Aeroacoustic measurements were performed at a 0° and 30° sweep angle. The broadband noise component, low-frequency tonal peaks, and high-frequency vortex shedding hump of the far-field noise were unaffected by the
sweep angle. These results showed that aeroacoustic measurements of an unswept high-lift model yield the same noise characteristics of a wing with a practical (30°) sweep angle.

The comparability of slat noise measurements from a common high-lift research model in an open-jet, a hard-wall, and a hybrid wind tunnel test sections was lastly investigated. PIV measurements were performed to capture the 2D time-averaged velocity field in the slat cove flow. The PIV flow measurements showed that the mean flow characteristics in the slat cove were comparable when a similar aerodynamic condition was set based on the $C_p$ distribution of the model. Microphone phased array measurements were performed to compare far-field noise levels in a wide frequency range from 1 kHz to 25 kHz. The implementation of a coherence loss model for correction of the open-jet microphone phased array measurements is presented. The far-field noise spectra were found to be most comparable between the hard-wall and hybrid test section measurements. The open-jet far-field noise generally showed higher noise levels, especially at lower angles of attack with a maximum difference of 5 dB. In addition, the high-frequency vortex shedding hump in the far-field spectra showed different behavior in the open-jet test section suggesting that the physical flow mechanism may be altered. This study showed that beamforming maps and integrated ranges for slat noise measurements are comparable, provided that the geometric angles of attack are adjusted to fit the mean flow in the slat cove region.

In conclusion, the results of these studies contribute to a better understanding of the uncertainties of aeroacoustic measurements for airframe noise in a small-scale facility. The developed framework for benchmarking, correcting, and presenting microphone phased array measurements shown can be used to reduce systematic measurement errors. In addition, the work helps to better select a test section configuration for a specific measurement campaign. Finally, the measurement data will be made available in an open database for benchmark purposes, with measurements performed in other facilities. In particular, the NACA-63018 and 30P30N datasets are available for benchmarking in the Hybrid Anechoic Wind Tunnel Workshop. In addition, the 30P30N data can be used in the Benchmark for Airframe Noise Computations Workshop to validate numerical simulations.
Samenvatting

Vliegtuiggeluid is een belangrijke bron van omgevingsgeluid en heeft een negatieve invloed op de menselijke gezondheid. Beleidsmakers en belanghebbenden worden geconfronteerd met toenemend protest van omwonenden waardoor ze genoodzaakt worden hun beleid aan te passen. Een oplossing voor het vliegtuiggeluid probleem is de ontwikkeling en validatie van nauwkeurigere instrumenten voor geluidsvoorspelling en geluidsreductietechnologieën. Deze instrumenten en technologieën zijn cruciaal voor vliegtuigfabrikanten omdat ze helpen onzekerheden te verminderen in het ontwerpproces van een nieuw vliegtuig. Windtunnelmetingen spelen daarbij een essentiële rol, omdat ze op een kostenefficiënte manier waardevolle, op fysica gebaseerde, data opleveren in een gecontroleerde omgeving. Deze windtunnelgegevens kunnen vervolgens worden gebruikt om bijvoorbeeld nieuwe semi-analytische voorspellingsmodellen voor vliegtuiggeluid te creëren of om bestaande semi-empirische voorspellingsmodellen uit te breiden voor huidige of vernieuwende vliegtuigontwerpen. Daarnaast zijn numerieke aeroakoestische (CAA) simulaties nog steeds sterk afhankelijk van validatie met windtunnelgegevens.

Bij aeroakoestische metingen in windtunnels wordt tegenwoordig vaak gebruik gemaakt van een fasegestuurde microfoonopstelling voor het lokaliseren en kwantificeren van geluidsbronnen. Dit type meting is echter onderhevig aan grote onzekerheid. Dit komt voornamelijk door een gebrek aan: (1) inzicht in hoe de windtunnel de aero-akoestische metingen beïnvloedt, (2) een algemeen kader om aeroakoestische metingen te corrigeren naar een standaard vrije-veldconditie en (3) een identificatie en reductie van systematische fouten. Het werk in dit proefschrift is gericht op het begrijpen en verminderen van de onzekerheid in aeroakoestische metingen in windtunnels. Hiermee wordt uiteindelijk de ontwikkeling van voorspellingsinstrumenten en geluidsreducerende technologieën vereenvoudigd en nauwkeuriger.

Er wordt een studie gepresenteerd waarin metingen van het geluid van een enkel vleugelprofiel worden vergeleken in een open straal, een gesloten en een hybride windtunnel testsectie. Dit type vergelijking, met drie verschillende windtunnel testsecties is uniek in de huidige literatuur. Voor deze studie werden een NACA-0012, een NACA-0018, een NACA-63018 en een DU97W300 vleugelmodel gebruikt. Het DU97W300 model validerde basis 2D aerodynamische correcties in elke testsectieconfiguratie. De NACA-0012, NACA-0018 en NACA-63018 werden gebruikt om akoestische metingen uit te voeren bij lage aerodynamische invalshoeken. Geluidsbronstatistieken werden gemeten met fluctuerende drukmetingen nabij de achterrand van de vleugelprofielen. De statistieken van de fluctuerende wanddruk laten een verwaarloosbaar verschil zien tussen de resultaten verkregen in de verschillende testsecties. Absolute geluidsniveaus werden gemeten en vergeleken met behulp van een fasegestuurde microfoonopstelling in een frequentiebereik tussen 500 Hz en 5 kHz. De gemiddelde maximale afwijking van geïntegreerde vrije-veldgeluidsniveaus was ±3 dB, waarbij de verschillen doorgaans lager waren. De resultaten van de gesloten testsectie laten een significante variatie zien bij 500 Hz. Dit wordt waarschijnlijk veroorzaakt door een aanzienlijk verslechterde prestatie van de fasegestuurde microfoonopstelling die specified in testsectie gebruikt wordt. Relatieve geluidsniveaus werden verkregen door de geluidsreductietoeter te meten van een geluidsreducerende technologie. De gemiddelde maximale afwijking was niet groter dan 3 dB, waarbij verschillen doorgaans vergelijkbare geluidsniveaus opleverden binnen ±1-3 dB.
Deze resultaten laten zien dat de experimentele hardware en het gebruikte akoestische correctiekader vergelijkbare geluidsniveaus zullen opleveren voor metingen met een fasegestuurde microfoonopstelling in verschillende windtunnel testsecties.

Het effect van de zwaaihoek van een vleugelmodel op de geluidskarakteristieken ervan werden onderzocht met een multi-element vleugelmodel, de 30P30N. Aeroakoestische metingen werden uitgevoerd bij een zwaaihoek van 0° en 30°. Verschillende geluidskarakteristieken werden niet beïnvloed door de zwaaihoek. Deze resultaten toonden aan dat aeroakoestische metingen van een vleugelmodel met 0° zwaaihoek dezelfde geluidskarakteristieken moet opleveren als een vleugel met een praktische (30°) zwaaihoek.

Als laatste werd de vergelijkbaarheid onderzocht van geluidsmetingen van een gemeenschappelijk onderzoeksmodel dat hoge lift genereerd in een open straal, een gesloten en een hybride windtunnel testsectie. PIV-metingen werden uitgevoerd om het 2D-tijdgemiddelde snelheidsveld van de vleugelvoorrandklep vast te leggen. De PIV-metingen toonden aan dat de gemiddelde karakteristieken vergelijkbaar waren wanneer een vergelijkbare aerodynamische conditie werd ingesteld op basis van de $C_p$-verdeling rond de vleugelvoorrandkleppen. Metingen met een fasegestuurde microfoonopstelling werden uitgevoerd om geluidsniveaus in het vrije veld te vergelijken in een breed frequentiebereik van 1 kHz tot 25 kHz. De implementatie van een coherentieverliesmodel voor de correctie van de metingen van een fasegestuurde microfoonopstelling in een open straal test sectie wordt gepresenteerd. De geluidsspectra in het vrije veld bleken het meest vergelijkbaar te zijn tussen de metingen in de gesloten en de hybride testsecties. Het geluidsniveau in open straal testsectie vertoonde over het algemeen hogere geluidsniveaus, vooral bij lagere aanvalshoeken tot een maximaal verschil van 5 dB. Bovendien vertoonde het hoogfrequente geluid, veroorzaakt door de wervelstroom van de vleugelvoorrandklep, in de geluidsspectra ander gedrag in de open straal testsectie. Dit suggereert dat het fysieke mechanisme in deze testsectie kan worden gewijzigd. Deze studie toonde aan dat metingen met een fasegestuurde microfoonopstelling vergelijkbaar zijn in verschillende wind tunnel test secties, op voorwaarde dat de geometrisch invalshoek van het vleugelmodel wordt aangepast om de $C_p$-verdeling rond de vleugelvoorrandklep gelijk te krijgen.

In conclusie dragen de resultaten van deze studies bij aan een beter begrip van de onzekerheden van aeroakoestische metingen voor vliegtuiggeluid in een kleinschalige windtunnel. Het ontwikkelde kader voor het valideren, corrigeren en presenteren van de metingen van een fasegestuurde microfoonopstelling kan worden gebruikt om systematische meetfouten te verminderen. Daarnaast helpt het werk onderzoekers een betere
keuze te maken in het selecteren van een windtunnel testsectieconfiguratie voor een specifiek doeleind. Ten slotte zullen de meetgegevens beschikbaar worden gesteld in een open database voor validatiedoeleinden, waarbij een vergelijking met metingen in andere windtunnelfaciliteiten kan worden gedaan. Met name de datasets van de NACA-63018 en 30P30N zijn beschikbaar als maatstaf in de Hybrid Anechoic Wind Tunnel Workshop. Bovendien kunnen de 30P30N gegevens worden gebruikt in de workshop Benchmark for Airframe Noise Computations om numerieke simulaties te valideren.
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Chapter 1

Introduction

1.1 Noise Pollution

Noise pollution is an increasing health problem to our society in recent years. The industrialization of our society in the mid-18th century kick-started a seemingly boundless growth of the global population, urbanization, transportation, and energy demand. Inevitably, the sound produced in our environment by human activity has increase in parallel. While sound is essential to human life, it can also lead to adverse health effects [Kryter, 1950]. Excessive sound can damage the auditory system resulting in, e.g., tinnitus or even hearing impairment. Unwanted sound is often experienced as a nuisance referred to as noise. The exposure to excessive noise can lead to stress, sleep deprivation, cognitive impairment, and an increased occurrence of hypertension and cardiovascular disease [Basner et al., 2014]. An increasing number of studies show that noise also affects animal behavior [Myrberg, 1990], animal well-being [Kight and Swaddle, 2011] and the biodiversity [Reid et al., 2019]. Some of the earliest (and unethical) noise studies with animals even show that sound in extremis leads to surface body heating [Gierke et al., 1952] eventually leading to death [Allen et al., 1948].

In Europe, noise is estimated to be the second-largest environmental risk factor to the burden of disease following particle air pollution, and is equal to secondhand smoke inhalation [Hamminen et al., 2014]. The World Health Organization (WHO) estimated that the disability-adjusted life-years (DALYs) lost from environmental noise in western Europe are 61,000 years from ischaemic heart disease, 45,000 years from cognitive impairment from children, 903,000 years from sleep disturbance, 22,000 years from tinnitus
and 654,000 years from annoyance [Regional Office for Europe, 2011]. In other words, the WHO study suggests that at least one million healthy life years are lost each year because of environmental noise. Therefore, the WHO has developed noise guidelines to moderate the effects on human health [Regional Office for Europe, 2018].

In the Netherlands, the National Institute for Public Health and Environment (RIVM) reported in 2019 that approximately 1.3 million people experience noise nuisance from environmental sources [Welkers et al., 2019]. Road traffic accounts for 970,000 people, air traffic for 270,000 people, trains for 100,000 people, and wind turbines for 7,000 people. Due to increasing health concerns, the Dutch government introduced noise regulations, with the first noise pollution law established since 1976. These noise regulations have been further extended in recent years and follow the European Union (EU) guidelines which aim at further reducing the negative impact of noise on society based on state-of-the-art research. The focus of these guidelines is to reduce noise exposure quantified with the so-called $L_{\text{den}}$ and $L_{\text{night}}$ noise level indicators. These noise level indicators distinguish between noise exposure during the whole day ($L_{\text{den}}$) and during the night ($L_{\text{night}}$). The current WHO guidelines for environmental noise are given in Tab. 1.1. As a result of these noise guidelines and other research studies, governments in the Netherlands, Europe and the world are continuously looking for ways of reducing the impact of environmental noise on human health.

<table>
<thead>
<tr>
<th>Source</th>
<th>$L_{\text{den}}$</th>
<th>$L_{\text{night}}$</th>
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<tr>
<td>Road traffic</td>
<td>43 dB</td>
<td>45 dB</td>
</tr>
<tr>
<td>Railway</td>
<td>54 dB</td>
<td>44 dB</td>
</tr>
<tr>
<td>Aircraft</td>
<td>45 dB</td>
<td>40 dB</td>
</tr>
<tr>
<td>Wind turbine</td>
<td>45 dB</td>
<td>N/A</td>
</tr>
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Table 1.1: Recommended exposure levels to environmental noise sources by the WHO.

1.2 The Aircraft Noise Problem

Aircraft noise is a major contributor to environmental noise, and its contribution is increasing in the in recent years [Babisch et al., 2009]. Adverse health effects are nowadays widely associated with aircraft noise [Franssen et al., 2004]. In the Netherlands, it was estimated that nearly 2% of the Dutch population was exposed to more than 50 dB $L_{\text{den}}$ from aircraft traffic noise [Welkers et al., 2019] in 2015/16. Slightly less than 1% of the
1.2. The Aircraft Noise Problem

Dutch population was exposed to more than 45 dB L\text{night} in 2015/16. In both cases, the recommended limits proposed by the WHO are exceeded.

Significant progress has been made in reducing aircraft noise since the introduction of commercial aviation. An overview of the history of (jet engine powered) aircraft noise certification levels is given in Fig. 1.1, from [Choudhari, 2002]. While noise certification levels have reduced, global air traffic has increased at a seemingly boundless rate. Figure 1.2 shows the total number of passengers carried by air transport per year, as reported in [ICAO, 2022]. The primary driving forces for the increase in global air transport are the growing world population and economy, enabling more people to travel and more cargo to be transported. As a result, environmental noise has increased for residents living near airports. Frustrated residents worldwide have consequently launched more protests opposing this development and the inadequate policies of their governments.

**Figure 1.1:** History of aircraft certification noise levels, adopted from [Choudhari, 2002].
Amsterdam Schiphol Airport in the Netherlands is an exemplary case of a sizable airport located in an urban environment. Schiphol airport is positioned in the center of one of the most densely populated areas in the world. In addition, the airport is ranked in the top 15 airports of the world in terms of aircraft traffic. Persuaded by residents’ protests, the Dutch government and Schiphol Airport have been looking for ways to reduce aircraft noise exposure in recent years. At the governmental level, this has resulted in renewed urban planning policies and additional monitoring of aircraft noise around the airport. In addition, Schiphol Airport is investing in methods to optimize aircraft flight paths, flight schedules, and fleet composition to further reduce the total exposure of residents to aircraft noise.

While the COVID-19 pandemic resulted in an unprecedented collapse of global air traffic in 2020, it is expected that air traffic will steadily recover to pre-covid levels [Dube et al., 2021]. It has become clear that this recovery process should use the ‘build back better’ concept, which focuses on significantly reducing the environmental and social impact of aviation in the future. With climate change action also pressing the need for rapid changes, the aviation industry currently faces significant technical challenges. The development of fast and accurate aircraft noise prediction frameworks, e.g., in [Bertsch, 2013; Boeker et al., 2008; Lopes and Burley, 2016] and efficient noise reduction strategies, e.g., in [Thomas et al., 2017; Casalino et al., 2008; Khorrarni et al., 2018] is therefore imperative for the successful continuation of the aviation industry.
1.3 Aircraft Noise Sources

Figure 1.3 gives an overview of the noise sources on an aircraft. Typically, aircraft noise sources for commercial aircraft are divided into two categories; engine noise and airframe noise.

Engine Noise

Engine noise is the largest contributor to aircraft noise in the take-off condition [Bertsch, 2013]. Depending on the engine type, noise is generated by different components of the engine. For propeller engines, noise is generated by the propellers, which generate both broadband and tonal noise [Smith, 1989]. For turbofan/turbojet engines, noise is generated by the fans, the jet, and the compressor. This generates broadband noise and discrete tones in the far-field. Compared to airframe noise, engine noise is typically up to 20 dB higher in the take-off condition [Bertsch, 2013]. Figure 1.4a shows a comparison of the predicted noise sources for an Airbus A319 in the take-off condition with engine noise dominant over the airframe noise.

Airframe Noise

In the approach condition, airframe noise is generally louder than the engine noise (see Fig. 1.4b). Nonetheless, airframe noise is equally crucial to the global aircraft noise reduction problem. Even though engine noise is louder during take-off operations, it is generated for a shorter period of time than airframe noise. This is because the take-off flight path is much steeper than the flight path in approach. As a result, the total noise exposure levels of engine noise and airframe noise can, in general, be considered equal.

As illustrated in Fig. 1.3, various components of the aircraft generate airframe noise. Noise generated by the landing gears and wings, i.e., high-lift devices, is the most dominant. High-lift device noise is generated at various wing elements such as the leading-edge slats, flap side-edge, tracks, wingtips, or trailing edges. Other airframe noise sources originate from the wheel bay cavities, jet and flap interaction, or gear-wake and flap interaction.
Chapter 1. Introduction

Figure 1.3: Aircraft noise components with an Embraer 190.

Figure 1.4: Airbus A319 noise source ranking (predicted), adopted from [Bertsch, 2013].

The ranking of noise sources highly depends on the aircraft type and its operating conditions. Figure 1.5 shows a breakdown of the air traffic per aircraft type at Amsterdam Schiphol Airport. The most commonly used aircraft type is the so-called narrow-body aircraft type, with the Boeing 737-800 the most common at Amsterdam Schiphol Airport. This type of aircraft is a single-aisle design used for short and medium-range distances. Several research studies have been performed to localize and quantify the noise sources of this aircraft type using an acoustic imaging technique in fly-over or wind tunnel measurements. Figure 1.6 shows acoustic images calculated from fly-over measurements of a Boeing 737-800 at Amsterdam Schiphol Airport in [Snellen et al.,]
Figure 1.6a shows a fly-over measurement where the engine noise is dominant, whereas Fig. 1.6b shows a fly-over measurement where the airframe noise, i.e., landing gear and high-lift devices, are dominant. More detailed acoustic images of the airframe noise generated by a Boeing-737 NG (third-generation) aircraft type are shown in Fig. 1.7, from [Stoker et al., 2008]. These images, taken from wind tunnel tests, show the dominant noise sources on the high-lift device, namely the leading-edge slat and flap-side edge. Note that the landing-gear noise mainly has a low-frequency contribution and is therefore not visible in these images. The airframe noise studies in [Snellen et al., 2017; Stoker et al., 2008] show (amongst others) the essential sources of airframe noise on a typical narrow-body aircraft type. These airframe noise sources are the landing gears, the leading-edge slats, and the flap side-edge.

**Figure 1.5:** Air transport movement per aircraft type at Amsterdam Schiphol Airport (2019). Narrow-body aircraft type (○) and wide-body aircraft type (□)
Chapter 1. Introduction

Figure 1.6: Acoustic images from flyover measurement of a Boeing 737-800 (third generation) at Amsterdam Schiphol Airport. Functional beamforming for the frequency range of 50 Hz to 11,200 Hz. Images taken from [Snellen et al., 2017]

Figure 1.7: Acoustic image from wind tunnel measurements of an 11% scaled Boeing 737 NG (third generation) semi-span model in NASA’s National Transonic Facility. Source maps at 21.7 kHz (left) and 61.3 kHz (right) for a typical approach condition. Images taken from [Stoker et al., 2008]

1.4 The Role of Wind Tunnel Testing in the Noise Reduction Challenge

The reduction of aircraft noise requires a multidisciplinary approach with efforts at different levels [Delfs et al., 2018]. Firstly, on the component level, the aircraft can be optimized for low noise emission by changing, e.g., the landing gear [Dobrzynski
et al., 2006], the flaps on the wings [Guo, 2004] or the engines [Zaman et al., 2011]. An extensive bibliographical review of noise reduction technologies for aircraft components can be found in [Casalino et al., 2008]. Secondly, noise reduction at the aircraft level can be achieved by optimizing the overall aircraft design, engine operating conditions, or wing settings [Spakovsky, 2019]. Finally, on the system level, the aircraft flight path or ground topology can be optimized to reduce the noise exposure levels on the ground [Visser and Wijnen, 2001]. These ground noise levels are typically estimated using noise prediction models for individual aircraft components which are summed, and are calculated at discrete locations along the flight path. Well-known aircraft noise prediction models at the system level are PANAM [Bertsch, 2013] and ANOPP [Lopes and Burley, 2016].

Wind tunnel testing is a cornerstone of the development of noise reduction concepts for airframe noise. Large-scale wind tunnel tests of scaled aircraft models allow for the localization and quantification of individual airframe noise components in a controlled environment. Low-noise aircraft settings can be evaluated along with the corresponding aerodynamic condition. The measurement data can then be used to propose low-noise operating conditions of the aircraft during the approach. In addition, low-noise technologies, such as add-ons can be evaluated. Small-scale wind tunnel tests of simplified aircraft components can be used to develop analytical noise prediction models and validate computational aeroacoustic (CAA) simulations. Analytical noise prediction models often require modeling detailed physical parameters that can only or more accurately be measured in small-scale wind tunnel facilities. CAA simulations are still limited to simplified geometries at a Reynolds number similar to that found in small-scale wind tunnel testing, which is significantly below a full-scale aircraft. Nevertheless, these CAA simulations are widely used during the design process of an aircraft. Combining wind tunnel measurement data allows for the development of semi-empirical noise prediction models. These models gain more accuracy if many measurements at different operating conditions are used. Wind tunnel measurements allow for the collection of large datasets within a reasonable time and with low economic cost. Ultimately, wind tunnel testing therefore helps to improve and develop a wide range of noise prediction and noise reduction tools.

Wind tunnel tests are sensitive to considerable uncertainty for several reasons, such as the microphone phased array technique and systematic measurement errors. In addition, there is no standard method for translating noise measurements to a standardized free-field condition. Therefore, the comparability of noise measurements performed in wind
tunnels is not well known. This uncertainty subsequently hampers noise prediction tools developed from wind tunnel tests as the uncertainty is specific for the wind tunnel itself. Reducing uncertainties in aeroacoustic measurements performed in wind tunnels is beneficial to aircraft manufacturers and policymakers because it reduces product development costs and improves the evaluation of aircraft noise in human health.

1.5 Research Objectives and Thesis Outline

The primary aim of this thesis is to better understand the uncertainty of aeroacoustic measurements for airframe noise in a small-scale wind tunnel facility. This will lead to an optimal use of small-scale wind tunnel testing for the preparation of industrial scale tests and eventually towards full-scale flight comparison. Chapter 2 gives a theoretical background of common concepts of aerodynamic noise generation and its measurement. A brief literature review will be given discussing the causes of uncertainty in wind tunnel testing and currently available comparability studies. Chapter 3 gives a description of the experimental aeroacoustic facility and methods at the University of Twente. Most of the experimental setups and methods were developed from the ground for the work presented in this thesis. A comprehensive description of the design and characterization of the aeroacoustic wind tunnel facility at the University of Twente is therefore given and of the experimental methods used. A detailed description of the instrumented airfoil models and in-house developed microphone phased array technique is also given. In Chapter 4 the comparability of trailing-edge noise measurements of single-element airfoils in an open-jet, a closed, and a hybrid wind tunnel test section is investigated. The near-field sound source statistics were measured with wall-pressure microphones, and the far-field noise is determined using a microphone phased array. Absolute and relative noise levels (i.e., noise reduction from trailing-edge serrations) are compared in each test section configuration. Chapter 5 discusses the comparability of slat noise measurements performed on an unswept and swept (30°) multi-element high-lift model. The comparability of slat noise measurements in an open-jet, a closed, and a hard-wall wind tunnel test section is discussed in Chapter 6. The flow inside the slat cove was measured using 2D time-averaged PIV measurements. Far-field noise measurements were performed using a microphone phased array. Finally, in Chapter 7 we present the conclusions and an outlook for future research.
REFERENCES

References


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Chapter 2

Airframe Noise

Airframe noise results from a set of flow mechanisms that have complex underlying physics. The reduction and measurement of airframe noise are therefore also not straightforward. In this chapter, we described some fundamentals of acoustic and aeroacoustic concepts. Firstly, we start with sound propagation and the generation of aerodynamic sound. Secondly, the noise sources from a simple single-element airfoil are explained. Thirdly, the noise sources from high-lift devices are explained and discussed. Finally, basic concepts of aeroacoustic measurements and the associated difficulties and uncertainties are discussed.

2.1 The Wave Equation

The propagation of sound in air can be described using the linear wave equation given as

\[
\frac{1}{c_0^2} \frac{\partial^2 p'}{\partial t^2} - \nabla^2 p' = 0 \tag{2.1}
\]

where \(c_0\) is the speed of sound and \(p' = p - p_0\) is the pressure perturbation. The equation can be derived from the continuity and the momentum equation with a set of assumptions. First, the perturbations are isentropic, i.e., there is no heat transfer and viscosity. Second, the perturbations are small. Finally, the pressure and density of the fluid are constant in the medium and independent of time. The medium can also be assumed to have a constant mean velocity \(\bar{U}\) for which the convective wave equation
Chapter 2. Airframe Noise

can be derived given as
\[ \frac{1}{c_0^2} \left( \frac{\partial}{\partial t} + \mathbf{U} \cdot \nabla \right)^2 p' - \nabla^2 p' = 0. \] (2.2)

In spherical coordinates, the wave equation is given as
\[ \frac{1}{c_0^2} \frac{\partial^2 p'}{\partial t^2} - \frac{1}{r} \frac{\partial^2}{\partial r^2} (rp') = 0 \] (2.3)

where \( r \) is the radial coordinate. A solution to this equation for outward propagating waves is given as
\[ p'(r, t) = \frac{f(t - r/c_0)}{r} \] (2.4)

which shows that the pressure perturbation decays in amplitude with \( 1/r \). This is commonly referred to as the spherical spreading loss. It is practical to consider the pressure perturbations as waves with a complex harmonic time dependence. The general solution for such waves is
\[ p'(r, t) = A e^{i2\pi f(t - r/c_0)} \frac{1}{r} \] (2.5)

where \( A \) is the amplitude of the wave, \( f \) the frequency and \( t \) the time. This solution is useful for the source localization techniques which will be explained in Sec. 3.3.

2.2 Sound Generation by Flow

2.2.1 The Acoustic Analogy

The generation of aerodynamic sound in a flow can be generally described by Lighthill’s equation [Lighthill, 1952] as
\[ \frac{\partial^2 \rho'}{\partial t^2} - c_0^2 \frac{\partial^2 \rho'}{\partial x_i^2} = \frac{\partial^2 T_{ij}}{\partial x_i x_j} \] (2.6)

where
\[ T_{ij} = \rho v_i v_j + [(p - p_0) - (\rho - \rho_0)c_0^2] \delta_{ij} - \sigma_{ij} \] (2.7)
2.2. Sound Generation by Flow

is the so-called Lighthill’s stress tensor where $\delta_{ij}$ is the Kronecker delta function and $\sigma_{ij}$ is the viscous stress. Lighthill’s equation describes sound propagation from a turbulent flow region where the source term is given as the right-hand side of Eq. 2.6. In the derivation of the equation, no assumptions are made on the source region, while a linear assumption is made for the sound propagation at the observer position. The Lighthill equation shows that there are three sources of aerodynamic sound; the (non-linear) Reynolds stress $\rho v_i v_j$, the deviation from an ideal fluid with density $\rho_0$ and speed of sound $c_0$, and $\sigma_{ij}$ the viscous stress tensor.

The acoustic analogy by Lighthill provides a general equation that describes the generation of aerodynamic sound from a turbulent flow region in free space. However, aerodynamic sound in practical applications is generated on solid surfaces such as wind turbine blades or aircraft wings. A general solution in the presence of solid surfaces in the flow was proposed in [Curle, 1955] and is given as

$$\rho'(\vec{x}, t) c_0^2 = \int_{-T}^{T} \int_{S} \left( [p_{ij} + \rho v_i v_j] \frac{\partial G}{\partial y_i} + G \frac{\partial (\rho v_j)}{\partial \tau} \right) n_j d^2 \vec{y} d\tau$$

$$+ \int_{-T}^{T} \int_{V} \left( \frac{\partial^2 G}{\partial y_i y_j} \right) T_{ij}(\vec{y}, \tau) d^2 \vec{y} d\tau$$

(2.8)

(2.9)

where $\vec{x}$ is an observer location, $\vec{y}$ is a location inside the source volume, $p_{ij} = p \delta_{ij} - \sigma_{ij}$, $\tau = t - ||\vec{x} - \vec{y}|| / c_0$ is the emission time and $G$ is a Green’s function which is a solution of the inhomogeneous wave equation

$$\frac{1}{c_0^2} \frac{\partial^2 G}{\partial t^2} - \frac{\partial^2 G}{\partial y_i^2} = \delta(\vec{x} - \vec{y}) \delta(t - \tau).$$

(2.10)

Using the free-field Green’s function, Eq. 2.8 can be rewritten to

$$\rho'(\vec{x}, t) c_0^2 = \int_{S} \left[ \frac{\partial (\rho v_j)}{\partial \tau} \right]_{t=\tau} \frac{n_j d^2 \vec{y}}{4\pi ||\vec{x} - \vec{y}||} - \frac{\partial}{\partial x_i} \int_{S} [p_{ij} + \rho v_i v_j]_{t=\tau} \frac{n_j d^2 \vec{y}}{4\pi ||\vec{x} - \vec{y}||}$$

$$+ \frac{\partial^2}{\partial x_i \partial x_j} \int_{V} [T_{ij}(\vec{y}, \tau)]_{t=\tau} \frac{d^2 \vec{y}}{4\pi ||\vec{x} - \vec{y}||}.$$ 

(2.11)

(2.12)

This equation is an important results because it shows how sound it produced from three contributions; a monopole, a dipole and a quadrupole source.
2.2.2 Monopole, Dipole and Quadrupole Sources

The monopole term in Eq. 2.11 is the mass flux term over the surface

$$\rho'(\vec{x},t)c_0^2 = \int_S \left[ \frac{\partial(\rho v_j)}{\partial \tau} \right]_{t=\tau^*} \frac{n_j d^2\vec{y}}{4\pi |\vec{x} - \vec{y}|}.$$ (2.13)

For a stationary solid object with an impermeable surface this term is zero. Monopole sound is produced, e.g., by the unsteady injection of mass or by a pulsating sphere.

The dipole source term in Eq. 2.11 is the surface force loading term

$$\rho'(\vec{x},t)c_0^2 = -\frac{\partial}{\partial x_i} \int_S \left[ p_{ij} + \rho v_i v_j \right]_{t=\tau^*} \frac{n_j d^2\vec{y}}{4\pi |\vec{x} - \vec{y}|}.$$ (2.14)

If we consider the surface to be solid then the Reynolds stress term $\rho v_i v_j$ is zero and sound is generated by the fluid stress $p_{ij}$ term. The equation can be rewritten to the form

$$\rho'(\vec{x},t)c_0^2 \approx \frac{x_i x_j}{4\pi |\vec{x}|^2 c_0^3} \int_S \left[ \frac{\partial(p_{ij}n_j)}{\partial \tau} \right]_{t=\tau^*} d^2\vec{y}.$$ (2.15)

by assuming that the surface is acoustically compact. A scaling law for the far-field sound intensity can be derived which will give

$$I_r \propto \frac{\rho_0 U^6 S^2 \cos^2 \theta}{(4\pi |\vec{x}|)^2 c_0^3 L^2}$$ (2.16)

where $I_r$ is the sound intensity, $U$ the mean flow velocity, $S^2$ the surface area of the body, $\theta$ the observer angle relative to the dipole axis and $L$ a characteristic length scale of the turbulence.

The volume integral in Eq. 2.11 is the quadrupole term

$$\rho'(\vec{x},t)c_0^2 = \frac{\partial^2}{\partial x_i \partial x_j} \int_V [T_{ij}(\vec{y},\tau)]_{t=\tau^*} \frac{d^3\vec{y}}{4\pi |\vec{x} - \vec{y}|}.$$ (2.17)

which can be rewritten to the form

$$\rho'(\vec{x},t)c_0^2 \approx \frac{x_i x_j}{4\pi c_0^2 |\vec{x}|^3} \left[ \int_V \frac{\partial^2 T_{ij}}{\partial \tau^2} d^3\vec{y} \right]_{t=\tau^*}.$$ (2.18)
2.2. Sound Generation by Flow

This term states that sound is produced by the second time derivative of Lighthill’s stress tensor. It can be found that the far-field sound intensity scales as

\[ I_r \propto \frac{\rho_0 U^8 V^2}{(4\pi \|\vec{x}\|)^2 c_0^5 L^4 \left(\frac{x_i x_j}{\|\vec{x}\|^2}\right)^2} \]  

(2.19)

where \( V \) is the volume of the turbulent flow region and \( L \) is the typical length scale of the turbulence in the flow region. For low Mach number flow, i.e., \( M<<1 \) it can be shown that the quadrupole source term is smaller than the dipole source.

Although Lighthill’s acoustic analogy is not directly applicable to solve the sound generated in practical applications, it does provide a general explanation of aerodynamic sound sources. The Ffowcs Williams and Hawkings (FWH) equation is an extension of the Lighthill equation, which includes arbitrary source motion and is widely applied to solve practical problems. The FWH equation, combined with CFD simulations, is the most widely used method for performing computational aeroacoustics (CAA) simulations.

Lighthill’s acoustic analogy also demonstrates why a distinction is made between airframe noise sources and engine noise from an aircraft. This is because engine noise is classified as a quadrupole sound source, whereas airframe noise is classified as a dipole sound source. Commercial aircraft generally have turbofan engines which comprise a jet engine core surrounded by a fan blade. Jet engines produce jet noise, a quadrupole noise source that scales with \( U^8 \). Therefore, reducing the exit velocity of the jet significantly reduces the overall noise production of the aircraft. However, doing so would reduce the generated thrust, which is an impractical trade-off. Fortunately, the exit velocity of the jet can be reduced while maintaining thrust by increasing the surface area of the fan blades surrounding the jet. This type of turbofan engine is called a high-bypass engine for its higher mass flow through the fan blades relative to the jet core. For this reason, modern turbofan engines have a large diameter. On the other hand, airframe noise sources are dipole noise sources in which the interaction of turbulence with the solid body plays a primary role. Therefore, aerodynamic shape optimization or surface treatment should be applied to minimize this sound source type. Ultimately, the noise generating mechanisms are different for each aircraft component, and therefore each requires a specific examination to understand, model, and reduce the overall noise generation.
2.3 Noise From a Single Element Airfoil

In the previous section, we discussed the fundamental sources that generated aerodynamic sound in a turbulent flow region and their general characteristics. In this section, we will discuss the different noise mechanisms on a single-element airfoil.

Laminar Boundary Layer Noise  Laminar boundary layer noise (see Fig. 2.1) occurs when a laminar boundary layer is present on the airfoil [Nash et al., 1999; Plogmann et al., 2013; Arbey and Bataille, 1983; Kingan and Pearse, 2009]. This mechanism occurs when the chord-based Reynolds number is low. The precise Reynolds number range in which laminar boundary layer noise occurs depends on the angle of attack and airfoil shape. Noise is generated at the trailing edge of the airfoil due to the scattering of Tollmien-Schlichting instability waves from the laminar boundary layer. The radiated sound waves amplify these instability waves in the boundary layer and form an efficient feedback mechanism. As a result, the far-field noise spectrum is characterized by several tones at discrete frequencies.

Turbulent Boundary Layer Noise  The turbulent boundary layer noise (i.e., trailing edge noise) mechanism occurs when the boundary layer over the airfoil becomes turbulent (see Fig. 2.2). The turbulent pressure fluctuations inside the boundary layer are scattered at the trailing edge discontinuity, resulting in broadband noise generation [Amiet, 1976]. This noise mechanism is predominant in the far-field noise generation of, e.g., wind turbines in low atmospheric turbulence conditions. A classical review of trailing edge noise prediction models can be found in [Howe, 1978]. Nowadays, semi-empirical models are still widely used, following the model proposed in [Brooks et al., 1989]. Semi-analytical models based on Amiet’s trailing edge noise model [Amiet, 1976] are also commonly used. However, this model still relies on empirical models for the wavenumber-frequency spectrum of the unsteady wall pressure near the trailing edge [Kamruzzaman et al., 2015; Lee and Shum, 2019]. Efforts toward a non-empirical
2.3. Noise From a Single Element Airfoil

trailing edge noise model are described in [Lee and Shum, 2019], primarily using the
so-called TNO-Blake model [Parchen, 1998]. A brief explanation of Amiet’s trailing
edge noise model will be given here because this mechanism is studied in Chapter 4.

Amiet [Amiet, 1976] proposed a simplified trailing edge noise prediction model where
the far-field noise spectrum \( S_{pp}(\omega) \) for an observer in the mid-span plane is given by

\[
S_{pp}(x, 0, z, \omega) = \left( \frac{\omega c z b}{4\pi c_0 \sigma^2} \right) |L(x, z, k_x, U_0)| l_y(\omega) \phi_{pp}(\omega) \tag{2.20}
\]

where \( \omega \) is the angular frequency, \( c \) is the airfoil chord, \( \sigma^2 = x^2 + (1 - U_0/c_0)^2 z^2 \) is the flow
corrected distance between source and observer, \( b \) is the airfoil span, \( |L(x, z, k_x, U_0)| \) is an
analytical expression called the airfoil response function, \( l_y(\omega) \) is the spanwise correlation
length and \( \phi_{pp}(\omega) \) is the wall pressure spectrum. Figure 2.3 illustrates the coordinate
system. The parameters between brackets in Eq. 2.20 are setup-dependent parameters,
whereas \( l_y(\omega) \) and \( \phi_{pp}(\omega) \) are empirical parameters which need to be determined from
experimental measurements. The wall pressure spectrum \( \phi_{pp}(\omega) \) can simply be obtained
from a single microphone measurement near the trailing edge of an airfoil whereas \( l_y(\omega) \)
requires an array of microphones along the spanwise and streamwise direction near the
trailing edge on the airfoil. The spanwise correlation length can be determined from the
integral

\[
l_y(\omega) = \int_0^\infty \sqrt{\gamma^2(\omega, \Delta \xi_y)} d(\Delta \xi_y) \tag{2.21}
\]

where \( \gamma^2(\omega, \Delta \eta_y) \) is the coherence function between two microphone pairs separated by
a spanwise distance of \( \Delta \eta_y \). The direct measurement of this coherence function is often
impractical. Therefore, a model for the coherence function is typically used [Corcos,
1964; Brooks and Hodgson, 1981] which is given as

\[
\gamma^2(\omega, \Delta \eta_y) = e^{-(2\omega b_c U_c)\Delta \xi_y} \tag{2.22}
\]

where \( b_c \) is the so-called Corcos constant and \( U_c \) the convection velocity which needs to be
determined from the phase spectrum between streamwise microphone pairs. Chapter 4
focuses on the trailing edge noise mechanism where wall pressure measurements are used
to characterize the sound source statistics following Amiet’s trailing edge noise model.
Chapter 2. Airframe Noise

Blunt Trailing Edge Noise  With a blunt trailing edge, vortex shedding in the wake of the airfoil is induced. This causes periodic fluctuations around the trailing edge, which results in narrow-band noise generation to the far-field. Figure 2.4 illustrates the noise mechanism.

Separation Noise  Boundary layer separation noise occurs when separation of the boundary layer occurs near the trailing edge of the airfoil (see Fig. 2.5). This creates large-scale structures in the boundary layer leading to an increase of low-frequency noise in the far-field noise spectrum [Lacagnina et al., 2019].
2.3 Noise From a Single Element Airfoil

Stall Noise If the flow separation region increases, stall can occur. Figure 2.6 illustrates the noise mechanism. The large-scale structures become even more prominent, leading to distinct low-frequency humps in the far-field noise spectrum. Other flow regimes have also been reported in the deep stall, which lead to narrow-band peaks caused by shear layer instabilities and vortex shedding in the wake [Moreau et al., 2009; Lacagnina et al., 2019]. A semi-empirical stall noise prediction model is given in [Bertagnolio et al., 2017].

Leading Edge Noise Leading-edge noise (i.e., inflow turbulence noise) is generated when the inflow turbulence has a sufficiently high intensity [Amiet, 1975] (see Fig. 2.7). This generates broadband noise in the far-field. Leading-edge noise is dominant over the trailing edge noise mechanisms depending on the intensity of the inflow turbulence. The inflow turbulence can also affect the development of the airfoil’s boundary layer, thereby affecting the behavior of the other noise mechanisms [Botero et al., 2021]. Amiet’s leading-edge noise prediction model is widely used to predict the noise from this mechanism [Amiet, 1975].
Chapter 2. Airframe Noise

Tip vortex noise

At the tip of an airfoil, noise is generated by the interaction of the tip vortex with the airfoil’s surface [Brooks and Marcolini, 1986] (see Fig. 2.8). A tip vortex is produced by the pressure difference between the airfoil’s upper and lower sides, which forces the flow to curl around the tip. This mechanism generates mainly high-frequency broadband noise.

Figure 2.8: Tip vortex noise mechanism.

2.4 Noise From a Multi-Element Airfoil

In Sec. 1.3 we have discussed the ranking of airframe noise sources from which it followed that the landing gear and the high-lift devices are the most dominant. In this thesis, we will focus on the high-lift devices and will not consider landing gear noise in any studies. Therefore, a brief explanation of the aerodynamic and aeroacoustic characteristics of high-lift devices is given, and a review of the state-of-the-art noise prediction models.

A schematic cross-section of the high-lift devices commonly found on a narrow-body aircraft type is shown in Fig. 2.9. The wings comprise a leading-edge slat element, the main element, and a flap element, each with a specific function. Firstly, the leading-edge slat can be deployed, which primarily functions to increase the maximum stall angle of the wing configuration. Secondly, the main element provides the majority of the total lift. Lastly, the flap element can be deployed to increase the whole wing’s camber line, thereby increasing the total lift force generated. The configuration of the individual high lift devices can be changed to specific settings during flight. During take-off, the high-lift devices are mainly deployed to increase the maximum stall angle and to enhance lift. This allows the aircraft to gain altitude in a short period of time. In cruise, all elements are retracted (i.e., stowed) to reduce drag. During landing, the slat and flap elements are deployed to increase lift and generate drag. This allows the aircraft to pitch down and decrease speed. Deploying the high lift devices also provide stability during landing.
2.4. Noise From a Multi-Element Airfoil

Figure 2.9: Schematic cross section of an aircraft wing with the high-lift devices.

Most aircraft types use a conventional slat element, as shown in Fig. 2.10a. However, a few aircraft (such as the 737-700) also use the leading-edge Krueger flap type shown in Fig. 2.10a. A Krueger flap increases the overall lift generated by the wing but, contrary to a leading-edge slat, has a minimal effect on the maximum stall angle. Therefore, leading-edge Krueger flaps are typically only used on the inboard wing section between the fuselage and engine. This is done because a leading-edge slat is less practical in this part of the wing and ensures that stall occurs at the root of the wing first. This ensures the stability of the aircraft when a stall occurs because tip stall could otherwise lead to a rolling movement of the aircraft. A study in [Bahr et al., 2016] showed that the noise characteristics of a Krueger flap follow the same behavior as that of the conventional slat.

Figure 2.10: Leading-edge high lift devices on an aircraft wing.
2.4.1 Noise Mechanisms

The flow around high-lift devices is complex, leading to various potential aerodynamic noise sources. Let us first consider the noise mechanisms of a simplified 2D high-lift airfoil geometry. Figure 2.11 shows a visualization of the vortical structures and noise generation from a DDES simulation of the 2D 30P30N high-lift geometry from [Knacke and Thiele, 2013]. At the cusp of the leading-edge slat, a shear layer is formed between the high-velocity free-stream flow and low-velocity flow in the slat cove. This creates strong turbulent fluctuations in the shear layer, which reattach to the slat surface upstream of the trailing edge. Subsequently, these turbulent pressure fluctuations are accelerated through the gap between the main element and slat. Noise is efficiently generated at the slat’s trailing edge due to the intense pressure fluctuations and high velocity of the flow. On the upper side of the main element, the wake from the shear layer merges with the boundary layer from the main element. A boundary layer also forms on the main element’s lower side, which separates at the flap cove. A shear layer is formed at the flap cove, which is accelerated through the gap between the main element and flap. The pressure fluctuations of this shear layer and its velocity through the flap gap are much lower than that of the shear layer in the slat cove. Therefore, the noise generation at the flap cove is lower than the noise produced at the slat. The wake from the main element merges with the flap element’s boundary layer on the upper side. A small flow separation region typically arises on the upper side of the flap element near the trailing edge. The boundary layer on the upper and lower side of the flap element creates a trailing edge noise mechanism. However, this noise generation mechanism is much lower than the noise generation mechanism at the leading-edge slat.

Considering a three-dimensional wing, several other noise sources can be identified [Dobrzynski, 2010]. These noise sources are the slat tracks, the inboard slat side-edge (slat horn), the flap tracks, and the flap side-edge. Slat tracks are known to produce noise that can be higher than that of purely the clean slat in certain conditions [Murayama et al., 2015, 2016]. However, slat track noise is not extensively studied and quantified in the current literature. This is primarily because scaled models do not accurately capture the geometric details of slat tracks and realistic slat track geometries are not publicly available to the scientific community. Slat horn noise is the result of three-dimensional flow from the opening between the fuselage and slat when the slat devices are deployed [Piet et al., 2005; Molin et al., 2003]. In general, slat horn noise has an insignificant contribution to the overall noise produced by the high-lift devices because
it is a local phenomenon. Similar to slat tracks, flap tracks can also generate noise [Dobrzynski et al., 2008]. However, the contribution is often low, although it could become a significant noise source for future low-noise aircraft. Flap side-edge noise is a primary contributor to the overall noise from high-lift devices. It is the most important noise source, together with leading-edge slat noise. However, this noise mechanism will not be further discussed in this thesis because it has not been the focus of this research.

![Figure 2.11: Visualization of the vortical structures and noise generation from a DDES simulation of the 30P30N geometry, adopted from [Knacke and Thiele, 2013].](image)

### 2.4.2 Slat Noise

The physical mechanisms responsible for slat noise are discussed in [Choudhari and Khorrami, 2007] and shown in Fig. 2.12. At the lower side of the slat, a boundary layer is formed, which detaches at the slat cusp and forms a shear layer. Instabilities are formed in the shear layer, which are amplified along the shear layer path. The shear layer path typically reattaches to the slat surface upstream of the trailing edge. This leads to strong pressure fluctuations at the reattachment point. Part of the flow is entrained in the slat cove, whereas most flow is rapidly accelerated towards the slat trailing edge. The turbulent pressure fluctuations from the shear layer (and boundary layer on the upper side of the slat) are scattered efficiently at the slat’s trailing edge, leading to broadband noise radiation in the far-field.

In addition to broadband noise, several other characteristics can be observed in the far-field spectrum of slat noise. Firstly, a bulk cove oscillation was observed in measurements in [Pascioni and Cattafesta, 2018] which caused a shear layer flapping motion. Their measurements suggest that this mechanism generates a tonal peak in the far-field spectrum at a Strouhal number of approximately 0.15 when the slat chord is the characteristic length. Secondly, additional tonal peaks appear in the far-field slat noise.
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spectrum in low Reynolds number testing. The occurrence of these peaks is attributed to a feedback loop between coherent large-scale instabilities in the shear layer and the noise generated by the slat [Terracol et al., 2016]. Lastly, vortex shedding at the trailing edge of the slat causes a narrow-band hump in the high-frequency range of the far-field noise spectrum. This vortex shedding mechanism is related to the thickness of the slat trailing edge [Khorrami et al., 2000] and is therefore also not observed in full-scale measurements.

**Figure 2.12:** Physical mechanisms responsible for slat noise, adopted from [Choudhari and Khorrami, 2007].

**Slat Noise Prediction Models**

A noise prediction model for the leading-edge slat device was first proposed in [Fink, 1977]. Fink proposes a semi-empirical method using empirical parameters based on an extensive dataset of flyover measurements. According to Fink, leading-edge slat noise is generated by three mechanisms. Firstly, trailing edge noise from the slat itself generates noise. Secondly, bluff-body noise is generated at high velocity due to exposed actuators and tracks in the slat gap. Thirdly, the slat wake on the upper side of the main element is assumed to affect the main element’s boundary layer thickness and increase its turbulence intensity. This is assumed to increase the trailing-edge noise generated by the main element. Fink then proposes to model slat noise similar to the ‘clean wing’ trailing edge noise but with a broadening of the spectrum towards high frequencies combined with a 3 dB level increase. Even though this method relies
2.4. Noise From a Multi-Element Airfoil

heavily on empirical parameter fitting, it is still widely used in airframe noise prediction frameworks such as the ANOPP [Lopes and Burley, 2016] and ESDU frameworks.

Another semi-empirical method to predict slat noise is proposed in [Pott-Pollenske et al., 2006; Bertsch, 2013] and used by DLR. In this model, the power spectral density of the far-field slat noise \( L_{\text{slat}} \) is predicted as

\[
L_{\text{slat}}(St) = L_{\text{norm/spec}}(St) + \Delta L_{\text{geo}} + \Delta L_{\text{vel}} + \Delta L_{\text{dir}} \tag{2.23}
\]

where \( L_{\text{norm/spec}}(St) \) is a normalized reference spectrum, \( \Delta L_{\text{geo}} \) is a geometry-dependent adjustment factor, \( \Delta L_{\text{vel}} \) is a velocity scaling factor with respect to a reference condition and \( \Delta L_{\text{dir}} \) is a polar and azimuthal directivity correction.

The normalized reference spectrum is given as

\[
L_{\text{norm/spec}}(St) = \begin{cases} C_1 + 3 \log_{10}(St) + \Delta L_{\text{conf}} & \text{for } St \leq St_{\text{max}} \\ C_2 - 18 \log_{10}(St) & \text{for } St > St_{\text{max}} \end{cases} \tag{2.24}
\]

where \( St = f c_s/v_\infty \) is the Strouhal number with \( c_s \) the slat chord length and \( v_\infty \) the reference velocity. The maximum Strouhal number \( St_{\text{max}} \) is given as

\[
St_{\text{max}} = 10^{(C_2-C_1-\Delta L_{\text{conf}})/21} \tag{2.25}
\]

where \( C_1 \) and \( C_2 \) are empirical constants. The configuration dependent adjustment factor \( \Delta L_{\text{conf}} \) is given as

\[
\Delta L_{\text{conf}} = -15.6 + 0.416(\delta_s + \delta_f) - 0.0025(\delta_s + \delta_f)^2 \tag{2.26}
\]

with \( \delta_s \) and \( \delta_f \) the deflection angle of the slat and flap element respectively. The geometry dependent adjustment factor \( \Delta L_{\text{geo}} \) is given as

\[
\Delta L_{\text{geo}} = \log_{10}\left(\frac{r_{\text{ref}}^2}{r^2}\right) + \log_{10}\left(\frac{c_s \Delta w/\cos(\Lambda)}{c_{s,\text{ref}} \Delta w_{\text{ref}}/\cos(\Lambda_{\text{ref}})}\right) \tag{2.27}
\]

where \( r \) is the observer distance, \( \Delta w \) the slat span width and \( \Lambda \) the slat sweep angle. Parameters with the subscript \( \text{ref} \) are the reference parameters. The velocity adjustment factor is given as

\[
\Delta L_{\text{vel}} = 50 \log_{10}\left(\frac{v_\infty}{v_{\text{ref}}}\right) + 30 \log_{10}\left(\frac{\cos(\Lambda)}{\cos(\Lambda_{\text{ref}})}\right) \tag{2.28}
\]
Directivity is accounted for with $\Delta L_{\text{dir}}$ given as

$$\Delta L_{\text{dir}} = \Delta L_{\theta} + \Delta L_{\Phi}$$ (2.29)

$$\Delta L_{\theta} = 10 \log_{10}(\sin^2(|\theta - \delta_s - \gamma + \alpha_g|) + 0.1 \cos^2(|\theta - \delta_s - \gamma + \alpha_g|))$$ (2.30)

$$\Delta L_{\Phi} = 10 \log_{10}((|\cos(\Phi - \nu)| + \cos |\Phi + \nu|)/2)$$ (2.31)

where $\Delta L(\theta)$ is the polar correction, $\Delta L(\Phi)$ is the azimuthal correction with $\gamma$ the slat installation angle, $\alpha_g$ the geometric angle of attack of the wing, and $\nu$ the dihedral angle.

The DLR model proposes that slat noise has a fifth power scaling with the velocity, similar to that of a non-compact trailing edge source. Moreover, the spectral shape and magnitude depend only on geometrical wing parameters: the slat chord length $c_s$, slat deflection angle $\delta_s$, and flap deflection angle $\delta_f$. No considerations are made regarding the specific shape of the leading-edge slat or physical features of the flow. This model has been successfully validated with flyover measurements [Fleury and Malbequi, 2013].

Guo proposed a slat noise prediction model, which is based on modeling statistical properties of some physical flow features in terms of the surface pressure coherence and characteristic time and length scales [Guo, 2012]. However, the model requires fitting empirical parameters to a sufficiently large dataset. Moreover, it should be noted that Guo does not propose a fully defined model, meaning that the modeling of specific parameters can be done arbitrarily based on experimental observations. Guo proposes to model the power spectral density of slat noise in the far-field in the form

$$L_{\text{slat}} = \rho_0^2 c_0^4 A_G A_F W(M) F(f_d, M) D(\theta, \Phi) \frac{c_s \Delta w}{\Delta^2 r^2} e^{-\alpha_0 r}$$ (2.32)

where $\rho_0$ is the mean density, $c_0$ is the mean speed of sound, $A_G$ is a geometry dependent amplification factor, $A_F$ is a flow dependent amplification factor, $W(M)$ is a function to model the Mach number dependence, $F(f_d, M)$ is the spectral shape function, $D(\theta, \Phi)$ is the polar and azimuthal directivity correction, $\Delta^2$ is a convection amplification factor and $e^{-\alpha_0 r}$ a function to account for atmospheric attenuation.

The spectral shape function is given as

$$F(f_d, M) = \frac{M^2 c_s}{c_0^2} \frac{St^2}{(1 + \mu_0^2 St^2)(1 + \mu_1^2 (1 + M^2 St^2)(1 + \mu_2^2 M^2 St^2)(1 + \mu_3 M St))}$$ (2.33)
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with $\mu_n$ constants related to the characteristic time and length scales of the local flow. Equation 2.33 is essentially a fifth-order polynomial function that can be fitted to experimental far-field noise data. The spectral shape function has a Mach number dependency which varies for different parts of the spectrum. The $W(M)$ function is introduced to ensure a fifth power law scaling with Mach number

$$W(M) \int_{f_1}^{f_2} F(f, M) df = M^5$$

where $f_1$ and $f_2$ give the frequency range which includes at least the most dominant contribution of the power spectral density of the far-field slat noise. Two other important parameters can be identified in Eq. 2.32 which are the geometric amplification factor $A_G$ and flow dependent amplification factor $A_F$. The variation of geometric properties of the slat and their effect on the far-field noise can be modeled with the $A_G$ parameter. Guo suggests that

$$A_G = A_G(\alpha_g, \gamma, \Lambda, g_s, h_s, c_s, \Delta w)$$

where $g_s$ is the gap, and $h_s$ is the overhang between the leading-edge slat and the main element. No specific formulation is proposed to model $A_G$ using geometric slat parameters in [Guo, 2012, 2010]. Sometimes, it may not be convenient to model slat noise on geometric parameters alone. Therefore, local flow variations can be taken into account in the $A_F$ parameter. Guo identified the following parameters to influence slat noise

$$A_F = A_F(\Omega_c, b_c, L_r, \Omega_w, b_w, m_g, C_L)$$

where $\Omega_c$ is the cove vorticity, $b_c$ the shear layer thickness, $L_r$ is the shear layer reattachment location relative to the slat trailing edge, $\Omega_w$ is the slat wake vorticity, $b_w$ is the thickness of the slat wake, $m_g$ is the mass flow through the slat gap and $C_L$ is the lift coefficient of the wing. The introduction of the $A_F$ parameter entails modeling geometric changes of the slat element to the far-field noise that would be captured in the flow quantities. Therefore, the precise modeling of the $A_G$ and $A_F$ factor can be arbitrarily chosen, depending on which parameters are identified as most important. Aircraft manufacturers can extract these primary parameters from the extensive datasets at their disposal. Unfortunately, such datasets are unavailable to the scientific community due to proprietary reasons. No specific models for the $A_G$ and $A_F$ factors can be found in the current scientific literature. In [Botero-Bolivar et al., 2020], we present an analysis of the
semi-empirical model proposed by Guo and use macroscopic flow parameters to model the (combined) factors $A_G$ and $A_F$. The flow parameters used were the dimensionless shear layer path length and the difference in the velocity magnitude inside and outside the shear layer. Fourteen slat geometries were simulated using the Lattice-Boltzmann Method in PowerFLOW, which provided the necessary dataset. The noise prediction model was also validated with experimental data.

An analytical noise prediction tool to model slat noise is proposed in [Molin and Roger, 2000; Molin et al., 2003; Molin, 2019]. This slat noise prediction model is based on Amiet’s leading-edge noise prediction model. Contrary to the general understanding of the slat noise mechanism, this model considers that the noise is primarily caused by the interaction of velocity fluctuations from the slat flow with the leading-edge of the main element. The far-field slat noise power spectral density is given as

$$S_{pp}(\vec{x}, \omega) = \left( \frac{\rho_0 \omega M_0 z^\frac{3}{2}}{\sigma^2} \right)^2 \frac{b^2}{2} |L(x, z, k_x, U_0)| l_y(\omega) S_{ww}(\omega)$$

where $l_y(\omega)$ is the spanwise correlation length of the incident turbulence on the leading-edge, $S_{ww}(\omega)$ is the power spectral density of the velocity fluctuations normal to the airfoil, and $M_0 = U_0/c_0$ where $U_0$ is the mean velocity of the turbulent flow arriving at the leading-edge of the main element. The inflow turbulence is then modeled using the Liepmann turbulence model, which requires three input parameters:

- $U_0$, the mean velocity of the turbulent flow arriving at the leading-edge of the main element
- $\Lambda_T$, the integral length scale of the inflow turbulence. This length scale corresponds to the largest eddies in the flow
- $w'^2$, the mean squared value of the velocity fluctuation normal to the leading-edge of the main element.

The slat noise generated by this leading-edge noise mechanism is considered to be generated by two contributions (see Fig. 2.13). The first contribution is from the large-scale vortical oscillations of the slat cove vortex, where the integral length scale $\Lambda_T$ is approximated by the size of the slat cove vortex and $w'^2$ is approximated at the center of the vortex. The second contribution is from the small-scale fluctuations in the shear layer in the slat cove, where the integral length scale $\Lambda_T$ is determined as the shear layer size,
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i.e., thickness and $w^2$ is approximated in the shear layer, e.g., where the velocity fluctuations are the highest. This slat noise prediction model has been successfully validated with an extensive noise database of Airbus aircraft [Molin, 2019].

![Figure 2.13: Slat noise mechanisms, adopted from [Molin and Roger, 2000].](image)

A model to predict the frequency of the slat tones that occur in low Reynolds number testing is proposed in [Terracol et al., 2016]. The slat tone frequencies are given as

$$f_n = n \frac{U_0}{L_a} \frac{1}{M_0 + \frac{\alpha l}{\kappa v}}$$  \hspace{1cm} (2.38)

where $L_a$ is the distance between the slat cusp and reattachment point of the shear layer in the slat cove, $\alpha l = L_v/L_a$ with $L_v$ the length of the shear layer path, and $\kappa v = U_v/U_0$ with $U_v$ the mean convection velocity in the shear layer in the slat cove. The frequencies of the slat tones can be predicted if $L_a$, $L_v$, and $U_v$ can be determined. The shear layer path and its reattachment can be readily found from a mean flow measurement. However, the mean convection velocity in the shear layer, $U_v$, is less straightforward to determine as it requires a two-point correlation measurement. Terracol proposes that the mean convection velocity in the shear layer can also be approximated from the velocity magnitude along the shear layer path. This prediction model was successfully validated in [Terracol et al., 2016] and others [Pascioni and Cattafesta, 2018; Li et al., 2017].
2.5 Aeroacoustic Measurements

Aeroacoustic assessment of airframe noise components can be done in several ways. Firstly, measurements of real aircraft in full flight can be performed using flyover measurements. Secondly, scaled aircraft models can be tested in wind tunnels where realistic flight conditions can be simulated in a controlled environment. These tests can be performed using fully scaled models, half (i.e., semi-span) models, or simplified models of specific components of the aircraft. Lastly, numerical simulations can be performed using computational aeroacoustics frameworks. In this section, we briefly review the methods for aeroacoustic assessment and discuss their role.

2.5.1 Flyover Measurements

Flyover measurements using microphone phased arrays can be used to localize and quantify the noise from real aircraft in the approach or take-off condition at an airport [Sijtsma and Stoker, 2004; Humphreys et al., 2016]. Figure 2.15 shows an example of a flyover measurement setup at Amsterdam Schiphol Airport. Flyover measurements can be used for, e.g., the validation of noise prediction models [Pott-Pollenske et al., 2006, 2002; Viera et al., 2022], the evaluation of noise reduction concepts [Takaishi et al., 2016; Yamamoto et al., 2019], the impact of meteorological conditions on noise measurements [Lockard and Bestul, 2018], the assessment of sound quality metrics [Viera et al., 2020] or the assessment of aircraft noise variability [Snellen et al., 2017]. Flyover measurements
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are primarily useful for validation purposes using real aircraft in-flight. However, this type of measurement is not a cost-efficient tool for aircraft manufacturers to evaluate new low-noise designs or configurations at an early design stage. Flyover measurements are complex and costly to perform and may not always provide the necessary details of specific flow or noise mechanisms.

Figure 2.15: Flyover measurements setup showing the microphone phased array, from [Merino-Martinez et al., 2016].

2.5.2 Wind Tunnels Testing

Aeroacoustic measurements performed in wind tunnels are a more cost-efficient tool that can be used, e.g., to validate low noise designs and to optimize an aircraft configuration in an earlier design stage. Wind tunnels come in many different shapes and sizes, depending on the purpose of their use.

Large-scale wind tunnels use fully scaled aircraft models which are sufficiently representative of simulating the flow physics of a real aircraft. This requires fidelity of the aircraft model and Reynolds number similarity of the flow. DNW’s large low-speed facility (LLF) (see Fig. 2.16) is an example of a large-scale wind tunnel facility. However, the costs of doing measurements in a large-scale facility are high. This is because technically sophisticated scaled models are used, and wind tunnel maintenance and operational costs are high. Therefore, large-scale wind tunnel tests are mainly used by the industry to optimize an aircraft design or configuration.
Wind tunnel tests using semi-span (i.e., half-span) aircraft models are typically performed in mid-sized wind tunnel facilities. Examples of such a facility are the subsonic tunnel at NASA Langley Research Center [Bahr, 2021] or the low speed wind tunnel (LSWT) operated by Airbus [Spehr and Ahlefeldt, 2019] (see Fig. 2.17 and 2.18). Wind tunnels of this size can also be made cryogenic, meaning that the pressure and density in the tunnel can be changed to increase the Reynolds number of the flow. The European Transonic Windtunnel (EWT) is an example of a cryogenic wind tunnel type [Spehr and Ahlefeldt, 2019]. A mid-sized wind tunnel facility is typically used by industry but is also more accessible for scientific purposes in large research projects.

Aircraft components, e.g., landing gears or high-lift wing models, are commonly measured in small-scale wind tunnels. For example, figure 2.19 shows the small-scale wind tunnel facility at the University of Twente with a high-lift wing model. Scientific institutes widely use small-scale facilities because fundamental flow physics can be studied in great detail. This allows researchers to develop and validate theoretical models and numerical simulations, often by using simplified geometries. Also, new noise reduction concepts can be studied efficiently and comprehensively in small-scale facilities.

Figure 2.16: Large-scale wind tunnel facility of DNW, from [Oerlemans et al., 2007].
2.5. Aeroacoustic Measurements

**Figure 2.17:** Semi-span aircraft model in the subsonic tunnel at NASA Langley Research Center, from [Bahr, 2021].

**Figure 2.18:** Semi-span aircraft model in the low speed wind tunnel (LSWT), from [Spehr and Ahlefeldt, 2019].

**Figure 2.19:** Small-scale anechoic wind tunnel facility of the University of Twente.

2.5.2.1 Test Section Configurations

Aeroacoustic measurements of airframe noise in wind tunnels can also be performed in different test section configurations. These are the hard-wall, open-jet, and hybrid test section types.
The Hard-Wall Test Section  Figure 2.20 illustrates the hard-wall test section type. This is the most common test section type used for aerodynamic measurements. In this configuration, the flow is bounded by solid walls. A boundary layer is present on the test section walls, and a wake forms behind the airfoil model, giving regions of rotational flow. These regions affect the irrotational flow in several ways requiring aerodynamic boundary conditions [Barlow et al., 1999]. Firstly, a buoyancy effect occurs, which results from a reduction of the cross-sectional area of the potential flow core because of boundary layer growth on the walls. This causes the velocity to increase along the test section, resulting in a lower static pressure downstream of the airfoil model than upstream. As a result, the measured drag force will be higher. For thin airfoils, this effect can be considered negligible because of the relatively small frontal area. Secondly, the effective cross-sectional area of the potential core also changes because of the presence of the airfoil and its wake, resulting in solid and wake blockage, respectively. This reduction in the cross-sectional area increases the flow velocity because of mass conservation. Consequently, the dynamic pressure used to calculate the aerodynamic coefficients must be corrected. Finally, the solid walls impose restrictions on the streamlines. This leads to an increase in the curvature of the streamlines compared to the free-flight case. The result is an upwash effect which leads to a small increase in the effective angle of attack compared to the geometric angle of attack. Fortunately, these boundary effects are widely studied and can be easily corrected with existing models [Barlow et al., 1999].

Figure 2.20: Schematic of a closed test section configuration.
Acoustic measurements can also be done in the hard-wall test section. However, a hard-wall test section is a sub-optimal acoustic environment for several reasons. Firstly, the area in which microphones can be mounted is typically limited, which constraints, e.g., the ability to do directivity measurements. Secondly, microphones installed on the side walls are exposed to pressure fluctuations from the boundary layer on the walls. This contaminates the microphone signals, which severely decreases the signal-to-noise ratio. Fortunately, the influence of the pressure fluctuations from the boundary layer can be reduced by special surface treatment and/or microphone recession [Jaeger et al., 2000; Sijtsma and Holthusen, 1999]. Finally, the solid walls of the test section reflect sound waves, which contaminate the noise measurements. A review of the relevant acoustic corrections is given in Sec. 3.3.5.1 along with correction methods that can be found in the current literature.

The Open-jet Test Section Figure 2.21 illustrates the open-jet test section type. The side walls are removed in this test section configuration giving an open jet stream. A shear layer is present at the interface between the wind tunnel stream and quiescent air, yielding increased flow unsteadiness. There is no buoyancy effect, and solid and wake blockage effects are smaller (about 25% compared to the CTS) because of the free expansion of the jet. This also makes the open-jet test section more convenient for aerodynamic measurements of bluff bodies. However, testing high-lift airfoil models will result in a strong deflection of the jet [Brooks and Marcolini, 1984]. This results in a smaller effective angle of attack compared to the geometric angle of attack. In addition, a more severe streamwise curvature effect is produced, which changes the effective camber shape of an airfoil model. This effect is generally less important but is relevant for high-lift airfoils in relatively small open-jet test sections.

Acoustic measurements are ideally performed in the open-jet test section configuration. The OTS can be placed in an anechoic chamber creating an acoustic free-field. A microphone can be placed away from the flow, thereby improving the signal-to-noise ratio. The placement of free-field microphones is less constrained so that directivity measurements can be performed. However, sound waves now need to propagate through the shear layer, which affects the noise measurement in several ways. Firstly, the acoustic path will be altered because of refraction, so that the pressure amplitude and time delay corrections need to be applied. Secondly, turbulent velocity fluctuations in the free shear layer will disturb the wavefronts propagating through. This leads to coherence loss in microphone phased array measurements and so-called haystacking of tonal noise.
A discussion of the acoustic propagation effects in an OTS and available correction methods is given in Sec. 3.3.5.1.

**Figure 2.21:** Schematic of an open-jet test section configuration.

**The Hybrid Test Section**  In recent years, the hybrid test section (HTS) has become increasingly popular for aeroacoustic research. Figure 2.22 illustrates this test section type. An HTS is created by replacing the solid walls of the CTS with a transparent acoustic material. Side walls of stretched Kevlar cloth are widely used for this purpose [Smith et al., 2005; Devenport et al., 2013]. Aerodynamic boundary corrections in an HTS involve several additional mechanisms which arise because of the permeability of the walls. These mechanisms are well described in [Brown, 2016]. The primary effect of wall porosity in a hybrid test section is an effective angle of attack reduction. However, the effective angle of attack reduction in the HTS is much smaller compared to the OTS. Comprehensive boundary corrections for a hybrid test section, including blockage corrections, can be determined with panel methods or ad-hoc methods, e.g., in [Devenport et al., 2013; Brown, 2016].

Hybrid test sections are specifically designed to include the capabilities of a CTS while having entirely acoustic transparent walls. Acoustic transparent walls allow microphones to be placed away from the test section, improving the signal-to-noise ratio. However, the acoustic windows can produce a considerable amount of self-noise at high frequencies, which can critically affect noise measurements [Devenport et al., 2013; Szoke et al., 2022]. The acoustic windows will also lead to noise transmission loss, but this can be simply
corrected. Caution must be taken as the transmission loss correction can depend on the incident angle of the sound waves, the amount of Kevlar cloth tension, and the flow velocity in the test section [Brown, 2016; Pascioni, 2017; Bahr et al., 2018]. Refraction corrections apply similarly to the open-jet test section. Coherence loss and boundary layer absorption are typically negligible for small-scale hybrid test sections operating at low velocities.

![Schematic of a hybrid test section configuration.](image)

**Figure 2.22:** Schematic of a hybrid test section configuration.

### 2.5.3 Comparability of Aeroacoustic Measurements Performed in Wind Tunnels.

#### 2.5.3.1 Cross-Facility Comparisons

Systematic sources of uncertainty make it challenging to compare aeroacoustic wind tunnel tests performed in different wind tunnel facilities. The only study available in the current literature which compares measurements of the same industrial model in two different wind tunnels is found in [Spehr and Ahlefeldt, 2019]. This study measures a half-span Airbus aircraft model in two mid-sized closed test section wind tunnels. The first facility is the LSWT wind tunnel operated by Airbus. This is an open-circuit wind tunnel with a test section of 2.1 m X 2.1 m, reaching a velocity up to 80 m/s. The second facility is the ETW transonic wind tunnel facility in Cologne, Germany. This is a closed-circuit wind tunnel with a test section of 2.0 m by 2.4 m. Measurements in
both wind tunnels were performed under atmospheric conditions. A microphone phased array comprising 144 microphones was used in the LSWT, whereas a microphone phased array comprising 96 microphones was used in the EWT. The microphone arrangement was different in both wind tunnels. Measurements were performed at the same free-stream Mach number and model Reynolds number. In general, this study found a good similarity in the beamforming results from both wind tunnel tests. However, significant differences were found below 4.3 kHz and above 35 kHz, while integrated spectra agreed within ±2 dB between 4.3 kHz and 35 kHz. The differences are suggested to be caused by the data acquisition, the array calibration, the array design, and the position of the search grid used in the beamforming. However, the exact cause(s) for the differences could not be isolated.

A cross-facility comparison of aeroacoustic measurements from a NACA-0012 airfoil model is often made in the literature [Oerlemans and Migliore, 2004; Devenport et al., 2013]. Sometimes this is referred to as a ‘benchmark validation’ for an aeroacoustic wind tunnel facility [Sarradj et al., 2009; Mayer et al., 2019]. However, when comparing beamforming measurements with the data in [Brooks et al., 1989], care must be taken. This is because there can be a significant disagreement between the coherent output power (COP) method used in [Brooks et al., 1989] and the beamforming integration technique [Bahr et al., 2011]. Figure 2.23 shows a cross-facility comparison of trailing edge noise measurements from a NACA-0012 airfoil in [Devenport et al., 2013]. The Virginia Tech wind tunnel measurements were performed in a hybrid (Kevlar-walled) test section with a cross-section of 1.83 m X 1.83 m using an airfoil model with a 0.2 m chord. An airfoil model with a 0.23 m chord was used in NLR’s KAT wind tunnel, which has a cross-section of 0.38 m X 0.51 m and has the open-jet test section type. The NASA measurements were performed in an open-jet test section wind tunnel with a cross-section of 0.3 m X 0.45 m using an airfoil model with a 0.23 m chord. Corrections were applied to account for setup-dependent parameters. The results shown in Fig. 2.23 show an acceptable agreement of the overall spectral shape. However, noise level differences up to 5 dB are still observed for this relatively simple airfoil at zero degrees angle of attack (i.e., no loading). This difference is attributed to the different tripping device strategies that were used [Oerlemans and Migliore, 2004]. Moreover, the measurements at Virginia Tech were performed in a hybrid test section which could also affect the comparability of results.
2.5. Aeroacoustic Measurements

Figure 2.23: Comparison of trailing edge noise measurements from a NACA-0012 airfoil with a tripped boundary layer measured at Virginia Tech (○) [Devenport et al., 2013], NLR (△) [Oerlemans and Migliore, 2004] and NASA BPM model experiments (○) [Brooks et al., 1989], from [Devenport et al., 2013]. The dashed line indicates the noise from a prediction model.

2.5.3.2 Test Section Configuration Comparisons

The type of test section configuration will also increase the uncertainty of aeroacoustic measurements. This is due to several reasons. Firstly, the different boundary conditions imposed by the test section on the flow can affect the aerodynamic behavior and, thereby, aeroacoustic characterization of models. These effects and their extent are not yet well understood. Secondly, no common framework exists to correct beamforming results to a standard free-field condition. Finally, limited research is available on this topic because not many wind tunnel facilities can modify their test section configuration into all types.

The comparability of microphone phased array measurements of open-jet and hard-wall test section measurements in a large-scale facility is given in [Oerlemans and Sijtsma, 2004; Oerlemans et al., 2007]. These measurements were performed in DNW’s LLF wind tunnel using a 1:10.6 scaled Airbus A340 aircraft model. A 4 m diameter microphone array with 140 microphones was used in the open-jet, whereas a 1 m diameter microphone array with 128 microphones was used in the hard-wall test section. In addition, the microphone arrays were located at a different distance from the model, which resulted in a different opening angle. This study found similar noise source characteristics in the test section configurations. Coherence loss in the open-jet was significant, reducing the integrated noise levels up to 10 dB. Relative noise levels, obtained by comparing configuration changes, were comparable within 1 dB between the open-jet and hard-wall test sections. This is provided that the flow conditions over the aircraft model are similar.
Aeroacoustic measurements performed in a mid-scale open-jet and hard-wall test section wind tunnel are compared in [Bahr, 2021]. A half-span high-lift common research model (CRM-HL) was used in this study. The microphone phased arrays used in both test section configurations were non-identical. Aerodynamically similar conditions were compared by matching the static pressure distributions. It was found that the hard-wall test section measurements were contaminated by background noise at lower frequencies, whereas the open-jet measurements suffered coherence loss in the high-frequency range. A deconvolution algorithm accounted for image sources in the CTS and coherence-loss in the OTS. It was found that including image sources in the deconvolution algorithm had a negligible effect. The coherence loss correction, on the other hand, significantly improved the beamforming results.

A comprehensive study on the comparability of open-jet and hard-wall test section measurements in a small-scale facility is given in [Kroeber, 2013]. Two different wind tunnels were used, namely the AWB (open-jet configuration) and the GroWiKa (hard-wall configuration). Several sound sources were considered in this study. These are a loudspeaker with monopole directivity characteristics, an elliptical plate with cavities, and a high-lift wing model. All sound sources were installed in each test section configuration using the same geometric setup, i.e., the same distance between the sound source and microphone array. It was found that coherence loss in the open-jet test section led to a significant decrease in integrated noise levels by 4 dB at the highest flow velocity measured (35 m/s). Unfortunately, no coherence loss correction model for microphone array measurements was available at the time of this study. The localization of sound sources in the closed test section was found to be primarily affected by reflection from the solid walls. This effect mainly occurred at low frequencies, but the overall influence of the reflections on source localization was found to be minimal. Overall, it was found that the maximum deviation between all sound sources used was 4.6 dB.

A comparison of slat noise measurements from a 30P30N high-lift model in a hard-wall and hybrid test section is given in [Murayama et al., 2018]. These measurements were performed in the JAXA’s LWT2 wind tunnel, which has a 2 m X 2 m test section. This wind tunnel test section can be configured as a CTS or HTS with Kevlar walls. The microphone phased array in the CTS comprised 96 microphones with a diameter of 1 m located in the side wall at 1 m from the sound source. The microphone phased array in the HTS comprised 96 microphones with a diameter of 1.5 m located at 1.2 m from the sound source. Aerodynamic similar conditions were compared by matching the $C_p$...
distribution in both test section configurations. Unsteady surface pressure measurements at various locations on the slat element showed a negligible difference between the two test section configurations. However, integrated noise levels from beamforming maps showed apparent differences between the CTS and HTS measurements. In the low-frequency range, absolute noise levels were found to be significantly lower (5 dB) in the HTS than in the CTS. This was attributed to a higher background noise level in the CTS. A good agreement was found between the integrated noise levels in the mid-frequency range with narrow band peaks more clearly visible in the HTS. In the high-frequency range, noise levels were found to be substantially higher in the HTS compared to the CTS by more than 10 dB in some cases. However, an explanation for this discrepancy was not given.

References


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Chapter 3

Aeroacoustic Facility and Methods

This chapter describes the aeroacoustic facility and experimental methods used to obtain the results shown in this thesis. We start by describing the aeroacoustic wind tunnel facility at the University of Twente in Sec. 3.1. This facility has undergone significant changes since the start of this PhD project. The anechoic chamber of the facility was refurbished at the start as part of this PhD project in 2018. Section 3.1.1 describes the characterization of the refurbished anechoic chamber. The original test section of the wind tunnel, dating from the 1970s, was replaced with an interchangeable test section, allowing for an open-jet, a closed (i.e., hard-walled), and a hybrid configuration. These configurations are described in Sec. 3.1.2, 3.1.3 and 3.1.4. Several beamforming algorithms have been developed (in-house) for microphone phased array measurements. These algorithms are described in section 3.3 and validated with several benchmark cases. Also discussed are the various airfoil models used in this thesis in Sec. 3.2. Finally, the setup of flow-field measurements with particle image velocimetry (PIV) is described in Sec. 3.4.

3.1 The Aeroacoustic Wind Tunnel Facility at the University of Twente

In its original configuration (before this PhD project), the aeroacoustic wind tunnel of the University of Twente was an open-jet, closed-circuit wind tunnel. The facility was initially built in the 1970s as a closed test section and closed-circuit facility. It was converted into an aeroacoustic wind tunnel in 2001 [Verhaagen et al., 2011] by adding an
anechoic chamber around the test section. The maximum free-stream flow velocity in the test section is approximately 60 m/s. However, depending on the flow blockage induced by the wind tunnel models, this maximum flow velocity may be lower. Typically, airfoil models with a chord up to 0.3 m are used, reaching a maximum chord-based Reynolds number of 1.2 million.

Figure 3.1: View inside of the anechoic chamber of the aeroacoustic wind tunnel at the University of Twente.

Figure 3.2 shows a schematic illustration of the current test facility. The wind tunnel is powered by a Rotavent centrifugal fan, type RZR 13-1250 of a double inlet with 12 backward-curved blades (1). The electric motor has a maximum power rating of 132 kW, a maximum rotational speed of 3000 RPM, and is controlled by a Frenic-Eco inverter. The motor is connected to the centrifugal fan by a V-belt. An active cooling system comprising a tube bank is installed downstream of the fan, ensuring a stable air temperature in the wind tunnel circuit (2). Splitter-type silencers are installed downstream and upstream of the centrifugal fan (3). These silencers attenuate the noise produced by the fan, which would otherwise contaminate the anechoic chamber. The upstream and downstream silencers consist of 12 parallel splitter walls with a thickness of 100 mm built from a mineral wool core covered with a perforated plate housing. The
distance between the walls is 125 mm. Corner vanes redirect the flow around the corners of the circuit. A low turbulence level flow in the test section is guaranteed by seven screens which are located upstream of the contraction. The first screen is in the wide-angle diffuser between (4), and (5). A second screen is placed in the settling chamber, followed by a honeycomb structure. Five more screens are located downstream of the honeycomb structure in the settling chamber. The contraction has a ratio of 10:1, accelerating the flow towards the test section. The hard-wall and hybrid test sections are located in (7). In contrast, the open-jet test section is further downstream. An anechoic chamber encloses the test section, measuring 6 m x 6 m x 4 m (width x length x height). The collector is embedded in the walls of the anechoic chamber. G+H wedge absorbers, type ASONAD 550, are used to cover most of the walls of the anechoic chamber. The walls around the collector are covered with 350 mm thick flat absorbers with perforated metal sheet surface, type G+H SONEX WF350. The wind tunnel walls between the collector and the turning vanes consist of acoustic baffles with mineral wool cores for noise abatement. A vent ensures atmospheric pressure inside of the anechoic chamber. There is no diffuser section downstream of the test section to assemble a fully closed circuit.

Figure 3.2: Schematic of the Aeroacoustic Wind Tunnel of the University of Twente.
3.1.1 The Anechoic Chamber

Anechoic chambers aim to provide the idealized free-field conditions required in aeroacoustic measurements. Qualifying an anechoic chamber involves defining the sphere’s radius where the free-field conditions are completely satisfied. The qualification test of an anechoic chamber is performed following the International Standard ISO 3745:2012 Annex A, “General procedures for qualification of anechoic and hemi-anechoic rooms” [International Organization for Standardization, 2012] for frequencies ranging from 10 Hz to 10 kHz. The performance of the anechoic chamber is assessed by evaluating the decrease of sound pressure level emitted from an omnidirectional monopole source with distance. The inverse square law \(1/r^2\) should then be observed in ideal free-field conditions.

The qualification of the anechoic chamber of the aeroacoustic wind tunnel of the University of Twente was done using three different noise sources, producing white noise. An external contractor was hired to perform these measurements. First, an electrodynamic loudspeaker was used for frequencies between 100 Hz and 500 Hz. This loudspeaker was built into a sound insulation box with a PVC tube of 100 mm diameter flanged onto it, see Fig. 3.3. Second, a horn driver type pressure chamber system was used for frequencies between 1 kHz and 2 kHz, connected to a tube of 34 mm as shown in Fig. 3.3. Last, the horn driver pressure chamber system was connected to a tube of 10 mm diameter and used for frequencies between 4 kHz to 10 kHz, as shown in Fig. 3.4. The relation between the acoustic wavenumber \(k\) and the duct diameter \(a\) was respectively \(ka = 0.92, 1.25, \) and \(1.84\) for the three noise sources. According to the ISO measurement practice, this suffices to guarantee that the tube opening can be considered an omnidirectional monopole sound source. The source was placed at the center of the anechoic chamber. A motor-driven winch moved the microphone with constant velocity away from the sound source. The microphone continuously recorded the sound level during the measurement procedure. The measured signal was analyzed in one-third octave bands.
3.1. The Aeroacoustic Wind Tunnel Facility at the University of Twente

Figure 3.3: Noise sources used in the anechoic chamber certification. Left: Electrodynamic loudspeaker built into a sound insulation box with the PVC tube. Right: Pressure chamber system connect to a tube of 34 mm diameter.

Figure 3.4: Noise sources adopted in the anechoic chamber certification. Pressure chamber system connected to a tube of 10 mm diameter.

Five measurement paths were considered crossing the center of the room, as shown in Fig. 3.5. Paths one, two, and five end in the three upper corners of the room, whereas path three ends in a wedge in the middle of the wall. Path four ends vertically below the sound source. The sound pressure level is recorded, starting from a 0.5 m distance from the sound source, whereas path four starts at approximately 0.25 m. This is done to avoid measurement contamination by near-field effects. The total distance from the sound source to the wall is 3.8 m for path one and two, 3 m for path three, 1.5 m for path four, and 4.15 m for path five. An overview of the measurement equipment that was used can be found in [de Santana et al., 2018].
Measured sound pressure levels

The sound pressure levels along the five measurement paths are shown in Fig. 3.6 for one-third octave bands with a center frequency of 160, 500, 2,000, 5,000, and 10,000 Hz. Figure 3.6 shows that the spherical spreading law is fully respected along paths one, two, and three for frequencies $>160$ Hz. Along path four, the spherical spreading law is not respected close to the ground floor, which is located at a 1.5 m distance from the sound source. The measurements along the other paths were not done so close to the walls of the anechoic chamber. Measurement path five was most critical for determining the cut-off frequency of the anechoic chamber. Along this measurement path, the spherical spreading law was observed for a distance from the source of 1.6 m and 2.1 m for 125 Hz and 160 Hz, respectively. In conclusion, the measurements showed that the largest free-field sphere of the anechoic chamber is 0.95 m, calculated from the height of the anechoic chamber and quarter wavelength of the cut-off frequency of 160 Hz, according to ISO 3745 Annex A. The free-field conditions are completely satisfied for a 1.3 m sphere from 160 Hz. This is mainly constrained by the vertical measurement path, i.e., path four. In most cases, the free-field radius extends widely beyond 1.3 m. Table 3.1 shows the recorded path lengths without exceeding the tolerances according to ISO 3745 Annex A.
3.1. The Aeroacoustic Wind Tunnel Facility at the University of Twente

![Figure 3.6](image-url)

**Figure 3.6:** Measured sound pressure level (SPL) along the five measurement paths (---) to the spherical spreading law (----) and the margin allowed in the ISO 3745 norm (——). SPL levels are shown for one-third octave bands.
Chapter 3. Aeroacoustic Facility and Methods

<table>
<thead>
<tr>
<th>Frequency (Hz)</th>
<th>Recorded path length without exceeding tolerances (m)</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>Path 1</td>
</tr>
<tr>
<td>100</td>
<td>1.4</td>
</tr>
<tr>
<td>125</td>
<td>1.8</td>
</tr>
<tr>
<td>160</td>
<td>2.7</td>
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<tr>
<td>250</td>
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<td>6300</td>
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<td>8000</td>
<td>2.6</td>
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<tr>
<td>10000</td>
<td>2.6</td>
</tr>
</tbody>
</table>

Table 3.1: Maximum deviations for the measured sound pressure level compared to the spherical spreading law, according to the ISO 3745 norm.

3.1.2 The Hard-Walled Test Section (CTS)

Different test section types can be configured in the anechoic chamber of the aeroacoustic wind tunnel facility at the University of Twente. The first is the hard-wall test section, shown in Fig. 3.9, measuring 1.8 m x 0.9 m x 0.7 m (length x width x height). The original test section of the wind tunnel was a wooden test section made several years ago in the 1970s. Therefore, a new closed, i.e., hard-walled test section, was designed within the framework of this PhD project to fulfill the requirements to perform state-of-the-art aeroacoustic and aerodynamic measurements. Figure 3.8 shows the steel adapter, which was made so that a standardized and well-defined flange can be used to connect the test section. A filler material was applied to make the original wooden section of the wind tunnel flush with the steel adapter. The new hard-walled test section was subsequently designed as a modular test section with transparent walls to allow optical access. Figure 3.9 shows the test section. The test section’s frame comprises standardized aluminum profiles using components from Rose and Krieger. This yields an easily adaptable base structure so that new experimental setups can be mounted with little effort. Airfoils are vertically installed inside turntables with an inner diameter of approximately 0.67 m. A standardized mounting system was designed
and made for the turntables. We designed this system considering that adjustments should be made using the machine tools from the workshop next door. The sidewalls of the test section comprise three acrylic panels. These acrylic panels are clamped to the corner fillets using toggle clamps. The corner fillets account for boundary layer growth and secondary flow in the corners of the test section. Hot-wire anemometry was used to measure the turbulence intensity at the center of the hard-wall test section. We measured a turbulence intensity <0.04% at 30 m/s and 50 m/s. A high-pass filter with cut-off frequency $f_c = U_0/(2L)$ was applied where $L$ was the test section height [Pascioni et al., 2014].

![Figure 3.7: Schematic of the closed, i.e. hard-walled test section.](image1)

![Figure 3.8: Contraction adapter](image2)

![Figure 3.9: Hard-wall test section during construction](image3)
3.1.3 The Open-Jet Test Section (OTS)

The open-jet test section is placed downstream of the hard-walled test section, measuring 2.0 m x 0.9 m x 0.7 m (length x width x height). This is done so that the open-jet test section is closer to the collector, reducing the total displacement of a deflected jet. The top of the test section is supported using an overhang structure. This avoids using a vertical support structure that could interfere with a sound field. The top and bottom plates comprise lightweight sandwich plates made of foam and galvanized aluminum. We initially designed these plates so that flat plate absorber-type plates could replace them at a later stage for sound absorption. Airfoils are mounted to turntables at the top and bottom with an inner diameter of 1 m.

![Figure 3.10: Schematic of the open-jet test section.](image1)

![Figure 3.11: The open-jet test section.](image2)
3.1. The Aeroacoustic Wind Tunnel Facility at the University of Twente

Flow Field Characterization

We comprehensively characterized the open-jet test section after it was commissioned. Hot-wire anemometry measurements were performed to characterize the flow uniformity and shear layer. Due to limitations of the traverse system, only half of the width of the open-jet could be characterized. The Streamware Pro Frame from Dantec is used to perform the measurements. We used two hot-wires probes: a single-wire probe, type 55P15 from Dantec, and an x-wire probe, type 55P51 from Dantec. The single-wire probe was sampled for 10 seconds at a rate of 25.6 kHz with a low-pass filter of 10 kHz. A Pitot-static tube with a system accuracy of approximately 0.25% was used as the reference to calibrate both hot-wires. The single-wire probe was sampled for 10 seconds at 25.6 kHz, whereas the x-wire probe was sampled for 5 seconds at 20 kHz. A low-pass filter of 10 kHz was applied to the data from both probes. We calibrated both probes in a velocity range from approximately 10 to 60 m/s. The directional calibration of the x-wire probe was performed using a K10CR1/M Motorized Rotation Mount at 19 angles ranging from -45° to 45°. Temperature fluctuation effects on the CTA voltage are corrected using King’s law. The active cooling system of the wind tunnel was used to keep the temperature fluctuations within ±1° Celsius during the measurements. This kept the temperature within a valid range for the temperature correction.

A generic characterization of the flow field in the open-jet test section is performed at a free-stream velocity of 30 m/s using the 55P15 single wire probe from Dantec. The flow uniformity was measured in a volume ranging from $x/b = 0.2$ to 2.4, $z/b = 0$ to 1.3 and $y/b = -0.55$ to 0.55 where $b = 0.45$ m is half the test section width and $h = 0.35$ m is half the test section height. Figure 3.12a shows a contour slice in the x-z plane of the streamwise velocity component ($u_x$) at $y/b = 0$. It was found that the non-uniformity of the streamwise flow velocity is <1.5% within the potential core. The reference velocity $U_0$ is defined as the mean streamwise velocity measured at $(x/b,y/b,z/b) = (0.2,0,0)$. Figure 3.12a shows that the highest non-uniformity is measured at the origin of the shear layer. This is likely due to an induced z-component of the velocity caused by jet entrainment. The turbulence intensity of the streamwise velocity component ($TI_x = u'_x/U_0$) is shown in Fig. 3.12b revealing the development of the shear layer.

Figure 3.13a shows the free-stream turbulence intensity ($TI_x$) in the open-jet for a range of flow velocities at the location $(x/b,y/b,z/b) = (0.2,0,0)$. A high-pass filter was applied to exclude bulk oscillations, following [Pascioni et al., 2014]. The cut-off frequency was
defined as \( f_c = \frac{U_0}{2L} \) with \( L \) the test section height. This filtered frequencies below 35 Hz at the highest velocity. Figure 3.13b also shows the streamwise development of the free-stream turbulence intensity in the center of the test section. The turbulence intensity is shown to increase in the downstream direction. We typically mount airfoil models at \( 1.1 < x/b < 1.8 \).

\[
\begin{align*}
\text{(A) Contour slice of } u_x/U_0 \text{ in x-z plane at y/b=0.} & \quad \text{(B) Contour slice of } T_I_x \text{ in x-z plane at y/b=0.} \\
\text{Figure 3.12: Open-jet flow field measurements.}
\end{align*}
\]

\[
\begin{align*}
\text{(A) Measured at the location } (x/b, y/b, z/b) = (0.2, 0, 0). & \quad \text{(B) Streamwise development of } T_I_x \text{ at y/b = 0 and } z/b = 0. \\
\text{Figure 3.13: Free-stream turbulence intensity in the open-jet test section (high-pass filtered).}
\end{align*}
\]

We further characterized the shear layer using hot-wire anemometry. Most importantly, we were interested in the spreading rate of the shear layer and its velocity profiles. These
quantities are useful for, e.g., modeling coherence loss effects [Biesheuvel et al., 2019].
First, the streamwise velocity profile was measured. Figure 3.14a shows the measured $u_x$ velocity profile at five different streamwise locations. Assuming self-similarity, we scaled the velocity profiles and compare it to the Görtler velocity profile [Görtler, 1942] given as

$$\frac{u_x}{U_0} = \frac{1}{2}[1 + erf(\xi - \xi_0)]$$

where $erf$ is the error function, $\xi = \sigma \frac{z}{x}$ with $\sigma$ the spreading rate, and $\xi_0$ is a lateral offset parameter accounting for the outward drift of the shear layer. Figure 3.14b shows the scaled $u_x$ velocity profiles. The velocity profiles were found to match with the Görtler velocity profile with $\sigma = 14$ and $\xi_0 = 0.1$. This corresponds to a total spreading angle of approximately 8 degrees. The velocity profile at $x/b = 0.2$ deviates because of the boundary layer mixing from the test section upstream of the open-jet.

The spreading rate of the shear layer was calculated by extracting the shear layer thickness along the streamwise direction. There are two methods to estimate the shear layer thickness [Sijtsma et al., 2014]. Firstly, the velocity profile can be integrated to get the momentum thickness $\delta^*$. The boundary layer thickness can then be estimated by assuming $\delta = 5.83 \times \delta^*$. Secondly, the shear layer can be defined as the distance between the location where $u_z = 0.05 * U_0$ and $u_z = 0.95 * U_0$. Both methods were used to determine the shear layer thickness and gave similar results. Figure 3.15 shows the measured shear layer thickness determined with the second method. The spreading rate $\sigma$ was found to be approximately 14. Note that the shear layer thickness does not start as 0 if extrapolated. This is because of the boundary layer from the test section upstream of the open-jet.

![Figure 3.14: Shear Layer Properties.](A) $U_0 = 30$ m/s, $y/b = 0.$ (B) $U_0 = 30$ m/s, $y/b = 0$, $\sigma = 14$, $\xi_0 = 0.1.$
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Figure 3.15: Streamwise development of the shear layer thickness.

Figure 3.16 shows the turbulence velocity spectra measured at the center of the shear layer at several streamwise locations. The spectra are normalized with the measured shear layer thickness. This normalization reveals that the largest eddies in the energy-containing range are proportional to the shear layer thickness. The Kolmogorov -5/3 scaling law can be observed in the inertial subrange. Most importantly, we do not observe any periodic vortex shedding, which would be visible as peaks in the spectra.

Figure 3.16: Normalized turbulence velocity spectra $\Phi_{uu}$ (PSD) at $y/b = 0$ and $z/b = 1$. The frequency is normalized with the measured shear layer thickness.

Figure 3.17 shows the measured turbulence intensity profiles of the streamwise ($u_x$) and lateral ($u_z$) velocity component. The x-wire probe (type 55P51 from Dantec) was used for these measurements. The lateral velocity profile is an important input parameter
used to model the coherence loss effect on microphone phased array measurements. This will be discussed in Sec 3.3.5.1. The results confirm the self-similarity of the turbulence intensity profiles.

![Turbulence intensity profiles](image)

Figure 3.17: Turbulence intensity profiles with $U_0 = 30 \text{ m/s}$ at $x/b = 1.4$, $y/b = 0$ and $\sigma = 14$.

### 3.1.4 The Hybrid Test Section (HTS)

A hybrid test section configuration is created by replacing the sidewalls of the CTS with Kevlar panels. These panels comprise in-house developed tensioning frames with stretched Kevlar cloth. A schematic cross-section of the tensioning frame is shown in Fig. 3.19a. Plain weave Kevlar cloth with 0.12 mm thickness and a specific weight of 61 g/cm$^2$ is used, supplied by HP-Textiles. We typically tension the cloth to approximately 1500 N/m at the center of the panel. A STM-50 tension meter is used to measure the tension. The Kevlar cloth type and tensioning value are similar to that used in [Devenport et al., 2013] and others [Bahr et al., 2021; Mayer et al., 2019]. In the current setup, three separate panels are used on one side of the test section, whereas a single panel is used on the opposite side, see Fig. 3.19b. The arrangement of these panels can be changed depending on test requirements. Toggle clamps are used to fix the panels in the test section to the corner fillets.
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Figure 3.18: Schematic of the hybrid (kevlar-walled) test section.

Figure 3.19: Hybrid test section configuration.

3.2 Airfoil Models

3.2.1 NACA-0012 and NACA-0018

Several airfoil models are used throughout this thesis. A NACA-0012 and a NACA-0018 airfoil with a 200 mm chord and a 700 mm span were used for measurements shown in Chapter 4. No pressure taps were located on these models. We only performed measurements with these airfoils at an angle of attack of 0°. Both airfoils were instrumented with 6 Knowles FG-23329-P07 microphones near the trailing-edge. The microphones were recessed behind a 0.3 mm pinhole beneath the airfoil surface (see Fig. 3.25). The microphones were attached to the airfoil by applying an adhesive to the back of the
microphones. Therefore, the pressure vents located on the side of the microphones can still operate because of the small tolerances left between the airfoil and microphone. The microphones were spaced 3 mm apart in the streamwise and spanwise direction. The microphones closest to the trailing-edge were at the chordwise position of \( x/c = 0.93 \) in the center span of the model. The FG-23329-P07 microphones were sampled with a NI PXIe-4499 Sound and Vibration modules for 30 seconds at 65,536 Hz. We used a zig-zag trip of 0.5 mm height \((70^\circ \text{ and } 6 \text{ mm width})\) at 5% of the chord on the lower and upper side of the airfoil.

### 3.2.2 NACA-63018

The NACA-63018 airfoil model was borrowed from TU Delft and had a 200 mm chord and 700 mm wet span. We use this model for the measurements presented in Chapter 4. A total of 28 pressure taps are installed on the model, distributed along the chord as shown in Fig. 3.24a. These pressure taps allowed us to set the geometric angle of attack in each test section to the correct effective angle of attack in-situ. The static pressure was measured with three pressure scanners (Model 9116 and 9216 by Netscanner Systems) with a pressure range of 1 psi and a system accuracy of 0.05% of the full scale. The static pressure was sampled with a frequency of 50 Hz for 10 seconds. We installed microphones near the trailing-edge by replacing the original trailing-edge section with a 3D printed trailing-edge. Four analog MEMS (micro-electro-mechanical systems) microphones (Invensense ICS-40300) were installed on a 3D printed insert. These microphones were recessed by 1 mm behind a 0.7 mm pinhole. The microphones were spaced 5 mm apart in the streamwise and chordwise direction and are in an L-shaped arrangement (see Fig. 3.25). This arrangement was preferred to minimize the number of installed microphones because of spatial limitations while still allowing the estimation of the streamwise and spanwise correlation length. Unfortunately, the center-most microphone was defective in the measurement and could not be replaced. The microphones closest to the trailing-edge were at a chordwise position of \( x/c = 0.86 \). A NI PXIe-4499 Sound and Vibration module samples the MEMS microphones for 30 seconds at 65,536 Hz. We used a zig-zag trip of 0.5 mm height \((70^\circ \text{ and } 6 \text{ mm width})\) at 5% of the chord on both the lower and upper side of the airfoil.
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Calibration Method for the Wall-Pressure Microphones

The wall-pressure microphones were calibrated using the in-situ procedure described in [Mish, 2003]. This method compares the measured pressure signal from a given noise source to the measured pressure signal of the microphone which is being calibrated. Therefore, a calibration device should be used which ensure that both microphones are exposed to the same pressure signal. Figure 3.20 illustrates the calibration device that we used. A loudspeaker (Visaton FR8) is mounted to a cone and a short tube to ensure plane wave propagation up to a sufficiently high frequency. The reference microphone (GRAS 40PH) is mounted perpendicular to the main tube, whereas the opening of the main tube is placed over the wall pressure microphones. A small rubber ring is placed at this opening to ensure a proper seal.

![Figure 3.20: Schematic of the calibration device for wall pressure microphones.](image1)

![Figure 3.21: Measured frequency response calibration of a MEMS microphones (Invensense ICS-40300).](image2)

We performed the calibration in the following steps. Firstly, the sensitivity of the reference microphone was calibrated using a GRAS 40AG Pistonphone, which excited a 1 kHz tone with 94 dB. Secondly, the sensitivity of the calibration microphone was determined by placing the calibration device over the microphone and exciting the speaker with the 1 kHz reference tone. The sensitivity given by the manufacturer was then used as a reference verification of the obtained calibration value. Thirdly, the frequency response of the calibration microphone was calibrated by exciting the speaker with white noise. The transfer function between the reference and calibration microphone can then be found by comparing the auto-power spectrum of both microphones. Finally, the coherence between both microphone signals was used to quantify the reliability of the calibration measurement. The typical frequency response of an Invensense ICS-40300

...
MEMS microphone (relative to the reference microphone) is shown in Fig. 3.21. The typical frequency response is $<1$ dB in a range of 100 Hz to 10 kHz. This was as expected from the manufacturer’s specifications for this type of microphone.

3.2.3 DU97W300

The DU97W300 airfoil model has a chord of 250 mm with a wet span of 700 mm and was used for the measurements presented in Chapter 4. A total of 47 pressure ports were located along the chord of the model, enabling accurate measurement of the pressure distribution and calculation of the section lift by trapezoidal integration. The pressure ports were placed at an angle of 12° with respect to the flow. Figure 3.24b shows the pressure port locations on this model. The static pressure was measured with three pressure scanners (Model 9116 and 9216 by Netscanner Systems) with a pressure range...
of 1 psi and a system accuracy of 0.05% of the full scale. The static pressure was sampled with a frequency of 50 Hz for 10 seconds. Further details about this model and its design can be found in [de Valk, 2019]. We used no tripping devices on this model.

![Figure 3.25: Airfoil instrumentation.](image)

### 3.2.4 30P30N

A multi-element high-lift airfoil model is used for the measurements in Chapter 5 and 6. The geometry is the 30P30N, which is used as a common research model by the scientific community. This high-lift device geometry is based on the McDonnell Douglas DC-10 aircraft, which flew commercially from 1968 to 1988. Today the geometry is dedicated to the Benchmark for Airframe Noise Computations (BANC) workshop. The high-lift device comprises three elements (see Fig.3.26); the leading-edge slat, main element and flap. The deflection angle of the slat $\delta_s$ and flap $\delta_f$ are 30 degrees with respect to the chord line of the stowed configuration. The slat chord length $c_s$ is equal to $0.15c_{\text{stowed}}$ and the flap chord length $c_f$ is equal to $0.30c_{\text{stowed}}$.

The 30P30N wind tunnel model was manufactured by the German wind tunnel model manufacturer DeHarde GmbH. The main and flap element were made out of aluminum (EN AW-7022, T651), whereas the slat element was made of steel (RAMAX HH). The airfoil was inspected after manufacturing using a coordinate measuring machine which found that the fully assembled model was within 0.2 mm of design specifications. The model is based on the so-called tunnel configuration geometry of the BANC workshop. A few modifications were made to the geometry to meet manufacturing requirements for the model. The slat cusp was shortened by $\approx 1$ mm yielding a thickness of 0.5 mm.
In addition, the trailing edge of the main element was shortened by \( \approx 3.5 \text{ mm} \). This yielded a gap \( g_f \) and overlap \( o_f \) between the main element and flap of \( 2.00\% c_{stowed} \) and \(-0.92\% c_{stowed} \). This differs from the original reference geometry, where the gap and overhang is \( 1.27\% c_{stowed} \) and \( 0.25\% c_{stowed} \) respectively. The trailing edge thickness of the flap and slat were unchanged with respect to the reference geometry. The trailing-edge thickness was 1.75 mm on the flap and 0.5 mm on the slat. This yielded a gap \( g_s \) and overlap \( o_s \) between the slat and main element of \( 2.95\% c_{stowed} \) and \( 1.27\% c_{stowed} \).

A total of 4 (steel) brackets for each element were used to minimize deflection based on a finite element analysis. The bracket placement is shown in Fig. 3.27. The center span was kept clear of brackets such that a 'clean' wing was exposed to the potential core of the wind tunnel. This allowed for a clean noise characterization using beamforming techniques. Furthermore, the slat brackets were located on the suction side of the model. Two dummy slat bracket interfaces are also available in the slat cove to study slat bracket effects. The wing model enabled sweep angle measurements between \( 0^\circ \) and \( 30^\circ \). Two sets of brackets were therefore made for the \( 0^\circ \) and \( 30^\circ \) sweep angle. This was done to ensure that flow-aligned brackets could be used. The stowed chord length of the model was 300 mm, and the span is 1036 mm. The blockage ratio is approximately 5% based on the frontal area at \( 0^\circ \) angle of attack.

**Instrumentation**

The 30P30N model is equipped with 84 pressure ports (see Fig. 3.27 and 3.28). The slat, flap, and main element contain 10, 14, and 60 pressure ports, respectively. These pressure ports have a diameter of 0.3 mm and have a chordwise inclination angle of \( 15^\circ \) with respect to the direction perpendicular to the leading edge. Pressure ports were also placed along the span of the model to quantify the flow uniformity. The 84 pressure taps were sampled using six pressure scanners (Model 9116 and Model 9216 by NetScanner Systems). The pressure range is 1 psi on four pressure boxes and 2.5 psi on the two other pressure boxes because a higher pressure range was needed for the suction peak on the main element. Both pressure scanner models have a system accuracy of \( \pm 0.05\% \) of the full scale. The static pressure was sampled at 50 Hz for 10 seconds.
Figure 3.26: The 30P30N high-lift geometry.

Figure 3.27: The pressure orifice locations (●) on the 30P30N wind tunnel model.

Figure 3.28: 30P30N pressure port locations.

(A) Chordwise pressure port locations (side-view) ●. (B) Spanwise pressure ports (top-view) ● and wind tunnel walls (---).

Noise Treatment

Unwanted noise was generated by the slat brackets and at the wall junction [Murayama et al., 2018]. Noise treatment was therefore applied to the slat brackets and wall junction of the slat to reduce the production of unwanted noise. The slat brackets were treated with the soft side of one-sided velcro, which has a 2.5 mm height (shown in Fig. 3.29a). This material was applied to the sides of the slat brackets and the inner part of the slat brackets, which were exposed to the flow from the slat cove (see Fig. 3.29b). The slat
cove near the wall junction was treated with a so-called 'faux fur' material of 12 mm height. We found that this noise treatment significantly reduced the unwanted noise sources from the model. Figure 3.30 shows the effect that the noise treatment had on a typical beamforming map from a 30P30N acoustic measurement. A substantial noise reduction from the slat brackets and wall junction was observed across the entire frequency range. Without this noise treatment, it would have been troublesome if not impossible to isolate the noise from the slat element with this microphone array arrangement.

![Figure 3.29: The noise treatment of the 30P30N wind tunnel model.](image)
Figure 3.30: Effect of the noise treatment on the one-third octave beamforming maps for the 30P30N. Without noise treatment (left) and with noise treatment (right). $Re_e = 1 \cdot 10^6$ and $\alpha_e = 5.5^\circ$ ($\alpha_g = 19.5^\circ$).
3.3 Microphone Phased Array Measurements

Microphone phased array measurements have nowadays become the standard technique for aeroacoustic measurements. However, while the technique is widely used, it is far from straightforward to be used. This section gives a detailed description of the technique and its implementation and validation.

3.3.1 Basics of Conventional Beamforming

Time Domain Beamforming

The most basic and straightforward beamforming technique is the delay-and-sum technique. If an array with $N$ microphones measures a sound source of amplitude $a$ in location $\xi_s$, the pressure signals measured by the microphone with index $n$ and location $\vec{x}_n$ will be

$$p_n(\vec{x}_n, t - \tau_n) = a(\vec{\xi}_n, t)$$

where $\tau_n$ will be the time delay between the source and microphone observer, which will depend on the distance between them and the speed of sound. The delay-and-sum beamforming method uses the time delay information between each microphone output signal. However, a search grid must be defined because the sound source location is unknown beforehand. To locate a sound source, the pressure signals from each microphone are back-propagated to each search grid point $\xi_j$, and all microphone signals are summed

$$z_j(t) = \frac{1}{N} \sum_{n=1}^{N} v_n \cdot 4\pi r_{j,n} p_n(t - \tau_{j,n})$$

where $p_n$ is the measured signal of microphone $n$. The $4\pi r_{j,n}$ term takes into account the reduction in pressure amplitude due to spherical spreading. The time delay $\tau_{j,n}$ can be written in the form of

$$\tau_{j,n} = \frac{r_{j,n}}{c_\infty}$$
where $c_\infty$ is the speed of sound and $r_{j,n} = \|\vec{x}_n - \vec{\xi}_j\|$ the distance from the search grid point to microphone $n$. The microphone signals may also be multiplied by a weight factor $v_n$. This weight factor can be introduced to improve the beamformer’s performance which will be discussed later on. Note that the beamforming formulation makes two important assumptions. The first assumption is that the sound source is considered a monopole and radiates equally in all directions. Second, in case multiple sound sources are present, it must be assumed that their pressure waves are incoherent. This means that they the pressure waves have a completely random phase relation such that no interference between the pressure waves occurs.

Consider the case of the two microphones shown in Fig. 3.31. Firstly, we calculate the distance between the individual microphones and the search grid location $\xi_j$. Secondly, each microphone signal is shifted in time according to the time delay it takes a sound wave to reach each microphone. If a sound source is present in grid point $\xi_j$ (for example, in $\xi_s$ here), then the time-shifted microphone signals will be in phase, and their sum will have a large amplitude. If a sound source is not present in grid point $\xi_j$, then the time-shifted microphone signals will be out of phase, and their sum will have a small amplitude. A final source-map is obtained by displaying the calculated summations in all grid points. The delay and sum method for noise source localization is equivalent to the working principle of an optic lens found in, e.g., telescopes used in astronomy. Therefore, advanced beamforming algorithms used in aeroacoustics are often based on algorithms found in the astronomy field. We will discuss a few of these algorithms later on.
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**Figure 3.31:** Delay and Sum Beamforming.

**Frequency Domain Beamforming - Conventional Beamforming**

Conventional Beamforming in the frequency domain is based on the Fourier transform of Equation 3.2, given by

\[
Z_j(\omega) = \mathcal{F} \left( \frac{1}{N} \sum_{n=1}^{N} v_n 4\pi r_{j,n} p_n(t - \tau_{j,n}) \right) \quad (3.4)
\]

\[
Z_j(\omega) = \frac{1}{N} \sum_{n=1}^{N} v_n 4\pi r_{j,n} \tilde{p}_n(\omega) e^{-i\tau_{j,n}\omega} \quad (3.5)
\]

where \(\mathcal{F}\) is the Fourier operator and \(\tilde{p}_n\) the discrete fourier transform of microphone signal \(p_n\). A simpler form of Equation 3.5 is obtained when we rewrite it to the form

\[
Z_j(\omega_k) = w_{j,n}^\dagger(\omega_k) \tilde{p}_n(\omega_k) \quad (3.6)
\]
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Where $[\cdot]^\dagger$ is the Hermitian transpose operator, $k$ the frequency index number and

$$
\tilde{p}_n(\omega_k) = \begin{bmatrix}
\tilde{p}_1(\omega_k) \\
\vdots \\
\vdots \\
\tilde{p}_N(\omega_k)
\end{bmatrix}.
$$

(3.7)

The vector containing the weighting factors and phase delay information is called the weighted steering vector and is defined as

$$
w_{j,n}(\omega_k) = \begin{bmatrix}
v_1 \frac{4\pi r_j N}{\lambda} e^{-i2\pi f_k \tau_{j,1}} \\
\vdots \\
\vdots \\
v_N \frac{4\pi r_j N}{\lambda} e^{-i2\pi f_k \tau_{j,N}}
\end{bmatrix}.
$$

(3.8)

Since the microphones signals are recorded discretely, there will be a finite number of frequency bins that can be detected and we can simply write

$$
Z_{j,k} = w_{j,n,k}^\dagger \tilde{p}_{n,k}.
$$

(3.9)

The power of the output signal in search grid point $j$ for frequency bin $k$ can be calculated with $P_{j,k} = |Z_{j,k}|^2$. Since $Z_{j,k}$ is complex we can write

$$
P_{j,k} = Z_{j,k} Z_{j,k}^*,
$$

(3.10)

$$
P_{j,k} = (w_{j,n,k}^\dagger \tilde{p}_{n,k}) (w_{j,n,k}^\dagger \tilde{p}_{n,k})^*,
$$

(3.11)

$$
P_{j,k} = w_{j,n,k}^\dagger (\tilde{p}_{n,k} \tilde{p}_{n,k}^\dagger) w_{j,n,k}.
$$

(3.12)

Equation 3.12 is the expression used in conventional beamforming in the frequency domain. The matrix $(\tilde{p}_{n,k} \tilde{p}_{n,k}^\dagger)$ is also called the Cross Spectral Matrix (CSM) and is of significant importance

$$
\text{CSM}_{n,n,k} = \begin{bmatrix}
\tilde{p}_{1,k} \tilde{p}_{1,k}^\dagger & \cdots & \tilde{p}_{1,k} \tilde{p}_{N,k}^\dagger \\
\vdots & \ddots & \vdots \\
\tilde{p}_{N,k} \tilde{p}_{1,k}^\dagger & \cdots & \tilde{p}_{N,k} \tilde{p}_{N,k}^\dagger
\end{bmatrix}.
$$

(3.13)
3.3. Microphone Phased Array Measurements

Care must be taken when estimating the CSM such that the root mean square value of the microphone signals is used. Eq. 3.13 is given in terms of the squared magnitude

\[
\text{CSM}_{n,n,k} = |\hat{p}_{n,k}\hat{p}_{n,k}^*|^2
\]  

(3.14)

where \( \hat{p}_{n,k} \) is the discrete Fourier transformed pressure signal from microphone \( n \) at frequency \( k \). The root mean square value of the CSM can be obtained by considering that \( \hat{p}_{\text{rms}} = \hat{p} / \sqrt{2} \) from which it follows that

\[
\text{CSM}_{n,n,k} = \left| \frac{\hat{p}_{n,k}\hat{p}_{n,k}^*}{\sqrt{2} \sqrt{2}} \right|^2 = \frac{1}{2} |\hat{p}_{n,k}\hat{p}_{n,k}^*|^2. 
\]  

(3.15)

Equation 3.12 is often written in a simplified form (omitting subscripts) as

\[
\mathbf{A} = \mathbf{w}^\dagger \mathbf{Cw} \quad [\text{Pa}^2 \text{m}^2] 
\]  

(3.16)

where \( \mathbf{A} \) is the output power of the sound source, \( \mathbf{w} \) the weighted steering vector (Eq. 3.8) and \( \mathbf{C} \) the (root mean squared) CSM from Eq. 3.15. Note that this expresses the source power at the search grid location \( \xi_j \) and not at the observer location (which follows from the units of the beamformer).

The output of the beamformer can be further optimized by choosing alternative forms of the steering vector. A commonly used weighting of the steering vector is

\[
\mathbf{w} = \frac{\mathbf{g}}{\|\mathbf{g}\|^2} 
\]  

(3.17)

which is obtained from a least-squares minimization approach to the problem where

\[
\mathbf{g} = g_{j,n}(\omega_k) = \frac{1}{4\pi r_{j,n}} e^{-i2\pi f_k r_{j,n}}. 
\]  

(3.18)

Other steering vector formulations that are widely used are discussed in [Sarradj, 2012].

The weighting vector in Eq. 3.18 follows from a constraint that maximizes the beamformer output when the search grid point is on the actual sound source location which occurs when the steering vector is parallel to the actual propagation vector. This minimization problem can be defined as [Sijtsma, 2004]

\[
J = \|\mathbf{p} - a\mathbf{g}\|^2 
\]  

(3.19)
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where $p$ is the measured pressure vector from the microphones, $a$ is the (complex) sound source amplitude and $g$ is the steering vector. The solution to this problem is given as

$$a = \frac{g^*p}{\|g\|^2}$$  \hspace{1cm} (3.20)

and the output power of the beamformer then becomes

$$A = \frac{g^*Cg}{\|g\|^4} \quad [\text{Pa}^2\text{m}^2].$$  \hspace{1cm} (3.21)

It is the expression in Eq. 3.21 which is used for the sound mapping throughout this thesis. Other weighting factors can be applied that further optimize the spatial filter of the beamformer such as corrections for microphone density and effective aperture [Sijtsma, 2010]. However, these methods have not been applied here.

Spectral Estimation

An essential step in CBF is the computation of the CSM. The spectral estimation of the CSM in Eq. 3.13 can be performed in different ways. Here we have used the Welch’s method, also commonly referred to as the Welch Overlapped Segment Averaging (WOSA) method [Trobs and Heinzel, 2006]. A description of the WOSA method is given in Appendix 3.4. We use a constant frequency resolution. The specific parameters used in the spectral estimations shown in this thesis will use different parameters depending on the application. They will be further specified when these results are discussed.

Diagonal Removal

Acoustic measurements in wind tunnels often have a low or negative signal-to-noise-ratio (SNR). This is because the noise generated by e.g., an airfoil is often smaller than the background noise of the wind tunnel. A very poor SNR is most common in closed test section measurements where the microphones are installed in the sidewalls of the wind tunnel. The sidewall installation exposes the microphones to the pressure fluctuations from the boundary layer on the test section walls. Fortunately, these pressure fluctuations have an incoherent stochastic contribution to the microphone signals. This is the case as long as the distance between microphone pairs is large compared to the turbulence length scales in the boundary layer. If self-noise from the boundary layer on the walls is dominant, then the auto-power terms in the diagonal of the CSM will exceed
3.3. Microphone Phased Array Measurements

the cross-powers to a high degree. As a result, the beamformer source maps become contaminated because the auto-power terms still contribute to the source maps of the beamformer. Removal of the diagonal auto-power terms in the CSM mitigates this effect and significantly improves the output maps of the beamformer. With diagonal removal, the weighted steering vector in Eq. 3.17 is redefined to

\[
    w_{j,n}(\omega_k) = \frac{g_{j,n}}{\left( \sum_{n=1}^{N} |g_{j,n}|^2 - \sum_{n+1}^{N} |g_{j,n}|^4 \right)}. \tag{3.22}
\]

Particular attention should be paid when removing the diagonal of the CSM. This is because removing the diagonal makes the CSM negative definite. In other words, the source maps will include negative values which are non-physical and should therefore be removed in an additional step.

Flow Convection Correction

Sound waves propagating through a uniform flow are convected in the direction of the flow. This requires a modification of the steering vector formulation in Eq. 3.18. The flow convection problem is illustrated in Fig. 3.32. With flow convection, the steering vector formulation becomes

\[
    g_{j,n} = \frac{e^{-i2\pi f\tau_{j,n}}}{4\pi \sqrt{\left( \tilde{M} \cdot (\tilde{x}_n - \tilde{\xi}_j) \right)^2 + \beta^2 \|\tilde{x}_n - \tilde{\xi}_j\|^2}}. \tag{3.23}
\]

where \( \beta^2 = 1 - \|\tilde{M}\| \) and the time delay is calculated as

\[
    \tau_{j,n} = \frac{1}{c_\infty \beta^2} \left( -\tilde{M} \cdot (\tilde{x}_n - \tilde{\xi}_j) + \sqrt{\left( \tilde{M} \cdot (\tilde{x}_n - \tilde{\xi}_j) \right)^2 + \beta^2 \|\tilde{x}_n - \tilde{\xi}_j\|^2} \right). \tag{3.24}
\]

Shear Layer Correction

In an open-jet test section, the sound waves convect with the flow and refract at the shear layer. Figure 3.33 shows a schematic illustration of the problem. Amiet’s shear layer correction method [Amiet, 1978] is applied following the procedure described in [Bahr et al., 2011]. Other methods are also available, which are discussed in [Sarradj, 2017].
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Amiet’s shear layer correction is as follows. First, the total arrival time is defined as

$$\tau_a = \frac{r_1}{c_1} + \frac{r_2}{c_0} \quad (3.25)$$

where $r_1 = ||\vec{x}_i - \vec{\xi}_j||$ and $r_2 = ||\vec{x}_n - \vec{x}_i||$ with $\vec{x}_i$ the intersection location of the acoustic path with the shear layer. The intersection location is calculated numerically (i.e. with the Newton-Raphson method) with a set of equations given as

$$\frac{x_i}{\sqrt{x_i^2 + (1 - M_x^2)(y_i^2 + z_i^2)}} - \frac{(1 - M_x^2)}{r_2} \frac{x_n - x_i}{r_2} - M_x = 0 \quad (3.26)$$

$$\frac{y_i}{\sqrt{x_i^2 + (1 - M_x^2)(y_i + z_i^2)}} - \frac{(1 - M_x^2)}{r_2} \frac{y_n - y_i}{r_2} = 0 \quad (3.27)$$

$$z_i = h. \quad (3.28)$$

The speed of sound in the flow is then calculated with

$$c_1 = \frac{x_i}{r_1} M_x c_0 + \sqrt{\left(\frac{x_i}{r_1} M_x c_0\right)^2 + c_0^2 - (M_x c_0)^2}. \quad (3.29)$$

The total arrival time $\tau_a$ and total acoustic path $r_1 + r_2$ can then be calculated.

Alternatively, a simpler method can also be used, described in [Sijtsma, 2010]. This method assumes an average Mach number between the sound source and microphone. Equation 3.23 and 3.23 are then simply evaluated with an average Mach number $M_{avg}$ calculated as

$$M_{avg} = M \frac{\zeta - z_{sl}}{\zeta - z} \quad (3.30)$$

where $\zeta$ is the z-coordinate of the search grid point, $z$ the z-coordinate of the microphone location, and $z_{sl}$ the z-coordinate of the shear layer centerline. This method is occasionally used for closed test section measurements presented in this thesis, where the microphones are slightly recessed in the side walls.
3.3. Microphone Phased Array Measurements

3.3.2 Advanced Beamforming Algorithms

DAMAS

The concept of deconvolution is widely used in signal and image processing. Deconvolution is a principle based on the assumption that a recorded data signal or image \( Y \) is a convolution of the original image \( F \) and a certain transfer function \( H \)

\[
Y = F \ast H. \tag{3.31}
\]

In order to recover the original image, deconvolution of the 'dirty' image \( F \) and transfer function \( H \) can be performed. The principle of deconvolution can also be applied to an acoustic image obtained with CBF. In acoustic beamforming, the transfer function \( H \) is commonly referred to as the Point Spread Function (PSF) of the microphone array. The PSF(\( \xi_j, \xi_s' \)) describes the microphone phased array’s response to a point source in a defined location \( \xi_s' \) and it can easily be calculated.

In 2004, Brooks and Humphrey [Brooks and Humphreys, 2004] implemented the principle of deconvolution to acoustic beamforming and named it the DAMAS method. DAMAS assumes incoherent PSFs with unknown source strength in each search grid.

Figure 3.32: Flow convection effect.  
Figure 3.33: Shear layer effect.
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point. The problem can be mathematically described as a simplified linear system

$$\hat{Y}(f) = \hat{A}(f)\hat{X}(f),$$

(3.32)

where $\hat{Y}$ represents the beamformer source map (from conventional beamforming), $\hat{A}$ the PSF and $\hat{X}$ the unknown source strength at the grid points. Note that the beamformer source maps are a function of frequency. If $A$ is non-singular, then the solution of the system can be easily obtained with the inverse $\hat{X} = \hat{A}^{-1}\hat{Y}$. However, it is found that in acoustic beamforming, $A$ is non-singular with a very low rank because the acoustic problem is ill-posted, and sources are not statistically uncorrelated. However, with a simple iterative method, the solution can be found. To illustrate the iterative process, we write one of the equations of the linear system as

$$A_{j1}X_1 + \cdots + A_{jj'}X_{j'} + \cdots + A_{JJ}X_J = Y_j,$$

(3.33)

where $j$ and $j'$ are grid point index numbers of the search grid. With $A_{jj}=1$, Eq. 3.33 can be written to the form

$$X_j = Y_j - \left[ \sum_{j'=1}^{j-1} A_{jj'}X_{j'} + \sum_{j'=j+1}^{J} A_{jj'}X_{j'} \right].$$

(3.34)

Equation 3.34 is implemented in an iterative algorithm to find $X_j$, the source strength distribution in the search grid using

$$X_1^{(i)} = Y_1 - \left[ 0 + \sum_{n'=1}^{N} A_{1n'}X_{n'}^{(i-1)} \right],$$

$$X_n^{(i)} = Y_n - \left[ \sum_{n'=1}^{n-1} A_{nn'}X_{n'}^{(i)} + \sum_{n'=n+1}^{N} A_{nn'}X_{n'}^{(i-1)} \right],$$

(3.35)

$$X_N^{(i)} = Y_N - \left[ \sum_{n'=1}^{N-1} A_{Nn'}X_{n'}^{(i)} + 0 \right].$$

For the first iteration, the values of $X_j$ are taken as zero. A positive constraint on $X_j$ is enforced to ensure the system’s stability. After all grid points have been calculated in one iteration, the calculation is reversed, moving from $j = J$ back to $j = 1$ to smooth the iteration steps. The DAMAS algorithm requires many calculations, making it a very time-consuming algorithm. In general, the computational time for each iteration depends only on the number of grid points.
CLEAN-SC

The deconvolution method DAMAS has the disadvantage of assuming that the beamformer output map is built up of PSFs, representing uncorrelated monopole point sources. However, these sources rarely exist in practice, where sound sources typically have a certain spatial extent. Therefore, the CLEAN-SC method was proposed in [Sijtsma, 2007] to overcome some of the limitations of DAMAS. The global CLEAN-SC iterative procedure is described as follows [Sijtsma, 2007]:

1. Obtain a beamforming source map with conventional beamforming (i.e. dirty map).
2. Search for the peak location in the dirty map.
3. Subtract the appropriate PSF from the dirty map.
4. Save the 'clean beam' in a separate 'clean map' and degrade the CSM.
5. Start the next iteration.

When sufficient iterations are performed, all 'clean beams' are collected in the 'clean map,' and the remaining dirty map can be added optionally. Mathematically the procedure is as follows. Firstly, the initial iteration, $i = 0$, of the algorithm starts with defining the 'degraded' CSM

$$D^{(i)} = D^{(0)} = \text{CSM}. \quad (3.36)$$

The dirty map is then calculated with conventional beamforming using

$$A_j^{(0)} = w_j^* \tilde{C} w_j, \quad (3.37)$$

where $\tilde{w}_j$ is the weight vector. Secondly, the peak source location $\xi_{max}$ is identified. Thirdly, the contribution of this source is subtracted from the 'dirty map' with

$$A_j^{(i)} = A_j^{(i-1)} - w_j^* G^{(i)} w_j, \quad (3.38)$$

where $G^{(i)}$ is the CSM induced by the source in $\xi_{max}$ and is calculated with

$$G^{(i)} = A_{max}^{(i-1)} h^{(i)} h^* \quad (3.39)$$
Note that the $i$ superscript denotes the iteration step index. If diagonal removal is applied, $G^{(i)}$ is written as

$$\bar{G}^{(i)} = A_{\max}^{(i-1)} (h^{(i)} h^{*(i)} - H^{(i)}),$$

where

$$H_{mn}^{i} = \begin{cases} 0, & \text{for } (m,n) \in S \\ h_m^{(i)} h_n^{*(i)}, & \text{for } (m,n) \notin S \end{cases}$$

with

$$h^{(i)} = \frac{1}{(1 + w_m^{*(i)} H^{(i)} w_n^{(i)})^{1/2}} \left( \frac{\bar{D}^{(i-1)} w_{\max}^{(i)}}{P_{\max}^{(i-1)}} + H^{(i)} w_{\max}^{(i)} \right).$$

The expression for $h^{(i)}$ is implicit and is therefore solved iteratively. Fourth, the clean beam in $\xi^{(i)}_{\max}$ is stored in a separate map

$$Q_j^{(i)} = A_{\max}^{(i-1)} \Phi(\xi_j - \xi_{\max}^{(i)}),$$

where $\Phi$ is a normalized clean beam function of maximum value $\Phi(0) = 1$. For example, an elliptical Gaussian function can be used. Finally, the iteration is completed by degraded the CSM with

$$D^{(i)} = D^{(i-1)} - A_{\max}^{(i-1)} g_{\max}^{(i)} g_{\max}^{(i)}.$$  

The new source power map is subsequently calculated.

$$A_j^{(i)} = w_j^{(i)} \bar{D}^{(i)} w_j.$$  

After all iterations are complete, the final source map is written as

$$A_j = \sum_{i=1}^{I} Q_j^{(i)} + A_j^{(I)}.$$  

A stop criterion can be set when the degraded CSM contains more information than in the previous iteration

$$\| \bar{D}^{(i+1)} \| \leq \| \bar{D}^{(i)} \|. \quad (3.47)$$
3.3. Microphone Phased Array Measurements

3.3.3 The Source Power Integration Technique

Source maps from acoustic beamforming enable the localization and quantification of multiple individual noise sources. A single ideal point source’s total sound pressure level is represented as the maximum peak value in the source maps. However, the total sound pressure level of a collection of distributed sound sources cannot be found in this way. In this case, the source power integration technique [Brooks and Humphreys, 1999] has to be applied. This technique sums the values of a source map within a precisely defined region of integration (ROI) and normalizes this to the array’s response to a point source of unit strength. The normalization is not applied when a source map from a deconvolution algorithm such as CLEAN-SC or DAMAS is integrated. Mathematically, the source power integration is given as

\[ P_{\text{exp}} = P_{\text{sim}} \frac{\sum_{j \in \text{ROI}} A_{j,\text{exp}}}{\sum_{j \in \text{ROI}} A_{j,\text{sim}}} \]  \hspace{1cm} (3.48)

where \( P_{\text{exp}} \) is the total sound pressure level of the noise sources within the ROI, \( P_{\text{sim}} \) the sound power of the simulated source (typically 1), \( A_{j,\text{exp}} \) the source map obtained from the measurement with Eq. 3.21 and \( A_{j,\text{sim}} \) the source map of a simulated point source in the center of the ROI. Particular attention needs to be paid to the definition of the ROI and the use of the source power integration technique. Firstly, the ROI should not contain side-lobes from noise sources outside the ROI. Secondly, noise level values above the noise floor of the measurement should be neglected from the integration. This is especially important when the number of search grid points within the ROI is large compared to the distribution of the noise sources of interest. In this case, the total integration of the search grid points containing noise floor values can add up to a value equal to or larger than the actual sources of interest. The influence of side-lobes and noise floor values can be reduced simply by excluding noise level values in the ROI, which are smaller than a certain threshold, e.g., < 6 dB. The threshold value can be derived from, e.g., the array’s performance or the typical noise floor for an empty wind tunnel measurement. However, often it is chosen empirically. Finally, the ROI should be large enough to account for the broadening of noise sources resulting from coherence loss in open-jet measurements. This broadening is typically small.

The source power integration technique is demonstrated with two simulated cases: a point source and a distribution of point sources, i.e. a line source. No flow is considered in these simulations, and diagonal removal is not applied for simplicity. Figure 3.34 shows an overview of the parameters chosen for the cases. We use a microphone phased
array arrangement containing 64 microphones. This setup is typical for the microphone phased array measurements performed in the aeroacoustic wind tunnel of the University of Twente. The point source is located at 1.5 m from the microphone array in the center of a search grid and emits white noise from 500 Hz to 25 kHz. The frequency resolution is 500 Hz for simplicity, and the search grid resolution is set to 0.01 m. The source strength amplitude $a$ is set to $1\sqrt{2}$ Pa-m, i.e. 1 Pa-m in terms of its root mean square value. The SPL at the center of the array, i.e. $(x,y,z) = (0,0,0)$ is then given as

$$SPL_{array\ center} = 20 \log_{10} \left( \frac{\sqrt{A}}{4\pi R_{s,0} p_{ref}} \right)$$

where $A = \frac{1}{2} |a|^2$ is the source power which in this case is simply one, and $R_{s,0}$ is the distance from the sound source location to the array’s center. This gives that the SPL at the array center is 68.5 dB. Figure 3.35a shows the integrated SPL from the source map compared to the exact solution and the auto-power spectrum of a single microphone located near the center of the array. The source power integration technique is able to recover the sound source strength with high precision.

We also simulate a line source resembling airfoil trailing-edge or leading-edge noise. Figure 3.34b gives an overview of the simulation parameters. The line source extends from $y=\text{-}0.35$ m to $y=0.35$ m and comprises 100 incoherent point sources. Each point source in the line source has a source strength amplitude $a$ of $1\sqrt{2}$ Pa-m. The search grid parameters are the same. The SPL at the source center from the line source is given as

$$SPL_{array\ center} = 10 \log_{10} \left( \frac{A}{32\pi p_{ref} R_{l,0}} \left\{ \arctan \left( \frac{y_{l,\text{min}}}{R_{l,0}} \right) + \arctan \left( \frac{y_{l,\text{max}}}{R_{l,0}} \right) \right\} \right)$$

where $R_{l,0}$ is the distance between the line source and array in the z-direction, $y_{l,\text{min}}$ is the minimum y-coordinate of the line source and $y_{l,\text{max}}$ is the maximum y-coordinate of the line source. This gives a SPL value of 66.8 dB at the center of the array. Figure 3.35b shows the integrated SPL from the source maps from CBF, CLEAN-SC, and DAMAS compared to the exact solution and the auto-power spectrum of a single microphone located near the center of the array. The results show the consistency of the beamforming results, although minor differences of $\pm 1$ dB are still observed. The oscillating behavior of the integrated spectrum from CBF is attributed to the specific performance of the microphone array arrangement.
3.3. Microphone Phased Array Measurements

Figure 3.34: Source power integration parameters. Sound source (●) and ROI (—).

Figure 3.35: Source map of the simulated noise sources at 5 kHz with ROI (—).

Figure 3.36: Array performance parameters.
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3.3.4 Microphone Phased Array Design

An appropriate microphone phased array design is crucial to identify and quantify noise sources accurately. The performance of a microphone phased array is often evaluated using the simulated PSF of the array. Figure 3.37 illustrates a 3-dimensional PSF at 1 and 4 kHz for a circular microphone phased array setup with a point source located in the origin. A main-lobe and various side-lobes characterize the PSF. The level and distribution of these lobes determine the performance of a microphone phased array.

Firstly, the spatial resolution, i.e., the capability of the array to distinguish between sources, is often evaluated with the beamwidth of the PSF. Figure 3.38a shows the definition of the beamwidth which is defined as the width of the main-lobe at 3 dB below its peak value. The beamwidth is based on the so-called Rayleigh resolution limit which for a circular array can be approximated with

$$b_w \approx \frac{R_1 c_0 f D}{1.22}$$

While the spatial resolution of a microphone phased array is often evaluated with the beamwidth, it is not the physical limit below which source can still be separated. The Sparrow’s resolution limit principle is more appropriate, although it is only a fraction smaller than the Rayleigh resolution limit. Moreover, some beamforming algorithms are nowadays able to go below the Rayleigh resolution limit, such as HR-CLEAN-SC [Sijtsma et al., 2017; Luesutthiviboon et al., 2019].

Secondly, a microphone phased array’s ability to distinguish between sources of different amplitude at various locations is often evaluated with the main-lobe to side-lobe ratio. This ratio is defined as the level difference between the main-lobe and the highest side-lobe within a specific region of interest (illustrated in Fig. 3.38b). The largest and most relevant side-lobe is often located close to the main-lobe. The level difference between this side-lobe and the main-lobe can often be used as the threshold value for the source power integration technique. In general, the side-lobe level will increase with distance from the main-lobe, depending on the array’s diameter and the number of microphones. However, these side-lobes are often not relevant since they are outside the typical range of interest in which noise sources are to be localized.

Noise sources are often distributed over a larger area. Therefore, the response and performance of a microphone phased array should be evaluated considering noise sources located in different positions. Fortunately, the PSF can be assumed to be shift-invariant when noise sources are evaluated at locations close to the origin, i.e., for small opening angles. However, if noise sources are located further away, then the PSF will not be
3.3 Microphone Phased Array Measurements

shift-invariant, and the array’s response should be evaluated individually [Suzuki, 2010].

Finding an optimal array design ultimately depends on the specific frequency behavior of noise sources and their spatial distribution. Several studies have therefore focused on optimizing microphone phased array designs [Sarradj, 2015; Malgoezar et al., 2016; Prime and Doolan, 2013; Arcondoulis and Liu, 2019] and artificial weighting functions [Quaegbeur et al., 2016]. Circular spiral designs are most popular because microphone spacings in these designs increase continuously, providing a steady performance over a wide frequency range.
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The measurements shown in this thesis are performed using a circular microphone phased array consisting of 64 GRAS 40PH microphones arranged in a Vogel Spiral pattern defined in spherical coordinates as

\[ r_n = R \sqrt{\frac{n}{N}} \]
\[ \phi = 2\pi n \left(1 + \sqrt{V}\right) \]

where \( R = 0.5 \text{ m} \) is the radius of the array, \( n \) is the microphone index, and \( V \) is a design parameter set to 5. This microphone array arrangement has a good trade-off between the beamwidth and MSR, and each microphone covers approximately the same area. [Sarradj, 2015]. Figure 3.34 shows the Vogel spiral arrangement. The typical performance parameters of this array arrangement in the OTS of the aeroacoustic wind tunnel of the University of Twente are shown in Fig. 3.39. Different modifications of this microphone array setup are used throughout this thesis. This will result in minor changes to the performance evaluation, shown for each setup separately when the results are presented.

3.3.5 Representation of Beamforming Results

It is relevant to accurately define how beamforming results are represented and which steps have been taken to obtain the final results. An overview of the beamforming process discussed throughout this section is shown in Fig. 3.40 for CBF. It is important to observe that the source maps of the beamforming operation (Eq. 3.21) are given
3.3. Microphone Phased Array Measurements

at the source location for the steering vector formulation which is used (Eq. 3.18). Consequently, all relevant sound propagation effects, which occur when the sound waves propagate from the source location to the microphones, need to be considered. The effect that the wind tunnel flow and shear layer have were already discussed in Sec. 3.3.1 and are accounted for in the steering vector computation. We briefly also discuss the relevance of other single microphone and microphone array corrections here.

\[ g = \frac{1}{4\pi} e^{-i2\pi f_x y} \]
\[ CSM = \frac{1}{k} |\beta\|^2 \]

**Figure 3.40**
3.3.5.1 Single Microphone Corrections

Starting from the uncorrected far-field auto-powers of a microphone measurements we express the noise level in terms of the sound pressure level as

\[
\text{SPL}_{\text{uncorrected}} = 10 \log_{10} \left( \frac{p_{\text{rms}}^2}{p_{\text{ref}}^2} \right)
\]  \hspace{1cm} (3.51)

where the \( p_{\text{ref}} \) is the reference pressure in air given as 20\( \mu \)Pa for the human ear. The SPL measured by a single microphone should be corrected, taking into account several corrections. These corrections are divided into three categories: sound propagation, microphone installation, and test-section-dependent corrections. Taking all corrections into account we write corrected SPL of a single microphone as

\[
\text{SPL}_{\text{corrected}} = \text{SPL}_{\text{uncorrected}} + \text{SPL}_{D,s} + \text{SPL}_{R} + \text{SPL}_{SL,r} + \text{SPL}_{A,t} + \text{SPL}_{sb,t} + \text{SPL}_{TS} + (\text{SPL}_{FR} \text{ or } \text{SPL}_{HR}).
\]  \hspace{1cm} (3.52)

where each correction term will be discussed next. General sound wave propagation effects are first discussed.

Sound Propagation Corrections

**Source Directivity**  Firstly, we consider the directivity of the sound source. As an example, airfoil leading-edge and trailing-edge noise typically have a cardioid radiation pattern, i.e., \( \cos^2 \left( \frac{1}{2} \theta \right) \). The source directivity for small emission angles is often assumed to be negligible. For example, the change of SPL from trailing-edge noise directivity for a microphone located at \( \theta = 270^\circ \pm 20^\circ \) is \( \approx 1.5 \text{ dB} \). Sound source directivity \( D_s \) can be corrected using

\[
\text{SPL}_{D,s} = D_s(f, \Theta, \Phi)
\]  \hspace{1cm} (3.54)

where \( \Theta \) and \( \Phi \) are the polar and azimuthal angle respectively. For microphone array measurements, directivity corrections are often made relative to the center location of the array where the integrated SPL levels are expressed.
3.3. Microphone Phased Array Measurements

**Spherical Spreading** Secondly, we consider spherical spreading from a point source. This can be accounted for by correcting to a reference distance $R_{ref}$ using

$$\text{SPL}_R = 20 \log_{10} \left( \frac{R}{R_{ref}} \right)$$

(3.55)

where $R_{ref}$ is a reference distance. Typical values for the reference distance are 1 m or 0.282 m. The latter distance corresponds to the radius of a sphere with a surface area of 1 m$^2$. The sound pressure level equals the sound power level in a free-field condition at this distance.

**Shear Layer Refraction** Third, the change of SPL due to shear layer refraction is considered. When sound waves propagate through a shear layer, the sound waves are refracted and spread. This leads to an attenuation of the sound pressure level measured by a microphone out of the flow [Ingard, 1959]. The change in SPL due to shear layer refraction can be corrected using

$$\text{SPL}_{SL,r} = A_{SL,r}(\theta, U_0).$$

(3.56)

For small emission angles $\theta$ and low Mach number flow, the attenuation can be assumed to be negligible.

**Atmospheric Absorption** Fourth, we consider the atmospheric absorption of sound waves due to viscous losses. This correction is mainly relevant for large sound propagation distances or high frequencies of sound waves. The method to calculate atmospheric absorption is extensively described in ISO 9613. Atmospheric absorption can be corrected using

$$\text{SPL}_{At} = A_t(f, h, T, p, R)$$

(3.57)

where $f$ is frequency, $h$ is humidity $T$ is temperature and $p$ is pressure. For small-scale wind tunnel testing, atmospheric absorption can be assumed to be negligible.

**Boundary Layer Absorption** Finally, we consider boundary layer absorption, which occurs due to viscous dissipation from the interaction with turbulence in a shear layer or boundary layer, similarly to atmospheric absorption. Only limited research is available on this topic. [Ahuja et al., 1978] performed measurements to determine the attenuation
of sound waves from shear layer turbulence in an open-jet. This study shows that the absorption of sound waves by shear layer or boundary layer turbulence is typically negligible. Tonal noise, however, suffers from spectral broadening as a consequence of the interaction with turbulence [Sijtsma et al., 2014]. The loss of acoustic energy of a tone can be corrected using

\[
\text{SPL}_{sb,t} = A_{sb,t}(f, U_0, \delta_{SL}) \tag{3.58}
\]

where \(\delta_{SL}\) is the shear layer thickness.

Microphone Installation Corrections

Besides sound propagation effect, it is also important to consider several microphone installation effects.

**Free-Field and Pressure-Field Response**  
Firstly, microphones are often optimized for either a pressure-field or a free-field utilization. If a free-field microphone is flush-mounted in a wind tunnel wall, i.e., in a pressure-field condition, then pressure doubling at the wall will occur, which needs to be considered. In addition, microphone membranes have a finite size and will respond differently to sound waves with a small and large wavelength compared to the membrane. These effects are considered in the free-field and pressure-field calibration of a microphone. Manufacturers generally provide this information for standard conditions. However, wall-mounted microphones are often placed in optimized mountings for wind tunnel use [Fleury et al., 2012; C. VanDercreek and Snellen, 2022]. Consequently, the pressure field response will differ, requiring additional in-situ calibration. Free-field and pressure-field transfer functions can then be accounted for using

\[
\text{SPL}_{FR} = H_{FR}(f, \Theta_{FR}, \Phi_{FR}) \tag{3.59}
\]

\[
\text{SPL}_{PR} = H_{PR}(f, \Theta_{PR}, \Phi_{FR}) \tag{3.60}
\]

where \(H_{FR}\) and \(H_{PR}\) are the free-field and pressure-field response functions. Note that this installation correction also includes the directivity of the microphone itself.
3.3. Microphone Phased Array Measurements

**Acoustic Transparent Materials** Secondly, we consider the use of acoustic transparent materials in combination with the microphone installation. These acoustic transparent materials cause a transmission loss of the sound waves propagating through the material. In a CTS, microphones are often mounted behind acoustic transparent materials, which reduce the self-noise from the boundary layer on the walls [Jaeger et al., 2000; Sijtsma and Holthusen, 1999]. Commonly used transparent materials are perforated plates, Kevlar cloth, or glass fiber cloth. In an HTS, stretched Kevlar cloth is widely used. Measurement to determine the sound transmission loss can be performed using either impedance tubes or specialized setups in anechoic chambers shown, e.g., in [Phong and Papamoschou, 2013; Devenport et al., 2013]. Transmission loss can be corrected using

\[
\text{SPL}_{TS} = A_{TS}(f, \Theta, \Phi)
\]  

and is typically dependent on the frequency of the sound source and the emission angles.

3.3.5.2 Microphone Array Corrections

Acoustic corrections for a microphone phased array system are also considered. These corrections typically involve a modification of the steering vector formulation or CSM.

**Shear-Layer Induced Coherence Loss** In the OTS, sound waves propagating through the shear layer are affected by the turbulent fluctuations, which not only leads to refraction or absorption but also to the random disturbance of wave fronts. As a result, coherence loss occurs between microphone pairs in the phased array [Wilson, 1998; Koop et al., 2005]. The coherence loss effect on phased array measurements can be implemented in several ways [Biesheuvel et al., 2019; Ernst et al., 2015; Pires et al., 2012; Bahr, 2021]. The validation and implementation of a coherence loss model used in this thesis is discussed in Chapter 6. The coherence loss requires a correction of the integrated SPL level of

\[
\text{SPL}_{CL} = A_{CL}(f, U_0, \delta_{SL}, \Delta r)
\]  

and depends on the frequency of the sound source, the free-stream velocity, the shear layer thickness and the distance between microphone pairs \(\Delta r\).
Reverberation  Sound wave reflections from solid surfaces affect microphone measurements in the CTS and can also affect measurements in the OTS and HTS, where side-plates are used to mount airfoils. These reflections create mirror sources that contaminate beamforming source maps, mainly at low frequencies. Several techniques have been proposed to reduce the influence of mirror sources. [Guidati et al., 2002] propose a ‘reflection-canceler’ by adding the steering vectors from mirror sources to the original steering vectors. [Sijtsma and Holthusen, 2003] propose a minimization procedure that minimizes reflections. [Fenech and Takeda, 2007] propose an image source model to reconstruct the source field in the CTS wind tunnel. A reverberation technique is proposed in [Fischer and Doolan, December 2017] which proposes a modification of the CSM without the need for information on the test section geometry. While available, de-reverberation techniques still have practical limitations and are not widely used. Therefore, no de-reverberation techniques have been implemented or used for the measurements presented in this thesis.

Background Noise  Background noise is routinely present in wind tunnel testing. General sources of background noise are, e.g., the wind tunnel drive fan or turning vanes. In the OTS, background noise is also generated from boundary layer scattering at the endplates and the interaction of the unsteady jet with the collector. In the CTS, self-noise from the boundary layer on the walls contaminates microphone signals. While these background sources can be reduced, they cannot be completely mitigated. As discussed before, diagonal removal is commonly used to remove uncorrelated background noise such as electronic noise. A more advanced background removal technique to reduce statistically stationary noise is presented in [Bahr and Horne, 2015]. However, this technique has several disadvantages, which complicate its practical use. A method to improve acoustic beamforming results in the presence of arbitrary correlated noise is given in [Raumer et al., 2021]. This method improved the signal-to-noise ratio of beamforming and improved the resolution of beamforming maps, especially at low-frequencies. A method to remove transient background noise contamination is also presented in [Bahr and Schultz, 2019] showing promising results. Although several methods exist for background noise reduction, they are not yet widely used and are also not used in this thesis.

Source Power Integration  The SPI technique requires the definition of the ROI along the chordwise and spanwise direction of an airfoil. Typically, the spanwise extent
of the ROI is smaller than the wetted span of the airfoil. This is done to avoid the inclusion of spurious sources at the side-wall junction. When comparing the integrated SPL levels from airfoil measurements, it is convenient to normalize to a reference spanwise extend of the ROI. This is done using

$$\text{SPL}_{\text{ROI}} = 10 \log_{10} \left( \frac{l_{\text{ref}}}{l_{\text{ROI}}} \right)$$

(3.63)

where \(l_{\text{ROI}}\) is the spanwise extend of the ROI along the airfoil span and \(l_{\text{ref}}\) is a reference airfoil span, typically 1 m.

### 3.3.6 A Benchmark Validation

The acoustic beamforming problem is mathematically ill-posed. As a result, the final output can be subjected to significant variability. It can depend on, e.g., the distribution of the sound sources, the implementation of corrections, or the criteria set in the source power integration. The scientific community set up a benchmark database to evaluate the variability of various beamforming techniques [Bahr et al., 2017; Sarradj et al., 2017]. This database contains both analytical and experimental cases representing different aeroacoustic problems. The in-house beamforming codes of the University of Twente have been benchmarked with a subset of cases from this benchmark database. These are the simulated data set cases ”b1” and ”b7”, representing a line source in a closed wind tunnel test section and a case containing multiple incoherent monopole sources of various strengths. The ”NASA2” experimental data set case was also used, which contains leading-edge and trailing-edge noise measurements from a 2D airfoil in NASA’s QFF open test section wind tunnel. The benchmark results from these test cases can be found in Appendix 3.4.
3.4 Flow Field Measurements with Particle Image Velociometry

Time-averaged PIV measurements of the slat cove region were performed in a 2D plane, capturing two velocity components. Figure 3.41 shows the pictures of the PIV setup in the OTS, CTS, and HTS. The HTS was slightly modified to allow the laser sheet to enter the test section. Three smaller panels are mounted to the test section walls on the lower side of the 30P30N model. The center panel comprises an acrylic panel, while the other two panels are stretched Kevlar cloth.

A laser sheet was generated with a double pulsed ND:YAG laser (Litron Nano L 145-15 PIV) with maximum output energy of 135 mJ at the wavelength of 532 nm and a maximum repetition rate of 15 Hz. A laser guide arm was used to pass the laser beam to the test section and sheet optics. Images are acquired using a LaVision Imager SX 9M camera with a 105 mm lens (SIGMA 105 mm F/2.8 DG Macro) and a green light filter (shown in Fig. 3.42). The cameras are calibrated with a LaVision 106-10 calibration plate measuring 106 mm x 106 mm (shown in Fig. 3.42). Orange fluorescent tape with 0.05 mm thickness was applied to minimize laser reflections on the illuminated surfaces of the 30P30N model. The tracer particles were generated using a PivPart45-M yielding micro-droplets with a mean diameter of 1.0 μm.
3.4. Flow Field Measurements with Particle Image Velocimetry

Figure 3.41: Pictures of the PIV setups.

(a) OTS.

(b) CTS.

(c) HTS.

Figure 3.42: Pictures of the PIV setups.

(A) Camera installation.

(B) Calibration plate.
Appendix

3.A The WOSA Method for Spectral Estimation

A popular method for the spectral estimation of a data signal is the Welch method, also known as the WOSA (windowed overlapped segmented average) method [Trobs and Heinzel, 2006]. This method considers a constant frequency resolution of the spectral estimation.

Let us consider a discrete time signal of $M$ samples $x(m), m = 0, ..., M - 1$ that has been sampled with frequency $f_s$. The signal will be evaluated at a set $L$ of Fourier frequencies $f(l), l = 0, ..., L - 1$. The data signal is divided into blocks of size $L(l)$ with a distance $D(l)$ apart. Note that the size of $L(l)$ depends on the sampling frequency and resolution bandwidth

$$L(l) = f_s / r(l). \quad (3.64)$$

The value for $r(l)$ should be chosen such that $L(l)$ is an integer and preferably a power of 2. This would increase the efficiency of the FFT algorithm (although not necessarily needed with modern-day fast computers and algorithms). The overlap $\psi$ between blocks may be defined by

$$D(l) = (1 - \psi)L(l). \quad (3.65)$$

A windowing function is applied to each segment of data $L(l)$ after which a Fourier transform is performed on the whole data window (including overlap). This results in a scalar product $B(l, q)$ where $q$ is the data segment index. Subsequently, the squared
3.A. The WOSA Method for Spectral Estimation

magnitude is obtained with $|B(l, q)|^2 = B \ast \text{conj}(B)$. The spectral estimate of the complete data signal is then the average of all the squared magnitudes of all blocks segments

$$\hat{p}(f(l)) = \frac{C}{Q} \sum_{q=1}^{Q} |B(l, q)|^2.$$ (3.66)

where $C$ is a normalization factor and $Q$ the total number of segments. For the normalization $C$, let us consider two sums that are needed for calibrating the spectral estimates

$$S_1(l) = \sum_{l_2} L(l) w(l, l_2),$$ (3.67)

$$S_2(l) = \sum_{l_2=1} L(l) w^2(l, l_2).$$ (3.68)

When the data signal is divided into segments without overlap, windowing is applied, as illustrated in Fig. 3.44. It is evident that a part of the data signal is completely discarded due to the windowing function being at its boundaries. By introducing an overlap, as illustrated in Fig. 3.45, less data is discarded.

A spectral estimate of a time signal can be represented in different forms. Commonly used spectral representations are shown in Tab. 3.2. According to [Trobs and Heinzel,
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[2006], spectral densities and spectra with constant frequency resolution can be converted to each other when the equivalent noise bandwidth ENBW is known. It is computed by

\[
ENBW = r \cdot NENBW = NENBW \cdot \frac{f_s}{L} = f_s \frac{S_2}{(S_1)^2},
\]

(3.69)

where \( NENBW = \frac{L S_2}{(S_1)^2} \) is the normalized equivalent noise bandwidth. [Trobs and Heinzel, 2006] gives the calibration coefficient for the power spectrum estimation and power spectral density respectively as

\[
C_{PS}(l) = \frac{2}{(S_1(l))^2},
\]

(3.70)

\[
C_{PSD}(l) = \frac{2}{f_s S_2(l)}.
\]

(3.71)

The WOSA method was programmed in MATLAB. Figure 3.46 illustrates the different spectral estimations and their associated units computed from a measurement of the wall pressure spectrum near the trailing edge of an airfoil. The frequency resolution is 6.1 Hz, which illustrates the apparent difference between the spectrum and the spectral density. If the frequency resolution is chosen to be 1 Hz, then the PS and PSD will evidently overlap. This is due to the fact that a PSD normalizes the spectrum. This normalization is especially useful when comparing spectral results without being concerned about the frequency resolution that has been used. However, care must be taken when comparing peak values of tonal noise (i.e. pure tones) using a spectral density. The chosen frequency resolution can still have a significant impact on the peak value that is calculated while the amplitude is normalized. This means that a coarser frequency resolution will result in a lower peak value of the tonal noise in a spectral density plot.

<table>
<thead>
<tr>
<th>Abbreviation</th>
<th>Name</th>
<th>Relation for constant ENBW</th>
<th>Unit</th>
</tr>
</thead>
<tbody>
<tr>
<td>PSD</td>
<td>Power spectral density</td>
<td>PS = PSD· ENBW</td>
<td>(V^2/\text{Hz})</td>
</tr>
<tr>
<td>PS</td>
<td>Power spectrum</td>
<td>PS = PSD· ENBW</td>
<td>(V^2)</td>
</tr>
<tr>
<td>LSD</td>
<td>Linear spectral density</td>
<td>LSD = (\sqrt{(PSD)})</td>
<td>(V\sqrt{\text{Hz}})</td>
</tr>
<tr>
<td>LS</td>
<td>Linear spectrum</td>
<td>LS = (\sqrt{\text{PSD}} = \text{LSD} \cdot \sqrt{\text{ENBW}})</td>
<td>(V)</td>
</tr>
</tbody>
</table>

Table 3.2: Naming convention for spectra [Trobs and Heinzel, 2006]
3.B Application of Beamforming Algorithms to Benchmark Cases

This chapter describes the benchmark cases that are hosted by the BTU Cottbus. These cases consist of analytical and experimental data that have been used to compare beamforming results of different users. A comparison of the results obtained by these users are presented in [Sarradj et al., 2017] for case 1 and 7 and in [Bahr et al., 2017] for the NASA and DLR case. Other cases are also available within this benchmark dataset. These specific cases were selected since they present the type of noise sources expected in wind tunnel measurements of airfoils in closed and open test section configurations.

3.B.1 BTU Benchmark case 7 - Four Monopole Sources (Simulated)

Case Description

This case presents four monopole sources at the corners of a 0.2 m by 0.2 m square centered in the origin of the x,y plane at a distance of 0.75 m from the array center (Fig. 3.47). The microphone array contains 64 microphones with an aperture of approximately 1.5 m. The case is divided into two subcases:
• **subcase a** all monopole sources have identical power.

• **subcase b**: the monopole sources have different power with maximum difference of 18 dB.

<table>
<thead>
<tr>
<th>Source</th>
<th>x [m]</th>
<th>y [m]</th>
<th>z [m]</th>
<th>Case a power [dB]</th>
<th>Case b power [dB]</th>
</tr>
</thead>
<tbody>
<tr>
<td>Source 0</td>
<td>0.1</td>
<td>-0.1</td>
<td>0.75</td>
<td>69</td>
<td>69</td>
</tr>
<tr>
<td>Source 1</td>
<td>-0.1</td>
<td>-0.1</td>
<td>0.75</td>
<td>69</td>
<td>63</td>
</tr>
<tr>
<td>Source 2</td>
<td>-0.1</td>
<td>0.1</td>
<td>0.75</td>
<td>69</td>
<td>57</td>
</tr>
<tr>
<td>Source 3</td>
<td>0.1</td>
<td>0.1</td>
<td>0.75</td>
<td>69</td>
<td>51</td>
</tr>
</tbody>
</table>

**Table 3.3:** Sound source location and strength for both subcases of case 7.

![Microphone array configuration and search grid of case 7 - Four Monopole Sources.](image)

**Figure 3.47:** Microphone array configuration and search grid of case 7 - Four Monopole Sources. Microphone array distribution (black), search grid (blue) and sound sources (red).

Results are presented as sound pressure at the array center. The BeamUT beamforming methods that were used and their parameters are given in Tab. 3.4. Similar grid resolution and ROI areas are chosen compared to the other contributors.

**Results**

The results for this benchmark case are shown in terms of the $\Delta L$ sound pressure level difference

$$\Delta L = L_{\text{estimated}} - L_{\text{true}},$$  \hspace{1cm} (3.72)
3.B. Application of Beamforming Algorithms to Benchmark Cases

which computes the difference between the true and estimated source strength.

<table>
<thead>
<tr>
<th>Algorithm</th>
<th>Parameters</th>
<th>ROI (x,y) [m]</th>
</tr>
</thead>
</table>
| DAMAS     | CSM Diagonal Removal  
  $x_{\text{min}} = -0.5 \text{ m}, x_{\text{max}} = 0.5 \text{ m}, y_{\text{min}} = -0.5 \text{ m}, y_{\text{max}} = 0.5 \text{ m}$  
  Grid resolution: 0.025 m  
  100 iterations | 0.1 $\times$ 0.1 square |
| CLEAN-SC | CSM Diagonal Removal  
  $x_{\text{min}} = -0.5 \text{ m}, x_{\text{max}} = 0.5 \text{ m}, y_{\text{min}} = -0.5 \text{ m}, y_{\text{max}} = 0.5 \text{ m}$  
  Grid resolution: 0.025 m  
  100 iterations, safety factor = 0.9, no stop criterion | 0.1 $\times$ 0.1 square |

Table 3.4: Overview of the BEAMUT algorithms and their parameters used for case 7.

Subcase a

The BeamUT results of subcase-a for each source are shown in Fig. 3.48. Results from other contributors to the benchmark case are additionally shown in Fig. 3.50. Figure 3.48 clearly shows that both DAMAS and CLEAN-SC estimate the source power level with a $\pm 1$ dB accuracy. Even though both algorithms are principally different, their results are remarkably close. The BeamUT algorithms show similar performance to the results of other contributors.

Subcase b

The BeamUT results of subcase-b are shown in Fig. 3.49 with results from other contributors shown in Fig. 3.51. This subcase presents sources of different amplitude. DAMAS and CLEAN-SC again show an accuracy within $\pm 1$ dB for source 0 and 1. However, the results diminish to an accuracy of 2 dB for source 2. The integrated levels of DAMAS for source 3 fluctuate even more significantly, whereas the CLEAN-SC results fall outside the $\pm 4$ dB margin of the figure. Results from the other contributors show similar behavior for source 3. However, to a smaller extent. It should be further investigated why the UTwente beamforming algorithms perform less for this given source.
Chapter 3. Aeroacoustic Facility and Methods

Figure 3.48: Results for subcase a of the case 7 - four monopole sources of same power.

Figure 3.49: Results for subcase b of the case 7 - four monopole sources of different power.
3.B. Application of Beamforming Algorithms to Benchmark Cases

**Figure 3.50:** Results of other contributors for subcase a of the case 7 [Sarradj et al., 2017].
Figure 3.51: Results of other contributors for subcase b of the case 7 [Sarradj et al., 2017].
3.B. Application of Beamforming Algorithms to Benchmark Cases

3.B.2 BTU Benchmark case 1 - Line Source in Closed Test Section Wind Tunnel (Simulated)

![Microphone array configuration and search grid of case 1 - Line Source in CTS Environment. Microphone array distribution (black), search grid (blue) and sound sources (red).](image)

**Figure 3.52:** Microphone array configuration and search grid of case 1 - Line Source in CTS Environment. Microphone array distribution (black), search grid (blue) and sound sources (red).

**Case Description**

The BTU benchmark case 1 provides a simulated line source in a closed wind tunnel test section environment. This type of source is exemplary for an airfoil’s trailing edge noise source. The line source has a length of 2 m and stretches from $z = -1$ m to $z = 1$ m. A uniform flow of $M_x = 0.22$ is present.

The microphone coordinates of the phased array and search grid are shown in Figure 3.52. The microphone array consists of 93 microphones and has an aperture of 1.8 m with its center at $y = -1$ m. An analytical definition of the source can be found in [Sarradj et al., 2017]. The line source strength was chosen such that

$$10 \log(C_{array\ center}) = 71.16 - (10 - 0.34127(f/f_0) - 0.87242(f/f_0)^2 + 0.16300(f/f_0)^3$$

$$- 0.0082341(f/f_0)^4),$$

where $f_0 = 1000$ Hz.
Gaussian white noise is added to the sound source signal. At a sampling frequency of 51.2 kHz, 60 seconds of white noise is added incoherently from microphone to microphone. The sound pressure level value of the noise is 86.98 dB. This results in a SNR ratio that varies between -25.7 dB to -15.7 dB from low to high frequency, respectively. The case still presents a severely contaminated signal even after the diagonal removal of the CSM.

The BeamUT algorithm parameters are shown in Tab. 3.5. The grid resolution and ROI areas are chosen to match the ROIs used by other contributors in [Sarradj et al., 2017].

<table>
<thead>
<tr>
<th>Algorithm</th>
<th>Parameters</th>
<th>ROI (x, z) [m]</th>
</tr>
</thead>
<tbody>
<tr>
<td>CBF SPI</td>
<td>CSM Diagonal Removal</td>
<td>0.08 × 2</td>
</tr>
<tr>
<td></td>
<td>$x_{\text{min}} = -1.0 \text{ m}$, $x_{\text{max}} = 1.0 \text{ m}$, $y_{\text{min}} = -1.0 \text{ m}$, $y_{\text{max}} = 1.0 \text{ m}$ Grid resolution: 0.02 m</td>
<td></td>
</tr>
<tr>
<td>DAMAS</td>
<td>CSM Diagonal Removal</td>
<td>0.12 × 2</td>
</tr>
<tr>
<td></td>
<td>$x_{\text{min}} = -1.0 \text{ m}$, $x_{\text{max}} = 1.0 \text{ m}$, $y_{\text{min}} = -1.0 \text{ m}$, $y_{\text{max}} = 1.0 \text{ m}$ Grid resolution: 0.02 m 100 iterations</td>
<td></td>
</tr>
<tr>
<td>CLEAN-SC</td>
<td>CSM Diagonal Removal</td>
<td>0.08 × 2</td>
</tr>
<tr>
<td></td>
<td>$x_{\text{min}} = -1.0 \text{ m}$, $x_{\text{max}} = 1.0 \text{ m}$, $y_{\text{min}} = -1.0 \text{ m}$, $y_{\text{max}} = 1.0 \text{ m}$ Grid resolution: 0.02 m 100 iterations, safety factor = 0.9, no stop criterion</td>
<td></td>
</tr>
</tbody>
</table>

Table 3.5: Overview of the BEAMUT algorithms and their parameters used for case 1.

Results

The BeamUT results are shown in Fig. 3.53. The CBF SPI results look close to the exact solution at high frequencies. However, the accuracy is less significant at the low frequencies, up to 2 dB. The DAMAS result shows good agreement with the exact solution and has similar performance when compared to the DAMAS algorithm of NASA and BTU in Fig. 3.54. The CLEAN-SC algorithm shows poor results, similar to other contributors’ CLEAN-SC algorithms. Sarradj explains that this is due to the high level of noise contamination in the CSM, which harms identifying coherent sound sources.
3.B. Application of Beamforming Algorithms to Benchmark Cases

Figure 3.53: Line source case results.

Figure 3.54: Results from other contributors of case 1 [Sarradj et al., 2017].
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3.B.3 BTU Benchmark NASA2 case - NACA 63-215 airfoil in NASA QFF

This case provides experimental data of acoustic measurements of a NACA 63-215 airfoil in the NASA QFF wind tunnel. The airfoil was placed at an angle of attack $\alpha = 1.2^\circ$, at which it generates zero lift. The wing’s chord length is 0.406 m, and its span 0.914 m. The mean flow Mach number is $M_x = 0.17$. The leading and trailing edge noise sources are estimated from the beamforming results. This experimental dataset was also used in the original DAMAS publication [Brooks and Humphreys, 2004].
3.B. Application of Beamforming Algorithms to Benchmark Cases

<table>
<thead>
<tr>
<th>Algorithm</th>
<th>Parameters</th>
<th>ROI (y,x) [m]</th>
</tr>
</thead>
</table>
| CBF SPI   | CSM Diagonal Removal & No Diagonal Removal  
  $x_{\text{min}} = 0.3684$ m, $x_{\text{max}} = 1.6382$ m, $y_{\text{min}} = -0.635$ m, $y_{\text{max}} = 0.635$ m  
  Grid resolution: 0.025 m | $0.8 \times 0.3$ |
| DAMAS     | CSM Diagonal Removal  
  $x_{\text{min}} = 0.3684$ m, $x_{\text{max}} = 1.6382$ m, $y_{\text{min}} = -0.635$ m, $y_{\text{max}} = 0.635$ m  
  Grid resolution: 0.025 m  
  100 iterations | $0.8 \times 0.3$ |
| CLEAN-SC  | CSM Diagonal Removal  
  $x_{\text{min}} = 0.3684$ m, $x_{\text{max}} = 1.6382$ m, $y_{\text{min}} = -0.635$ m, $y_{\text{max}} = 0.635$ m  
  Grid resolution: 0.025 m  
  100 iterations, safety factor = 0.9, no stop criterion | $0.8 \times 0.3$ |

Table 3.6: Overview of the BEAMUT algorithms and their parameters used for the NASA case.

Case Description

The microphone array consists of 33 microphones at a distance of of $z = 1.524$ m from the airfoil’s trailing edge. The Small Aperture Directional Array (SADA) has an aperture of 0.2 m. Microphone signals were acquired with a sampling frequency of 143 kHz with a total of 2,048,00 samples within 14.3 seconds time. The search grid parameters were chosen to match the grid provided by NASA. The ROI for both the leading and trailing edge spectra were arbitrarily chosen, but their respective sizes are identical. The BeamUT algorithm parameters that were used for this case are given in Tab. 3.6.

Results

A comparison of one-third octave band maps is shown in Fig. 3.57 to illustrate the effect of diagonal removal on the CBF results. The horizontal black lines indicate the airfoil geometry, whereas the vertical lines indicate the wind tunnel test section walls. The peak levels are reduced, and some of the background noise is suppressed. As we will see, diagonal removal will, therefore, also affect the spectral estimations of the leading and trailing edge noise. Another comparison is shown in Fig. 3.58 of the 12.5 kHz one-third octave bin output maps from BeamUT with results presented in [Bahr et al., 2017]. The conventional beamforming maps of BeamUT and UNSW appear similar within approximately a 1 dB difference of the peak value. The CLEAN-SC results of BeamUT show a lower peak value compared to the UNSW result ($\pm$ 4 dB). The BeamUT DAMAS
shows a $\pm 3$ dB higher peak value than NASA’s DAMAS algorithm. Although these comparisons show different peak value levels, the BeamUT DAMAS and CLEAN-SC display similar peak value levels.

The comparison of the integrated leading and trailing edge spectra is shown in Fig. 3.59 and 3.60 respectively. The integrated spectra are shown in terms of the one-third octave SPL per foot. A good agreement of the leading edge spectra is observed between 10 and 40 kHz. The spectral shapes of individual algorithms are similar. The conventional beamforming algorithms deviate from all other results below 10 kHz. Nevertheless, the UNSW and UTwente conventional beamforming spectra show similar behavior. It is also interesting to note the difference between the conventional beamforming with and without diagonal removal of the CSM. Diagonal removal effectively removes noise that is uncorrelated to the airfoil noise. Therefore, the integrated noise spectra with diagonal removal are lower than when no diagonal removal is applied. The trailing edge spectra show good agreement at low frequencies (2 - 10 kHz) in contrast with the leading edge spectra. Beyond 10 kHz, the spread between results becomes larger. This type of behavior seems to depend on whether or not the diagonal of the CSM is removed. Nonetheless, the BeamUT algorithms show mutually similar spectra.

![Figure 3.57: Comparison of one-third octave band maps from conventional beamforming of the 12.5 kHz center band frequency. The dynamic range is set to 6 dB in both figures.](image)

(A) UTwente CBF - No DR.

(B) UNSW CBF - DR.
3.B. Application of Beamforming Algorithms to Benchmark Cases

Figure 3.58: Comparison of one-third octave band maps of the 12.5 kHz center band frequency. The left column shows the contribution from UTwente whereas the right columns shows other contributions in [Bahr et al., 2017]. All output maps are scaled with 20 dB except the CBF results.
Figure 3.59: NASA case results for leading edge spectra.

Figure 3.60: NASA case results for trailing edge spectra.
References


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REFERENCES


Y. Mayer, B. Zang, and M. Azarpeyvand. Design of a kevlar-walled test section with dynamic turntable and aeroacoustic investigation of an oscillating airfoil. In *25th
REFERENCES


REFERENCES


REFERENCES


Chapter 4

Trailing-Edge Noise Comparability in Open, Closed, and Hybrid Wind Tunnel Test Sections.


Abstract

Open-jet, hard-wall and hybrid test section configurations are typically used in wind tunnel tests, aiming to represent the ideal free-flight condition. Each test section type requires special adaptations of the experimental hardware and specific correction methodologies which account for systematic measurement errors. Differently from previous research, this study investigates the comparability of measurements performed in three test section configurations in the same wind tunnel. With this approach, systematic errors related to the facility or boundary-layer tripping methodology are minimized. Basic 2D aerodynamic boundary corrections are evaluated using a DU97W300 airfoil. The corrected lift curve collapsed well while differences in stall behavior and $C_p$ distribution at larger angles of attack are non-negligible. The trailing-edge noise produced by a NACA-0012, NACA-0018 and NACA-63018 served as reference aeroacoustic sources.
Chapter 4. Trailing-Edge Noise Comparability in Open, Closed, and Hybrid Wind Tunnel Test Sections.

in this comparability analysis. Moreover, the noise reduction from trailing-edge serrations served as an additional reference for the comparability of relative noise levels. The chord-based Reynolds number of the experiments ranged from 260,000 to 660,000. Unsteady wall pressure measurements performed near the trailing-edge provided a reference for the aeroacoustic noise source terms, demonstrating a negligible change between the different test section types. The applied experimental hardware and acoustic corrections yielded aeroacoustic sources of comparable absolute far-field noise level in the three test section types within $\pm$ 1-3 dB. The relative noise levels are comparable within $\pm$ 1-3 dB. These results show that aeroacoustic measurements of trailing-edge noise performed in different test section configurations are equivalent, provided that adequate experimental and post-processing methodologies are systematically implemented.

4.1 Introduction

The present demand for energy resources and for the transportation of people and goods requires the urgent need for drastic changes in our society’s way of living. Achieving the ambitious targets for reduced green-house gas emission requires changes to, e.g., our consumption habits, waste output and use of resources which are facilitated by breakthrough scientific and technological advances. The energy industry is presently transitioning towards renewable energy sources, with wind turbines designated as part of the way to move forward. The aviation industry is required to undergo significant changes to reduce its large contribution to the global green-house gas emission. Changes in the propulsion system and aircraft design will be necessary to maximize the power conversion of renewable energy sources. The expansion of on-shore wind farms and the development of new aircraft concepts are hampered by concerns related to increased noise levels affecting the health and well-being of people [Michaud et al., 2016; Franssen et al., 2004] and animals [Kikuchi, 2008] in the surrounding environment. Governments have therefore applied strict noise regulations for the certification of wind farms and for air traffic around airports.

One of the most important noise sources of wind turbines and aircraft is from the trailing-edge noise mechanism, which is therefore widely studied in the literature [Amiet, 1976; Brooks et al., 1989; Chase, 1975; Howe, 1978; Williams and Hall, 1970; Stalnov et al., 2016] as well as its reduction [M.S., 1991; Lyu et al., 2016; Gruber et al., 2010]. Experimental studies of this trailing-edge noise mechanism are often performed in small-scale aeroacoustic wind tunnels providing valuable data for empirical modeling of the noise,
validation of numerical simulations and for evaluation of noise reducing concepts. The wind tunnel test section type may vary depending on the test requirements and facility restrictions. The main configuration types are a hard-walled, open-jet or a hybrid test section. Aeroacoustic tests performed in a hard-walled test section (CTS) have the microphone phased arrays installed in the test section walls. The boundary condition imposed by the hard-wall creates a flow that is the best approximation of the free-flight aerodynamic condition compared to the other test section types. However, flush-mounting the microphones in the walls of the test section gives unfavorable acoustic conditions for microphone phased array measurements, namely: 1) the microphone signals are contaminated with self-noise from the boundary layer on the test section walls [Jaeger et al., 2000]; 2) the solid walls reflect acoustic waves significantly affecting the beamforming process of microphone phased array measurements [Fischer and Doolan, December 2017]; and 3) microphones typically cannot be placed in arbitrary positions, making it difficult to perform directivity measurements of the noise. For these reasons, the hard-wall test section is unfavorable for aeroacoustic measurements. Nonetheless, this test section type is often used because of facility limitations, for example, in cryogenic wind tunnels where high Reynolds number measurements can be achieved [Ahlefeldt, 2017]. The acoustic issues of a CTS are eliminated by removing the hard-walls of the test section, creating an open-jet configuration type (OTS). Microphones are placed in the quiescent air, away from the open-jet flow. An anechoic chamber encloses the test section of the wind tunnel so that an acoustic free-field is created. The open-jet test section type improves the signal-to-noise ratio, reducing acoustic reflections from wind tunnel walls, and allowing for directivity measurements of airfoil models. However, the open-jet test section type also has specific limitations. Firstly, aerodynamic corrections to the free-flight condition can be significant, especially for high-lift airfoils [Brooks and Marcolini, 1984; Moreau et al., 2003] with a large chord length compared to the wind tunnel width. Secondly, the shear-layer, separating the jet flow and surrounding quiescent air, induces coherence loss between microphone pairs, which increases with flow velocity, microphone separation distance, and frequency [Ernst et al., 2015]. These effects can still limit aeroacoustic studies in an open-jet test section configuration. The hybrid test section type (HTS) configuration is nowadays emerging as the best trade-off for aeroacoustic tests. In the HTS, an acoustically transparent material replaces the hard-walls. In general, a stretched Kevlar cloth is used [Devenport et al., 2013]. This wall treatment confines the flow within the test section, allowing the acoustic waves to propagate through with minimal transmission.
loss. Aerodynamic corrections are still required, though it avoids the severe flow deflection in the open-jet [Brown, 2016]. Coherence loss effects are also significantly reduced because an open-jet shear layer is no longer present. However, self-noise from the Kevlar cloth significantly increases background noise levels in microphone phased array measurements at high frequencies [Bahr et al., 2021]. Different materials can be used to reduce the background noise level [Szoke et al., 2021].

Systematic errors in aeroacoustic wind tunnel tests

Systematic measurement errors are commonly present in aeroacoustic measurements of the trailing-edge noise. The primary sources for errors are:

1. no universal procedure exists to correct microphone phased array measurements to a standard and ideal free-field and free-stream condition,

2. facility-dependent elements such as background noise or slightly different flow conditions can affect the results of aeroacoustic measurements,

3. installation effects, such as boundary layer tripping devices, can significantly alter the noise source.

While the microphone phased array has become the standard technique for aeroacoustic measurements, its results can be different for various reasons. Firstly, the final noise spectrum obtained can depend on the beamforming method used [Merino-Martinez et al., March 2019]. Secondly, the uncertainty of each post-processing method is not clear. A study in [Yardibi et al., 2010] demonstrated that a lower-bound uncertainty of $\pm 1$ dB of the integrated noise levels from the conventional beamforming technique is achievable up to 2.5 kHz. This was achieved by a Monte Carlo approach with component uncertainties set at conservative values. Finally, differences in the software implementation of beamforming algorithms can lead to a noticeable variability of results [Bahr et al., 2017; Sarradj et al., 2017]. Facility dependent effects also influence the aeroacoustic measurements. The comparability of airframe noise measurements in an open-jet and hard-wall test section wind tunnel was investigated in [Oerlemans et al., 2007; Kröber and Koop, 2011; Bahr, 2021]. These studies showed that the coherence loss, induced by the open-jet shear layer, can cause large deviations of absolute noise levels at high frequencies. However, no standard correction method exists while various methods are
available [Biesheuvel et al., 2019, 2021]. The comparability of airframe noise measurements in two different hard-wall test section wind tunnels was studied in [Spehr and Ahlefeldt, March 2019]. They found that, generally, comparable microphone phased array results can be obtained. However, significant differences were still observed, which were assumed to be caused by differences in data acquisition or array calibration. Installation effects have also been reported to influence microphone phased array results. Spurious noise sources at the end plates of an airfoil contaminate the trailing-edge noise source [Tuinstra and Sijtsma, 2015]. Moreover, the tripping devices used to ensure a turbulent boundary-layer at the trailing-edge are known to cause a difference in the measured noise spectra [Ye et al., 2021; dos Santos et al., 2021].

Aeroacoustic measurements using microphone phased arrays can generally be considered as complex systems in which large variations of input variables are not uncommon. Systematic and random input errors, which are typically also correlated by definition, can therefore lead to a non-Gaussian output of aeroacoustic measurements. In other words, the output variance can be high, which can lead to significant differences when comparing, e.g., far-field noise spectra obtained in different wind tunnel facilities. This study aims to evaluate the comparability of trailing-edge noise measurements in a small-scale wind tunnel in different test section configurations. We aim to minimize the systematic errors inherently present in aeroacoustic wind tunnel measurements. Therefore, we performed the measurements in the same wind tunnel, using the same airfoil models and similar boundary layer tripping strategies. Third, the microphone phased array hardware is identical. Last, we used the same beamforming data post-processing algorithm. We evaluate basic aerodynamic corrections for each test section type using a DU97W300 airfoil model. Aeroacoustic measurements were performed on a NACA-0012, NACA-0018 and NACA-63018 airfoil at zero angle of attack. Both far-field noise and unsteady wall-pressure statistics close to the trailing-edge were measured, the latter being important to investigate if the source terms of the trailing-edge noise are similar in each test section configuration. We performed measurements with effective angles of attack below 10° with the NACA-63018 airfoil. Acoustic measurements at large angles of attack were not conducted for two main reasons. Firstly, spurious sources at the side walls influence and contaminate the microphone phased array measurements. Secondly, the increased lift at high angles of attack may increase the flow deflection in the open-jet up to a point where the pressure distribution over the airfoil becomes strongly modified [Brooks and Marcolini, 1984]. This would likely lead to a different characterization of the boundary layer at the trailing-edge and the radiated noise [Moreau et al., 2003], which strongly depends on airfoil geometry and the chord to wind tunnel width ratio.
Chapter 4. Trailing-Edge Noise Comparability in Open, Closed, and Hybrid Wind Tunnel Test Sections.

4.2 Methodology

4.2.1 Wind Tunnel Facility

The measurements described in this study were performed in the Aeroacoustic Wind Tunnel of the University of Twente. A description of the facility and test section can be found in Sec. 3.1.

4.2.2 Microphone Phased Arrays

A microphone phased array comprising 62 GRAS 40PH microphones is used in all test section configuration reported in this study. The analog microphones signals are sampled by 4 NI PXIe-4499 Sound and Vibration modules installed in a NI PXIe-1073 chassis. The acquisition time for the far-field noise measurements is 30 seconds, with a sampling frequency of 65536 Hz. Each microphone’s sensitivity was calibrated prior to the measurements using a GRAS 40AG pistonphone. An overview of the different microphone phased array setups is shown in Fig. 4.1.

Open-jet and Hybrid Test Section Array

In the open-jet test section configuration, the microphones were arranged in a Vogel spiral arrangement shown in Fig. 4.2a. This arrangement provided a good mainlobe-to-sidelobe ratio in a wide frequency range [Sarradj, 2015]. The array has a diameter of 1 m and is placed at a distance of 1.5 m from the centerline of the wind tunnel (see Fig. 4.1b). All microphones were mounted on an open aluminum structure for minimal sound field interference (see [Sanders et al., 2021a]). In the hybrid test section, the same microphone phased array is used at the same distance from the wind tunnel centerline (see Fig. 4.1c). The microphone phased array is placed behind the side-wall with the single Kevlar panel. The opposite side-wall of the hybrid test section comprises three smaller Kevlar panels, shown in Fig. 4.1d. Figure 4.2 shows a performance evaluation of the microphone phased array setups.
4.2. Methodology

Hard-wall Test Section Array

In the hard test section configuration, geometrical constraints forced the microphones to be installed flush-mounted in the wind tunnel side-walls at a reduced distance between the microphone array and the sound source. Aiming to improved comparability of the acoustic measurements, the microphone array was first down-scaled to reproduce identical emission angles. However, preliminary tests demonstrated that this configuration had insufficient performance due to larger main lobes of the point spread functions and increased background noise levels. Therefore, the microphone phased array installed in the hard-wall test section was modified. The Vogel spiral arrangement was kept, but the width of the array was increased and the height slightly decreased to fit the wind tunnel wall. Due to structural limitations, some microphones were moved to a random location in the center region of the array, resulting in the microphone arrangement shown in Fig. 4.2b. The beamwidth and mainlobe-to-sidelobe ratio of the microphone phased array arrangement evaluated from the numerically obtained point-spread-function (PSF) is shown in Fig. 4.3. The beamwidth of the microphone array used in the CTS is clearly smaller, yielding better spatial resolution. Nonetheless, the effect of the array performance on the integrated noise levels is small due to the source power integration method used which accounts for the array’s point-spread-function [Brooks and Marcolini, 1984].

The contamination of noise from the boundary layer on the flush-mounted microphone installation in the hard-wall wind tunnel walls hampers the performance of microphone phased array measurements. To reduce this contamination, the microphones were installed in an optimized mounting system, derived from the literature [Jaeger et al., 2000; Fleury et al., 2012]. The microphones are recessed by 10 mm behind a stretched Kevlar cloth in a conical aperture, with an opening angle of 110°. The recession isolates the microphone from the boundary layer, whereas the Kevlar cloth further separates the flow from the microphone while still allowing for the propagating of a sound wave with minimal transmission loss. Figure 4.4 shows a schematic of the microphone installation in the hard-wall test section.
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Figure 4.1: Experimental setups.

(a) Hard-wall test section (CTS).
(b) Open-jet test section (OTS).
(c) Hybrid test section (HTS).
(d) Photo of the Hybrid wind tunnel test section and anechoic chamber.

Figure 4.2: Microphone locations relative to the airfoil and wind tunnel.

(A) Vogel spiral arrangement used in the open-jet (B) Modified Vogel spiral arrangement used in the hard-walled test section.
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(a) Beamwidth. (b) Mainlobe to sidelobe ratio.

**Figure 4.3:** Microphone phased array performance evaluated on a (1 m × 1 m) search grid.

**Figure 4.4:** Schematic cross-section of the hard-wall test section panel.

4.2.3 Background Noise

Background noise levels in wind tunnels play an important role in obtaining accurate microphone measurements. Figure 4.5 shows the measured background noise level in each empty test section type by showing the power spectral density (PSD) of the centermost microphone in each array for a flow velocity of 30 and 50 m/s. The spectra are normalized to a reference distance of 1 m from the centerline of the wind tunnel. Note that the microphone spectra are already corrected according to the methods described in section 4.3.1. The spectra show that boundary layer self-noise is not entirely eliminated in the CTS at frequencies below approximately 1 kHz in the presented setup. This is
observed from the low frequency hump in the spectrum. In the mid-frequency range (1-6 kHz), we see that the background noise levels in each test section type are comparable. In the high frequency range (>6 kHz), we observe that the Kevlar cloth used in the HTS and CTS configurations produces a significant amount of noise, which is typical for this material. The airfoil noise in our measurements is typically measured below this frequency range and will therefore not influence the measurements presented in this study.

![Figure 4.5: Background PSD of the center most microphone in the array normalized to a distance of 1 m from the wind tunnel centerline. U$_0$ = 30 m/s (—) and 50 m/s (---).](image)

### 4.2.4 Airfoil Models and Instrumentation

Four airfoils are used in the present study. Aerodynamic boundary corrections are evaluated with measurements of a DU97W300 airfoil. Aeroacoustic measurements were performed on a NACA-0012, a NACA-0018 and a NACA-63018 airfoil. The NACA-0012 and NACA-0018 are airfoil geometries used in the trailing-edge noise category of the BANC (Benchmark for Airframe Noise Computations) workshop and are widely used for benchmarking with the BPM model [Brooks et al., 1989]. The NACA-63018 is an airfoil geometry used in the HAWT (Hybrid Anechoic Wind Tunnel) Workshop as a benchmark exercise for hybrid test sections. These benchmark databases allow us to compare the measurements with data from other wind tunnel facilities so that facility-dependent effects can be identified in the future. A description of the airfoil models and their instrumentation can be found in Sec. 3.2.
4.3 Acoustic Beamforming Method

Beamforming output maps were generated with the in-house codes of the University of Twente. These codes were benchmarked with the array benchmark database [Sarradj et al., 2017; Bahr et al., 2017]. For conventional beamforming results, a search grid ranging from $x = -0.3$ m to 0.3 m and $y = -0.4$ m to 0.4 m with a spatial resolution of 0.01 m is used. The cross-spectral-matrix (CSM) is computed by taking a window size of 8192 samples and applying a Hanning window to each block with a 50% overlap. Diagonal removal is applied to the CSM to remove any self-noise picked up by the microphones. No microphone weighting function is applied to the beamforming process.

Trailing-edge noise in the beamforming maps is isolated with the source-power-integration (SPI) technique [Brooks and Humphreys, 1999]. The region of integration (ROI) is centered in the middle of the span and has a length of 0.3 m in the spanwise direction and a width of 0.3 m in the chordwise direction. The noise levels, which were over 6 dB below the peak value in the beamforming map, were neglected in order to reduce the contribution from other sources than the trailing-edge. The obtained far-field integrated spectra are scaled to a reference distance and airfoil span of 1 m.

4.3.1 Test Section Dependent Corrections

As was discussed in the introduction, a complete overview of all necessary acoustic propagation corrections in each test section type will not be given here. Instead, we will discuss the most relevant corrections which were applied here so that comparable results are ensured.

In the open-jet, sound waves are convected with the flow and propagate through the shear layer of the jet where they are refracted. Amiet’s shear layer correction method [Amiet, 1978] is applied to account for this effect following the procedure described in [Bahr et al., 2011]. We consider that the coherence loss induced by the shear layer is negligible for the configuration used in this study. Previous measurements and modeling of the coherence loss in this test section with the same microphone phased array have shown that the coherence loss effect on integrated beamforming levels is below 2 dB for frequencies below 10 kHz and below 1 dB for frequencies below 5 kHz with a free-stream Mach number below 0.15 [Sanders et al., 2021b].
In the hybrid test section, sound waves have to propagate through the Kevlar cloth, leading to transmission loss. We measured this transmission loss following the procedure described in [Devenport et al., 2013]. A Wavecor model TW013WA01 speaker was placed on one side of the test section and a GRAS 40PH microphone was placed on the opposite side. The white noise from the speaker was then measured in two conditions. First, with the Kevlar side wall panels installed and second with the microphones at the same position, but without the side-walls of the wind tunnel. Moreover, the other walls of the test section were covered in acoustic foam to create a temporal anechoic environment. The measured transmission loss curve is accounted for in the CSM computation. Even though the correction is small (<1 dB at 10 kHz), we include it because of the simplicity of the correction. We also apply the sound wave convection and refraction correction proposed by [Amiet, 1978].

The hard-wall test section measurements require additional corrections since the microphones are mounted in the walls of the wind tunnel, see section 4.2.2 and Fig. 4.4. This leads to a pressure doubling of the microphone signals and a different frequency response curve. We therefore performed calibration measurements to determine the pressure field response curve using a temporal anechoic enclosure created inside of the test section. The microphone’s response to white noise from a speaker was measured when they were mounted in the walls of the wind tunnel and in a free-field condition with the wall mounting. This calibration procedure automatically accounts for the transmission loss through the Kevlar cloth, which is on top of the conical aperture of the microphone, see Fig. 4.4. No directivity corrections are applied since this effect is small based on a comparison of the measured spectra from a measurement. A flow convection correction is also applied, following the simplified shear layer correction method in [Oerlemans, 2009].

4.4 Results

4.4.1 Comparability of Aerodynamic Measurements

In aeroacoustic studies of trailing-edge noise, we are most interested in the pressure distribution and lift coefficient of an airfoil model. The pressure distribution is important because it determines the final state of the surface pressure fluctuations beneath the turbulent boundary layer, which are scattered at the trailing-edge and produce the noise.
4.4. Results

Considering this, corrected aerodynamic data should always be carefully assessed when trailing-edge noise measurements are concerned. While basic aerodynamic corrections allow for a simple correction to an effective angle of attack in free-air condition, they are only an approximation of the streamline curvature effect. In practice, the pressure distribution in free-air is never exactly replicated in the wind tunnel because of the boundary interference effects. The fundamental characteristics of the pressure distribution will likely be the same, but, e.g., the (suction) peak pressure and the pressure gradient can be affected. This is especially the case when the ratio between the airfoil chord and wind tunnel width is large or when high-lift models are tested. Consequently, the flow separation and stall behavior of an airfoil will be affected. Subsequently, it will affect the development of the turbulent boundary layer and its final characteristics at the trailing-edge.

The incompressible boundary corrections for the lift coefficient and angle of attack in each test section type are therefore addressed with a DU97W300 airfoil. According to the authors’ knowledge, this kind of comparison between test section types has not been shown in the literature before. The applied corrections can be found in Appendix 4.5. Figure 4.6a shows the uncorrected section lift coefficient as a function of the geometric angle of attack \( \alpha_g \) and lift coefficient \( \alpha_e \) as a function of the lift coefficient from measurements of the untripped DU97W300 airfoil model at a chord-based Reynolds number of \( 9 \times 10^6 \) and free-stream Mach number of 0.16. The figure illustrates the expected aerodynamic behavior in each test section. The largest angle of attack correction is required in the OTS because of the jet flow deflection induced by the model. This correction increases linearly with the lift coefficient, as can be expected from the angle of attack correction in Eq. 4.1 of Appendix A. Much smaller angle of attack corrections are required in the HTS. The angle of attack correction required in the HTS does not increase linearly compared to the OTS. This is because the angle of attack correction, mainly due to the Kevlar’s porosity, is approximately proportional to the square root of the lift coefficient as can be seen in Eq. 4.3 of Appendix A, see also [Devenport et al., 2013]. The smallest correction is required in the CTS, which is only due to the streamline curvature effect.

Figure 4.6b shows the corrected \( C_l - \alpha \) curve of the same measurements. The corrected results collapse well in the linear regime of the curve, as expected. The pressure distribution for \( \alpha_g = 0^\circ \) (\( \alpha_{e,OTS} = -0.6^\circ \), \( \alpha_{e,CTS} = 0^\circ \), \( \alpha_{e,HTS} = -0.3^\circ \)) in all test sections is shown in Fig. 4.7a. Even though the effective angles of attack in this figure do not match exactly, the figure illustrates that the overall shape and characteristics of the
pressure distributions at a low angle of attack match well. At larger angles of attack, the maximum lift and stall behavior in each test section is different. This behavior is also observed, e.g., in [Devenport et al., 2013] for a NACA-0012 airfoil. Figure 4.7b shows the measured pressure distribution for a larger angle of attack at \( \alpha_{g,OTS} = 17^\circ \) (\( \alpha_{e,OTS} = 11.1^\circ, C_l = 1.31 \)), \( \alpha_{g,CTS} = 10^\circ \) (\( \alpha_{e,CTS} = 10.1^\circ, C_l = 1.32 \)), \( \alpha_{g,HTS} = 10.1^\circ \) (\( \alpha_{e,HTS} = 10.1^\circ, C_l = 1.31 \)). The effective angle of attack is calculated from the boundary correction equations in the Appendix. While the section lift coefficient is similar, the pressure distribution in the open-jet test section is different. The suction peak value is higher, while the pressure gradient on the suction side is different. The figure illustrates the effect of the test section type on the characteristics of the pressure distribution at large angles of attack when the ratio between airfoil chord (or lift) and wind tunnel width is small. This ratio is important because it is mainly the lift force generated by the airfoil, which determines how much the momentum direction of the open-jet (i.e., jet deflection) is affected. It can be argued in this case that the difference in \( C_p \) distribution will affect the development and final characteristics of the turbulent boundary layer at the trailing-edge, leading to a different noise radiation.

The overall agreement in Fig. 4.6b is good, showing that basic aerodynamic corrections are sufficiently accurate for 2D airfoil models in this wind tunnel facility. The lift curve in Fig. 4.6b is observed to be lower than the measurements performed by [Timmer and van Rooij, 2003] and [Baldacchino, 2019]. Moreover, the stall behavior is less abrupt. Both effects can be related to the influence of the Reynolds number effect on the lift-alpha curve. The \( C_p \) distribution on the suction side near the airfoil’s trailing-edge is observed to flatten towards the trailing-edge. As a result, a steeper pressure gradient is induced on the suction side, which will reduce the total lift. This effect may be due to a flow separation effect caused by a lower Reynolds number or due to boundary interference effects, which are not corrected for.

Next, the aeroacoustics of measurements of a NACA-63018 airfoil at low effective angles of attack of 0°, 4° and 8° are compared. Trapezoidal integration of the pressure distribution cannot directly calculate the lift coefficient of this airfoil due to an incomplete distribution of the pressure ports along the entire airfoil surface. Therefore, the geometric angle of attack was related to the aerodynamic angle of attack for nonzero angle of attack in a different way. This was done in-situ by adjusting the angle of attack until we achieved a satisfactory agreement with the prescribed free-stream \( C_p \) distribution from XFOIL. The measured \( C_p \) distributions in each test section type for an effective angle of attack of 4° and 8° are shown in Fig. 4.8a and Fig. 4.8b, respectively, for a
free-stream velocity of 50 m/s with a chord-based Reynolds number of $0.65 \cdot 10^6$. No differences were observed between the $C_p$ distributions within the measured Reynolds number range. The pressure distributions are consistent between the test section types. Minor differences arise around the suction peak, which may be influenced by the tripping devices that are used. At larger angles of attack, a flattening of the $C_p$ distribution on the suction side near the trailing-edge is seen. As a result, the $C_p$ distribution on the suction side is lower than that predicted by XFOIL. This is caused by the tripping device because we did not observe this effect for an untripped airfoil. The geometric angles of attack obtained with this method are used for the acoustic comparability results in the next sections.

![Diagram](image1)

Figure 4.6: $C_l$-α curve and angle of attack correction of the untripped DU97W300 airfoil with $U_0 = 55$ m/s and $Re_c = 0.9 \cdot 10^6$, $Re_c = 3 \cdot 10^6$ in [Timmer and van Rooij, 2003] and $Re_c = 2 \cdot 10^6$ in [Baldacchino, 2019]. $C_l$ is the corrected lift coefficient.

![Diagram](image2)

Figure 4.7: $C_p$ distribution of the untripped DU97W300 airfoil.
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4.4.2 Comparability of the Aeroacoustic Source

Unsteady Wall-Pressure Statistics

This section examines the similarity of the wall-pressure statistics beneath the turbulent boundary layer near the airfoil trailing-edge. According to the model proposed by [Amiet, 1976], two empirical parameters determine the far-field radiated noise [Brooks and Hodgson, 1981]. These are the wall-pressure spectrum $\phi_{pp}$ and spanwise correlation length $l_y$ measured near the trailing-edge. The wall-pressure spectrum can be measured by a single microphone sensor, whereas the spanwise correlation length is obtained from the integral of the squared spanwise coherence function over all values of separation distances [Stalnov et al., 2016]. This integration can be accurately performed if a sufficient amount of spanwise microphones at different spanwise separation distances are placed. In our current setup, this is not the case. However, experiments have shown that the spanwise coherence function can also be modeled as having an exponential decay with microphone separation distance [Brooks and Hodgson, 1981]. Using this assumption and the Corcos model [Corcos, 1964] therefore allows us to use the coherence between a microphone pair along the span as an indicator for the spanwise correlation length scale. Both the spanwise and streamwise coherence between microphone pairs will be examined here since they characterize the relevant spatial and temporal structures, respectively.

Figure 4.9a shows the measured wall-pressure spectra from a single microphone for the NACA-0012 (mic. 5), NACA-0018 (mic. 5) and NACA-63018 (mic. 2) at $U_0=25$ m/s,
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Re$_c$ = 330,000 and 0° angle of attack. See Fig. 3.25 for the microphone indices and locations. Measurements at higher flow velocities were not performed with the NACA-0012 and NACA-0018 because the Knowles microphones are saturated in those conditions. Therefore, we limit ourselves to the data with a free-stream velocity of 25 m/s unless stated otherwise. The wall-pressure spectrum is scaled by the outer boundary-layer parameters, of which the displacement thickness is taken from XFOIL. The frequency is normalized by the Strouhal number using the boundary layer thickness, which is determined from the displacement thickness from XFOIL [Stalnov et al., 2016]. We see that the wall-pressure spectra collapses within ±1 dB when comparing the measurements of the same airfoil in the different test sections. Note that the wall-pressure spectrum is taken at x/c=0.86 on the NACA-63018 airfoil and at x/c=0.93 on the NACA-0012 and NACA-0018 airfoil. Although the wall-pressure spectra are normalized, a difference can still be seen between the different airfoils. This is expected to be due to the difference in pressure distribution and pressure history leading up to the trailing-edge, which influences the final state of the turbulent boundary layer.

The streamwise and spanwise coherence between microphone pairs is shown in Fig. 4.9b for the NACA-0012 (mic. 5 and mic. 2), in Fig. 4.9c for the NACA-0018 (mic. 5 and mic. 2), and in Fig. 4.9d for the NACA-63018 (mic. 3 and mic. 2). The measurements show a close agreement of the wall-pressure statistics regardless of the test section configuration at 0° angle of attack. These unsteady wall-pressure measurements therefore show that the sound source does not vary strongly among the different test section configurations. Based on this, we can study the comparability of trailing-edge noise measurements in different test section configurations.

Figure 4.10 shows the wall-pressure spectra at both the pressure and suction side for the NACA-63018 airfoil at an angle of attack of 4° and 8° with U$_0$ = 25 m/s (Re$_c$ = 330,000) and U$_0$ = 30 m/s (Re$_c$ = 400,000) respectively. The wall-pressure spectrum is shown for U$_0$ = 30 m/s at the 8° angle of attack because the U$_0$ = 25 m/s data was contaminated. Unfortunately, this data is not available from the HTS measurements, which are therefore not shown. For both effective angles of attack, we see that the wall-pressure spectra on both the suction and pressure side match well within 1 dB.
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Figure 4.9: Measured wall-pressure statistics of the NACA-0012 (□), NACA-0018 (○) and NACA-63018 (◦). $U_0 = 25$ m/s, $Re_c = 330,000$, $\alpha = 0^\circ$ and a tripped boundary layer at $x/c=0.05$.

Figure 4.10: Measured wall-pressure spectrum (PSD) on the NACA-63018. Pressure side (—) and suction side (---) with $p_{ref} = 20 \ \mu$Pa.

(a) $\alpha_e = 4^\circ$, $U_0 = 25$ m/s.

(b) $\alpha_e = 8^\circ$, $U_0 = 30$ m/s.
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4.4.3 Comparability of the Far-Field Noise

Absolute Noise Levels

We now compare the far-field noise measurements obtained from the microphone phased array measurements. The acoustic data is analyzed using the conventional beamforming method in the frequency domain and far-field integrated spectra were obtained using the source power integration technique [Brooks and Humphreys, 1999]. One-third octave beamforming maps of the NACA-63018 measurements with $U_0 = 30$ m/s and $Re_c = 400,000$ are shown in Fig. 4.11 for the center frequencies of 1 kHz, 2.5 kHz and 5 kHz. We normalized the beamforming maps to a reference distance of 1 m. The beamforming maps show that the trailing-edge noise is well observable even in the CTS measurements where the contamination from the boundary layer self-noise on the walls is still present. In terms of spatial resolution, the CTS measurements show better results, as is to be expected from the array’s performance. Noise levels in the CTS beamforming maps are up to 3 dB lower at higher frequencies for this particular measurements. However, this difference is not systematically seen in other measurements and is therefore considered a random error, as will be shown next.

Next, the integrated noise levels for each airfoil at various flow conditions are compared. We do not include frequencies below 500 Hz because the array resolution in all test sections is low, making integrated noise levels less reliable. Frequencies larger than 5 kHz are not included because the trailing-edge noise becomes less distinguishable at these frequencies due to a low signal-to-noise ratio. All integrated spectra are normalized to a reference distance of 1 m and airfoil span of 1 m. Note that the integrated levels from the CTS measurements are higher than those shown in [Sanders et al., 2021a]. This is because a mistake in the source power integration method was corrected here. Figure 4.12 shows the integrated noise levels from the NACA-0012, NACA-0018 and NACA-63018 at 0° angle of attack with $U_0 = 30$ m/s ($Re_c = 400,000$) and $U_0 = 50$ m/s ($Re_c = 660,000$). Overall good agreement of the noise spectra is seen in all test sections for all airfoils. Larger noise level deviations, up to $\approx 4$ dB are seen at low frequencies (1 kHz) for a flow velocity of 50 m/s in the OTS. Moreover, the noise level in the 500 Hz frequency bin is consistently higher in the CTS. This deviation can be due to decreased performance of the beamformer due to lower resolution and increase background noise or due to reverberation effects.
A comparison of the measured integrated noise levels from the OTS with results from the literature [Brooks et al., 1989; Oerlemans and Migliore, 2004] is also presented. The experimental data was normalized to a reference distance of 1 m, airfoil span of 1 m, and free-stream velocity of 30 m/s following a 5\textsuperscript{th} power law scaling for airfoil trailing-edge noise. The spectrum is also plotted as a function of the Strouhal number based on the airfoil chord. The NASA data [Brooks et al., 1989] is of a NACA-0012 airfoil of 0.2286 m chord which was tripped using a random distribution grit in strips at x/c=0.2, whereas the NLR data [Oerlemans and Migliore, 2004] is of a NACA-0012 airfoil of 0.282 m which was tripped with 0.25 mm thick zigzag tape at x/c=0.02 and x/c=0.05 on the suction and pressure side, respectively. The measurements were all performed at the same directivity angle, namely Θ = 90°. Overall, we find good agreement of the noise levels in the intermediate Strouhal number range, especially to the NLR data. However, both high and low frequencies show larger discrepancies. An obvious explanation for these differences cannot be given although it is suggested in [Oerlemans and Migliore, 2004] that the difference is possibly due to the use of different tripping devices.

Absolute noise levels for the effective angles of attack of 4° and 8° are shown in Fig. 4.13 for the NACA-63018 airfoil model. The geometric angles of attack were determined from the measurements shown in Fig. 4.8. For a free-stream velocity of 30 m/s, we see good overall agreement within ≈2 dB. The noise levels are seen to increase in the CTS at high frequencies, which is caused by an increase in the background noise level. For 50 m/s, the integrated spectral levels are found to be up to 3 dB lower in the HTS. Noise levels are higher in the CTS above a frequency of roughly 3 kHz, which is due to an increase in the background noise level and the contamination from spurious sound sources in the test section. Larger angles of attack are not shown or measured in any of the test section configurations due to several reasons. Firstly, the tripping device was installed at x/c=0.05. For large angles of attack, the stagnation point would pass this location. Secondly, extraneous noise sources were found to contaminate the beamforming maps at the sidewall interface for large angles of attack in all test section configurations. These noise sources are caused by the interaction of the boundary layer on the wall with the leading-edge of the airfoil. This lead to the trailing-edge noise source being poorly identifiable, especially when trailing-edge serrations were installed.

The mean difference between the integrated spectra of the absolute noise levels for all measurements is shown in Fig. 4.14. The measurements from each airfoil at all sampled flow velocities ($U_0 = 20, 30, 40$ and $50$ m/s) and effective angles of attack ($\alpha_e = 0°,$
4° and 8°) are considered to be statistically independent in each frequency bin. Beam-forming maps where the trailing-edge noise was not observable were omitted from the analysis. Those beamforming maps were generally contaminated with either spurious background noise from the wind tunnel or the side-wall junction. These contaminations are considered as effects due to the facility and installation, which are not caused by the test section configuration itself. Therefore, these beamforming maps were emitted from the analysis. The 95% confidence intervals are calculated using the Student’s t-distribution. It can be seen that overall, the noise levels are found to be well within \(\pm 3\) dB, as indicated with the dashed line in the figure. At the lowest frequency bin, the noise level in the CTS is consistently higher than in the other test sections. This can be attributed to either a poorer performance of the CTS array in that frequency range or reverberation effects. Overall, the agreement in absolute noise levels between the OTS and HTS is satisfactory.
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Figure 4.11: One-third octave beamforming maps for the NACA-63018 at $\alpha = 0^\circ$, $U_0 = 30$ m/s, $Re_c = 400,000$. Scaled to a reference distance of 1 m.
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(a) NACA-0012.

(b) NACA-0018.

(c) NACA-63018.

(d) A comparison with NACA-0012 measurements from literature ($St_c = f_c/U_0$).

Figure 4.12: Integrated noise levels in one-third octave bands normalized to a reference distance of 1 m, airfoil span of 1 m. Free-stream velocity $U_0 = 30$ m/s (—) and $U_0 = 50$ m/s (—). $Re_c = 400,000$ and $660,000$ with $\alpha = 0^\circ$.

(a) $\alpha_e = 4^\circ$.

(b) $\alpha_e = 8^\circ$

Figure 4.13: Integrated noise levels in one-third octave bands normalized to a reference distance of 1 m, airfoil span of 1 m. Free-stream velocity $U_0 = 30$ m/s (—) and $U_0 = 50$ m/s (—). $Re_c = 400,000$ and $660,000$ with the NACA-63018.
Relative Noise Levels From a Noise Reduction Device

To further investigate the comparability of noise levels, we investigate the noise reduction levels from trailing-edge serration add-ons mounted on the airfoils. Figure 4.15 shows the geometry of the trailing-edge serration, which was identical on all airfoils. This geometry is based on a study in [Gruber et al., 2011] which showed that the strongest noise reduction from trailing-edge serrations is obtained when they have an aspect-ratio (height/width) of 2:1. Therefore, we used a 1 mm thick trailing-edge serration with a length of $2h=20$ mm and width of $\lambda=10$ mm. The length $2h$ of the serrations corresponds to 10% of the baseline chord for the airfoil models. All trailing-edge serrations were installed as add-ons, meaning that they extend the total chord of the airfoils.

**Figure 4.15:** Geometry of the trailing-edge serrations.
The measured noise reduction levels for the airfoils at zero angle of attack are shown in Fig. 4.16 in one-third octave bands. We define the noise reduction at each frequency bin as $\Delta \text{SPL} = \text{SPL}_{\text{straight}} - \text{SPL}_{\text{serrated}}$ so that positive levels show noise reduction. For a free-stream velocity of $30 \text{ m/s}$ we see a maximum variation of $\approx 2 \text{ dB}$ in the noise reduction levels between the different test sections in Fig. 4.16. For the case of a free-stream velocity of $50 \text{ m/s}$ the noise reduction from the open-jet test section is slightly higher than the other test sections. For the NACA-0012 and NACA-0018 airfoil results, we see a maximum difference between noise reduction levels of $5 \text{ dB}$. Overall, we find a good agreement in both the spectral shape and level of the noise reduction in all test section configurations for the zero angle of attack condition. When comparing the noise reduction levels for the $30 \text{ m/s}$ and $50 \text{ m/s}$ free-stream velocities, we see that the noise reduction increases with flow velocity. This is related to the effect that the noise reduction is a function of $\lambda/\delta$, whereas the frequency shift is related to the Strouhal number ($f\delta/U_0$) scaling [Gruber et al., 2011]. Because the noise reduction obtained depends on the boundary layer parameters, different noise reduction levels are also found when comparing the results of each airfoil.

Another comparison with measurements from the literature [Arce-León et al., 2016] is made in Fig. 4.16d. This measurement data was taken at identical test conditions and thus is not normalized for simplicity. We find excellent agreement of both the noise level and spectral shape of the noise reduction. Results for the NACA-63018 at $4^\circ$ and $8^\circ$ effective angle of attack are shown in Fig. 4.17. The agreement in spectral shape and noise reduction levels for an effective angle of attack of $4^\circ$ is the same as that for the zero angle of attack case. For the largest effective angle of attack of $8^\circ$, we see larger variation in the noise reduction levels. The noise reduction levels differ by more than $5 \text{ dB}$ at high frequencies. This variation is strongly related to a decrease in signal-to-noise ratio, as it was difficult to distinguish the trailing-edge noise source from other sources in the beamforming map, e.g., spurious sources from the side-wall junction.

The mean difference between the integrated spectra of the relative noise levels for all measurements is shown in Figure 4.18. Beamforming maps where the trailing-edge noise was not observable were again omitted from the analysis. Each measurement is assumed to be statistically independent at each frequency bin for all free-stream velocities ($U_0 = 20$, $30$, $40$ and $50 \text{ m/s}$) and effective angles of attack ($\alpha_e = 0^\circ$, $4^\circ$ and $8^\circ$). The $95\%$ confidence intervals are calculated using the Student’s $t$-distribution. Overall, we see good agreement of the relative noise levels of $\pm 1-3 \text{ dB}$ between all test section types.
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Figure 4.16: Integrated noise reduction levels of the trailing-edge serrations in one-third octave bands. Free-stream velocity \( U_0 = 30 \text{ m/s (---)} \) and \( U_0 = 50 \text{ m/s (---)} \). \( \text{Re}_c = 400,000 \) and 660,000 with \( \alpha = 0^\circ \).

(a) \( \alpha_e = 4^\circ \).

(b) \( \alpha_e = 8^\circ \).

Figure 4.17: Integrated noise reduction levels of the trailing-edge serrations in one-third octave bands. Free-stream velocity \( U_0 = 30 \text{ m/s (---)} \) and \( U_0 = 50 \text{ m/s (---)} \). \( \text{Re}_c = 400,000 \) and 660,000 for the NACA-63018.
4.5 Conclusions

This study presents the comparability of trailing-edge noise measurements in an open-jet, hard-wall and hybrid test section wind tunnel. Systematic measurement errors in the aeroacoustic measurements are minimized, such as the wind tunnel, test section size, airfoil model size, tripping method, and acoustic beamforming processing, permitting the isolation of random errors only. Basic aerodynamic boundary corrections were applied to assess their capability of correcting the geometric angle of attack and lift coefficient from measurements of a DU97W300 airfoil model. The corrected $C_l - \alpha$ curve from each test section type collapses well in the linear regime of the curve. However, the stall behavior differs in each test section type. Moreover, for large angles of attack, it was shown that important characteristics of the pressure distribution over the airfoil change significantly, even though the section lift coefficient is of the same value. This different aerodynamic behavior cannot be disregarded when comparing corrected results from different test section types. This is especially the case when the airfoil is relatively large compared to the test section size. Although no acoustic measurements were performed of the DU97W300 airfoil, we expect this change in pressure distribution to also affect the radiation of trailing-edge noise due to the sensitivity of trailing-edge noise to boundary layer properties.

Aeroacoustic comparability measurements were therefore limited to small angles of attack. Unsteady wall-pressure measurements on the NACA-63018 airfoil at low angles of...
Chapter 4. Trailing-Edge Noise Comparability in Open, Closed, and Hybrid Wind Tunnel Test Sections.

attack show that the trailing-edge source has identical characteristics regardless of test section configuration when an identical $C_p$ distribution is reproduced.

A comparison of the absolute and relative noise level shows that far-field trailing-edge noise from microphone phased array measurements are comparable regardless of test section type within an acceptable margin. A systematic measurement methodology combined with a standardized method to post-process microphone phased array measurements is essential. The test section dependent corrections applied to the microphone data, described in Sec. 4.3.1, led to the comparable microphone phased array measurements between the test section type. The same microphone phased array setup was used in the open-jet and hybrid test section measurements. Coherence loss effects were negligible in the frequency range considered in this study. An optimized microphone mounting system and arrangement was used to perform the measurements in the closed test section configuration. A pressure field calibration was applied to correct to a standard ideal free-field condition. The absolute noise levels in all test section configurations are in close agreement both in terms of spectral shape and noise levels. Mean differences are found to be within ±3 dB, based on a statistical analysis of all measurements. Larger differences between the test section results are still observed in individual measurements. These differences are attributed to a lower signal-to-noise ratio or incidental noise contamination from other sources, such as the side-wall junction source. A comparison of relative noise levels measurements, using trailing-edge serrations, showed an agreement of ±1-3 dB, based on a statistical analysis of all measurements. Individual measurements can show larger differences which is attributed to a lower signal-to-noise ratio and contamination from side-wall noise sources, especially with increasing angle of attack. While the wall-pressure measurements showed only negligible differences in the sound source statistics between the test section types, larger differences were seen when comparing the far-field noise. These results also highlight the sensitivity to input errors of the far-field spectrum output from microphone phased array measurements.

Further research on the comparability of trailing-edge noise measurements in different test section configurations should focus on measurements of cambered airfoils or airfoils in high-lift, i.e., large angle of attack conditions. It is likely that the boundary-layer behavior on these types of airfoil measurements will be more heavily influenced by the test section type, both for tripped boundary layers and those with natural transition. Consequently, the stall behavior of airfoils will be affected, hampering the understanding and modeling of the influence of, e.g., boundary layer transition and separated flow on
4.A. Aerodynamic Corrections

the noise radiated by airfoils. However, the combined effect of the Reynolds-number on the boundary-layer behavior can even further complicate such a study.

Appendix

4.A Aerodynamic Corrections

In the open-jet test section, the effective angle of attack $\alpha_e$ (in degrees) can be calculated from the geometric angle of attack $\alpha_g$ in [Brooks and Marcolini, 1984] with

$$\alpha_{e,OTS} = \alpha_{g,OTS} - \frac{180}{\pi} \left( \frac{\sqrt{3} \sigma}{\pi} C_l - \frac{2 \sigma}{\pi} C_l - \frac{\sigma}{\pi} (4C_{M_1}^{\frac{1}{4}}) \right)$$  (4.1)

where $\sigma = \left( \frac{2 \pi^2}{18} \right) \left( \frac{c}{h} \right)^2$ with $c$ the airfoil chord and $h$ the wind tunnel width. No corrections are applied to the sectional lift coefficient $C_l$ since we assume that solid blockage, wake blockage, and buoyancy can be neglected. Both $C_l$ and $C_{M_1}^{\frac{1}{4}}$ are computed from the measured pressure distribution using the trapezoidal integration method. In the closed test section, the effective angle of attack can be calculated in [Barlow et al., 1999] with

$$\alpha_{e,CTS} = \alpha_{g,CTS} + \frac{57.3 \sigma}{2\pi} \left( C_{l,u} + 4C_{M_1}^{\frac{1}{4}} \right)$$  (4.2)

where $C_{l,u}$ is the uncorrected section lift coefficient. The corrected section lift coefficient takes into account the solid wake blockage with $C_l = C_{l,u}(1 - \sigma)$. Wake blockage and buoyancy are neglected. The effective angle of attack in the hybrid test section can be calculated in [Devenport et al., 2010] with

$$\alpha_{e,HTS} = \alpha_{g,HTS} - \frac{\text{sgn}(C_l)C}{U_\infty} \left( \frac{1}{2} \rho U_\infty^2 \left| \frac{C_l}{\pi c A} \right| \right)^n$$  (4.3)

where $C = 0.03879$ is a constant which is related to the porosity of the Kevlar cloth and was determined from measurements [Devenport et al., 2010], $A = 1$ and $n = 0.5743$. These constants are dependent on the material properties of the Kevlar. Because we use the same Kevlar material as in [Devenport et al., 2010], we assume these constants to be identical in our setup. The section lift coefficient $C_l$ is uncorrected because we assume that solid blockage, wake blockage, and buoyancy can be neglected. Note that
the angle of attack correction does not take into account the deflection of the Kevlar walls, which is assumed to have a negligible influence on the angle of attack correction here, as explained in [Devenport et al., 2013].

References


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Chapter 5

The Sweep Angle Effect on Slat Noise Characteristics of the 30P30N High-Lift Model in an Open-Jet Wind Tunnel.


Abstract

Small-scale wind tunnel tests are performed to investigate the effect of wing sweep on the characteristics of the leading-edge slat noise of the 30P30N high-lift research model. These measurements are conducted in the anechoic wind tunnel facility at the University of Twente. A comparison of the noise characteristics is made for various angles of attack at a chord-based Reynolds number of $0.9 \cdot 10^6$ and free-stream Mach number of around 0.15. The high-lift model is positioned at $0^\circ$ and $30^\circ$ sweep angle. Static pressure tap measurements are used for the aerodynamic characterization of the wing. Microphone phased array, and arc measurements provide sound source localization, quantification, and directivity properties. The conventional beamforming method in the frequency domain and the DAMAS method are applied to isolate the noise radiated by the 'clean'
slat. The 30° wing sweep is found to have a negligible effect on the broadband noise and the characteristic noise mechanisms present in the far-field spectrum. A similar Mach number power-law scaling of $M^4$ is observed as for the unswept case. The open-jet results are found to be comparable with other wind tunnel results of the 30P30N wing model under similar pressure distribution conditions.

### 5.1 Introduction

The reduction of aircraft noise is of primary concern to aircraft manufacturers due to the societal and environmental impact that aircraft noise has on the areas near airports. Of specific attention nowadays is the airframe noise component due to recent developments in engine noise reduction. The major sources of airframe noise are known to be the landing gears and the high-lift devices (i.e., wings) [Fink, 1977] of the airplane. The high-lift device noise is considered the most significant sound source for regional aircraft. More specifically, the leading-edge slat element is mainly responsible for the noise generated by the high-lift device [Dobrzynski, 2010].

Leading-edge slat noise is characterized by several noise mechanisms that collectively contribute to the far-field radiated noise spectrum (Fig. 5.1). Firstly, broadband noise is generated by the scattering of the turbulent boundary layers at the trailing edge of the slat [Guo, July 2012]. Tonal peaks in the low-frequency range are presumed to be due to a feedback loop between the shear layer impingement near the trailing edge and the coherent structures originating from the slat cusp [Terracol et al., 2016; Pascioni and Cattafesta, 2018] (see Fig. 5.2). This phenomenon is considered to be similar to the so-called Rossiter modes in cavity flow. A narrowband hump in the high-frequency range may be present. This contribution to the noise spectrum is associated with the vortex shedding from the slat trailing edge due to its finite thickness (Fig. 5.2) [Jenkins et al., 2004; Pascioni and Cattafesta, 2018]. The tonal peaks and vortex shedding hump are considered to be caused by a low Reynolds number effect that is present in small-scale wind tunnel testing.

Several high-lift device noise studies have been performed in small-scale wind tunnel facilities [Fleury et al., 2015; Kolb et al., 2007; Kröber and Koop, 2011; Li et al., 2017; Jawahar et al., 2019; Pascioni and Cattafesta, 2016; Perennes and Roger, 1998]. These studies have been beneficial regarding the localization and quantification of far-field radiated noise. They have also allowed the quantification of near-field flow properties.
that support understanding the noise generation mechanisms. While most studies are conducted in a closed test section, the open-jet (i.e., free jet) test section is still preferred for acoustic purposes. The major advantage of using an open-jet test section is to perform free-field measurements in the acoustic far-field. Also, microphone phased arrays for sound localization and microphone arcs for directivity measurements can be used. The disadvantage of an open-jet test section is its strong susceptibility to flow deflection, especially when wings in the high-lift configuration are used. This flow deflection and blockage are known to result in premature separation of the boundary layer on the flap element of the high-lift wing leading to full stall [Choudhari and Lockard, 2015]. Small angles of attack can still be studied, which is relevant to the airplane’s approach condition.

Most experimental and numerical studies of slat noise involve 2D wings in the high-lift configuration. However, in reality, most of the airplane geometries considered in these studies have swept wings that produce spanwise (i.e., cross) flow. Therefore, the effect of the sweep angle on the noise mechanism has to be considered. The additional cross flow components induced by the sweep angle may break down the coherent structures present along with the shear layer in the slat cove. As a result, the tonal noise components may be reduced or even completely vanish in small-scale testing [Manoha et al., 2018]. Trailing edge noise theory also predicts the effect of the sweep angle [Grasso et al., 2019] on the far-field radiated by slat noise. The power spectral density of the far-field noise...
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\( (S_{pp}) \) is predicted to be proportional to the squared cosine of the sweep angle \( \Lambda \) by

\[
S_{pp}(x, \omega) \sim \cos^2 \Lambda \sin \phi \cos^2 \left( \frac{1}{2} \theta \right) \rho_0^2 U_0^5 L b \frac{S(\omega)}{|x|^2 c_\infty}
\]

where \( \phi \) and \( \theta \) are the azimuthal and polar angle, respectively, \( \rho_0 \) the mean flow density, \( U \) the mean flow velocity, \( L \) the characteristic length scale of the turbulence, \( b \) the spanwise length of the wing, \( c_\infty \) the speed of sound, \( x \) the observer location, and \( S(\omega) \) the normalized pressure spectrum. The noise generated by the wing will decrease with the introduction of wing sweep when the characteristic length scale and spanwise extend of the wing will remain the same.

Another consequence of introducing wing sweep is that the wing’s aerodynamic properties will be affected. Firstly, only the velocity component perpendicular to the leading edge contributes to the lift generation. Therefore, the section lift coefficient will be smaller for the same free-stream velocity in the wind tunnel. Because we need to consider an equivalent lift situation for both the unswept and swept wing, the velocity perpendicular to the leading edge should be increased with the sweep angle. As a result, the mean flow velocity is increased by a factor \( 1/\cos(\Lambda) \), thereby increasing the noise level according to the velocity scaling in Eq. 5.1. The net effect should be a minor change in power spectral density levels of the far-field radiated noise. Secondly, the lift slope will be effectively reduced by the introduction of the sweep, which has to be taken into account when considering the geometric angles of attack in the wind tunnel [Manoha et al., 2018].

Several studies have focused on the effect of sweep angle on slat noise [Lockard and Choudhari, 2010; Manoha et al., 2018; Dierke et al., 2011]. Lockard & Choudhari concluded that the additional cross-flow component appeared to alter the slat cove dynamics based on URANS simulations. However, they did not observe an apparent influence of the cross-flow on the radiated noise. An experimental investigation in [Manoha et al., 2018] observed a reduction of the tonal peaks in the slat spectrum. The power spectral density of far-field noise from a swept wing was observed to be slightly increased over most of the frequency range, which contradicts the earlier theoretical basis. It was unclear if this increase lay within the uncertainty range of their measurement procedure.

This study investigates the effect of sweep angle on the aeroacoustic characterization of leading-edge slat noise. The study is conducted in a small-scale open-jet wind tunnel facility at the University of Twente with the 30P30N common high-lift research model. A comparison of the far-field radiated noise is made based on the equivalent section
5.2. Experimental Setup

5.2.1 Wind Tunnel Facility

The measurements described in this study were performed in the Aeroacoustic Wind Tunnel of the University of Twente. A description of the facility and test section can be found in Sec. 3.1.

5.2.2 Microphone Phased Array for Beamforming

Acoustic beamforming measurements were performed using a microphone phased array consisting of 64 GRAS 40PH free-field microphones (Fig. 5.6a). The system accuracy of these microphones is ± 1 dB in the 50 Hz to 5 kHz range, ± 2 dB in the 5 kHz to 20 kHz range, and ± 3 dB in the 20 kHz to 50 kHz range. The microphone arrangement was based on a Vogel spiral design derived from a study in [Sarradj, 2015] (with V = 5.0). This microphone arrangement provides a flat main-lobe to side-lobe ratio within an extensive frequency range (Fig. 5.6b). In addition, the microphone array had an aperture of 1 m resulting in the beamwidth properties shown in Fig. 5.6b. The microphone pressure was sampled using four National Instruments NI PXIe-4499 Sound and Vibration modules installed on a NI PXIe-1073 chassis. Samples were acquired for 45 seconds at a sampling frequency of 100 kHz. The microphones were calibrated using a GRAS 40AG Sound Calibrator Class I. GRAS type AM03464 wind-caps were used because of the flow deflection and severe recirculation in the anechoic chamber during the open-jet test of the high-lift model. The microphone array center was placed at a distance of 1.45 m from the centerline of the wind tunnel facing the pressure side of the 30P30N model. A wall comprising flat plate absorbers was placed next to the microphone phased array (see Fig. 5.7a) to minimize the background noise originating from the wind tunnel collector and turning vanes. The background noise level of an
empty wind tunnel compared to a typical slat noise measurement is shown in Fig. 5.3. A high SNR is observed.

**Figure 5.3:** Background noise level from the averaged single microphone spectrum of a typical 30P30N measurement.

Directivity measurements were performed using the same free-field GRAS 40PH microphones. The microphones were distributed along an arc with a radius of 1.65 m (Fig. 5.7b), covering the polar angles from $236^\circ$ to $308^\circ$. Data acquisition is performed identically to the microphone phased array measurements described above. A view of the experimental setups is shown in Fig. 5.4 and 5.5.

**Figure 5.4:** Experimental setup of the directivity measurements.  
**Figure 5.5:** Experimental setup of the microphone phased array.
5.3 Beamforming Methodology

5.3.1 Data Processing

The microphone data is processed using the in-house developed beamforming algorithms of the University of Twente called BeamUT. The spectral analysis of the microphone signals was performed by taking a constant number of FFT blocks of 8192 samples for every frequency. A Hanning window was applied to each block. The spectra were obtained by averaging the blocks over the complete (45 second) microphone signal using an overlap of 50% following the windowed overlapped segmented average procedure (WOSA). Self-induced noise was removed by diagonal removal of the CSM. No weighting functions were applied to the microphone signals for the results shown here. The individual frequency calibration curves for each microphone are taken into account in the CSM computation. These calibration curves are valid up to 30 kHz. Higher frequencies were therefore omitted. No microphone array corrections were made to account for the source directivity, atmospheric absorption, or coherence loss.

Output maps for the unswept configuration were generated on a search grid that extends from -0.3 m to 0.4 m in the (streamwise) x-direction and from -0.5 m to 0.5 m in (spanwise) y-direction. Figure 5.8a shows the search grid plane and model. For the swept 30P30N configuration, the search grid was extended in the x-direction to range from -0.3 m to 0.6 m (Fig. 5.8b). A uniform search grid spacing of 0.01 m was used. The baseline search grid is set in-plane with the wind tunnel centerline. The effect of rotating the search grid plane with the angle of attack on the beamforming results was found to be negligible (i.e., within 0.5 dB).

The iterative DAMAS algorithm [Brooks and Humphreys, 2004] was also applied to remove the sidelobe contamination from the CBF beamforming results. Identical search grid parameters were used, and the total number of iterations of the algorithm was set to 100. The FFT blocks were reduced to a size of 1024 to reduce the overall computational time. The CLEAN-SC algorithm was also applied but resulted in noisy far-field spectra and is therefore not considered.

Flow Convection and Shear Layer Correction.

Since open-jet measurements were conducted, corrections for the mean flow convection and shear layer refraction must be applied. The Amiet shear layer correction [Amiet,
1975 was used, assuming an infinitely thin shear layer. Furthermore, we assumed that the speed of sound in the jet $c_1$ and the stationary surrounding air $c_0$ are approximately the same. Both the microphone phased array, and microphone arc measurements are correcting the convective effect.

### 5.3.2 Source Power Integration

Integrated spectral levels are obtained using the source power integration technique [Brooks and Humphreys, 1999]. The regions of integration are shown in Fig. 5.8 as the red rectangle and are labeled S1. The ROIs are located on the center span and extend 0.3 m in the spanwise direction. This avoided the inclusion of sidelobes from the slat brackets and the wind tunnel sidewall junction. The streamwise length of the ROI was sufficiently large to capture the main-lobe at the lowest frequencies of interest. The integrated spectral levels were normalized to a reference observer distance of 1 m and a spanwise length of 1 m.

![Microphone arrangement](image1)

![Microphone array properties](image2)

**Figure 5.6:** The microphone phased array.
5.4. Aerodynamic Characterization

A basic aerodynamic characterization is shown by comparing the measured $C_p$ distribution with free-air CFD simulations. These simulations were performed using RANS and the Spalart-Allmaras turbulence model at a Reynolds number of 1.7 million [Choudhari and Lockard, 2015]. We limit the result shown here to the angles of attack $\alpha$ of $3^\circ$, $5.5^\circ$, and $8^\circ$ in the free-air reference. The $5.5^\circ$ free-air angle of attack is of specific interest as it is the main focus in the BANC workshop.
5.4.1 The Static Pressure Coefficient

The measured pressure coefficient distributions for the unswept ($\Lambda = 0^\circ$) and swept case ($\Lambda = 30^\circ$) are shown in Fig. 5.9 for $Re_c = 0.9 \cdot 10^6$. This condition corresponds to a velocity component of 45 m/s perpendicular to the leading edge. The results show only measurements that were performed without tripping devices.

The free-air pressure distribution is reproduced by matching the slat region and suction peak on the main element with a heuristic approach. This local similarity of the pressure distribution should result in identical slat noise mechanisms [Kröber and Koop, 2011]. We found a satisfactory agreement between the measured and free-air $C_p$ distribution from RANS-based CFD simulations. The effective angle of attack $\alpha_{e,s}$ is therefore defined here as the geometric angle for which the pressure distribution in the slat region is most similar to the free-air condition. This effective angle $\alpha_{e,s}$ differs from the standard notation of the effective angle of attack $\alpha_e$, which refers to the free-air angle of attack which has an identical total lift of the wing, i.e., equivalent lift state.
Several observations can be made about the pressure distribution comparison with the free-air condition. The suction peak value on the main element is in good agreement. Downstream of the suction peak is the separation bubble at \( x/c \approx 0.15 \), which is also in good agreement. Note that this separation bubble is due to a discontinuity in the surface which is necessary for stowing the leading-edge slat element. The pressure recovery towards the trailing edge of the main element is observed to be lower than the free-air condition. This is an obvious consequence of the flow caused by the open-jet, which also changes the effective chamber line of the airfoil [Brooks and Marcolini, 1984]. The suction peak on the flap element is slightly higher than the free-air configuration. This is attributed to the trailing-edge modification on the main element, discussed in
section 6.2.3. As a result, a larger separation area towards the trailing edge of the flap is seen compared to the reference.

A significant geometric angle is required in the open-jet to match the free-air angle of attack. The required correction of geometric angle of attack can reasonably well be estimated with the jet deflection correction model proposed in [Brooks and Marcolini, 1984]. Flap separation followed by stall over the main element was observed for geometric angles larger than approximately 24° in the unswept configuration. The Reynolds number is observed to have an obvious effect on the stall behavior as low Reynolds numbers induce earlier flow separation. It is not yet fully understood what the primary trigger of the flow separation is. Effects to consider are the solid and wake blockage, corner stall on the flap element, and wake effects of the slat brackets. The removal of the slat brackets excluded the latter possible cause. As discussed earlier, a lower geometric angle of attack is required for the swept-wing configuration due to a reduced lift slope (see [Manoha et al., 2018]). The swept-wing also postponed the flow separation behavior in our measurements because pressure distributions associated with higher free-air angles of attack were attainable compared to the unswept wing.

5.5 Aeroacoustic Characterization

5.5.1 Source Maps from Microphone Phased Array Measurements

Figure 5.10, 5.12 and 5.14 show the CBF source maps for $\alpha_{e,s} = 3^\circ$, $\alpha_{e,s} = 5.5^\circ$ and $\alpha_{e,s} = 8^\circ$ in the unswept wing configuration and Fig. 5.11, 5.13 and 5.15 show the same aerodynamic conditions for the swept wing configuration. The source maps were obtained using the CBF method. The source maps show that the slat noise in the center span is observable in the entire frequency range, except in the 1 kHz source maps. Noise from the flap cove on the main element dominates the source maps at 1 kHz (and below). For frequencies larger than 2 kHz, we observe that the slat noise is visible. Spurious sources from the slat brackets and wall junction still contaminate the source maps but are not overwhelming. Reflections from the top and bottom walls of the wind tunnel are also visible. In general, the source maps show a ‘clean’ noise emission from the slat in the center span of the model, even for high frequencies (>10 kHz). We find a good overall agreement when comparing the unswept and swept configuration noise.
5.5. Aeroacoustic Characterization

level values. Note that the color bars are scaled to the same values for each frequency to compare the unswept and swept configuration. It is difficult to compare the noise levels in some beamforming source maps visually. This is because the contribution from spurious noise sources hampers an appropriate comparison. The integrated spectra from source power integration provide the quantitative assessment.

**Figure 5.10:** Source maps from conventional beamforming with $\alpha_{e,s} = 3^\circ$ ($\alpha_g = 15.5^\circ$) and $Re_c = 0.9 \cdot 10^6$. Contour levels correspond to third-octave band SPL levels.
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Figure 5.11: Source maps from conventional beamforming with $\alpha_{e,s} = 3^\circ$ ($\alpha_g = 13^\circ$) and $Re_c = 0.9 \cdot 10^6$. Contour levels correspond to third-octave band SPL levels.

Figure 5.12: Source maps from conventional beamforming with $\alpha_{e,s} = 5.5^\circ$ ($\alpha_g = 19.5^\circ$) and $Re_c = 0.9 \cdot 10^6$. Contour levels correspond to third-octave band SPL levels.
5.5. Aeroacoustic Characterization

**Figure 5.13:** Source maps from conventional beamforming with $\alpha_{e,s} = 5.5^\circ$ ($\alpha_g = 16.5^\circ$) and $Re_c = 0.9 \cdot 10^6$. Contour levels correspond to third-octave band SPL levels.

**Figure 5.14:** Source maps from conventional beamforming with $\alpha_{e,s} = 8^\circ$ ($\alpha_g = 23^\circ$) and $Re_c = 0.9 \cdot 10^6$. Contour levels correspond to third-octave band SPL levels.
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5.5.2 Integrated Noise Levels

The Effect of Sweep Angle

The integrated slat noise levels of the swept and unswept configuration are compared in Fig. 5.16 for $\alpha_{e,s} = 3^\circ$ and $\alpha_{e,s} = 5.5^\circ$. This comparison is based on the equivalent lift configuration discussed earlier. The CBF and the DAMAS method are used to compare the results. DAMAS source maps of the unswept and swept wing at $\alpha_{e,s} = 5.5^\circ$ are shown in Appendix 5.7) for reference. The DAMAS source maps show a considerable reduction of the sidelobe contamination and a more precise identification of the slat noise source. The spectral levels from source power integration are plotted against the Strouhal number based on the slat chord length $St_s$ and the velocity perpendicular to the leading-edge. Figure 5.16 shows that the noise radiated by the swept-wing configuration is almost identical to that of the unswept configuration. The tonal noise peaks and vortex shedding hump are still present in the swept wing configuration and have similar characteristics. This suggests that these noise mechanisms are caused by strong flow mechanisms that remain unaffected by the cross-flow induced by the wing.
5.5. Aeroacoustic Characterization

sweep. Minor differences still exist. A higher noise level in the high-frequency range of the CBF spectrum is observed in the spectrum of the swept configuration compared to the unswept configuration. However, the DAMAS result shows similar levels for the unswept and swept configuration. This difference is therefore attributed to sidelobe contamination in the CBF output maps.

Mach Number Scaling

The (free-stream) Mach number scaling of the far-field spectra of the unswept and swept-wing configuration are examined in Fig.5.17. Similar Mach number scaling laws are observed in both configurations. Different parts of the spectrum are found to scale differently with the Mach number when plotted versus the Strouhal number. This is to be expected as explained in [Guo, July 2012]. The dominant part of the spectrum (\( St_s < 1.5 \)) is found to collapse best with a \( M^4 \) power law. The mid-Strouhal number range \( 1.5 < St_s < 10 \) is found to scale best with a \( M^{4.5} \) power law. Towards the high-frequency range, we see a tendency towards a \( M^5 \) scaling or even higher. For \( St_s > 25 \) a weaker Mach number scaling is observed. This could be due to the shear layer coherence loss effect, which is yet to be corrected for. Overall sound pressure levels result in a power-law scaling between \( M^4 \) and \( M^{4.5} \) which is similar to other experimental observations [Pascioni and Cattafesta, 2016; Pagani et al., 2016; Mendoza et al., 2002; Dobrzynski and Pott-Pollenske, 2001]. It should be noted that the Mach number range tested here is too small to make definitive conclusions about the Mach number scaling as Reynolds number effects can also play a role.
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Figure 5.16: Comparison of the integrated slat noise spectra with $Re_c = 0.9 \cdot 10^6$. The levels are expressed at the array center location and are normalized to an observer distance of 1 m and span length of 1 m. $M_0 = 0.14$ for the unswept configuration and $M_0 = 0.15$ for the swept configuration.

Figure 5.17: Mach number scaling of the integrated slat noise spectral level of the unswept and swept configuration ($\alpha_{e,s} = 5.5^\circ$). The levels are normalized to an observer distance of 1 m and span length of 1 m.
Angle of Attack Trend

Figure 5.19a and 5.19b show the angle of attack dependence of the far-field slat noise spectrum. Similar trends are observed from both sweep configurations and are comparable to trends observed by others. First, the low-frequency tonal peaks are observed to decrease in frequency and amplitude with an increasing angle of attack. This is associated with a reduction of the shear layer path length. These modal peak frequencies can be predicted with the model proposed in [Terracol et al., 2016]. This prediction is not made in the present study as information is required from the local flow field, which is yet to be measured. The vortex shedding hump is found to increase with the angle of attack. Based on the trailing edge thickness and local flow velocity, these peaks should collapse when plotted versus the Strouhal number. The shift towards a higher frequency is therefore associated with a larger local flow velocity as a consequence of the higher angle of attack [Pascioni and Cattafesta, 2018].
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(a) $\Lambda = 0^\circ$.

(b) $\Lambda = 30^\circ$.

**Figure 5.19:** Integrated slat noise spectral levels showing the angle of attack dependence ($Re_c = 0.9 \cdot 10^6$). The levels are expressed at the array center location and normalized to an observer distance of 1 m and airfoil span of 1 m.

### 5.6 Directivity Measurements

Microphone arc measurements provide polar slat noise directivity results of the overall sound pressure level as shown in Fig. 5.21. The microphone signals are bandwidth limited to the 0.2-10 Strouhal number range such that frequencies below the cut-off frequency of the chamber are excluded. The polar angles are corrected according to the geometric angle of attack of the wing $\alpha_g$ and inclination angle of the slat $\epsilon = \delta_s + \gamma_s$ where $\delta_s$ is the slat deflection angle ($30^\circ$) and $\gamma_s$ is the installation angle ($20^\circ$). A point dipole-like radiation pattern can be observed from the directivity measurements similar to the results shown in [Dobrzynski and Pott-Pollenske, 2001]. This point dipole radiation pattern is expected when considering that the dominant contribution to the far-field noise is from the frequencies nearby the peak frequency of the spectrum ($St_s \approx 1$). The frequencies associated with this dominant part will radiate in a compact (i.e., point) dipole pattern when they can be considered acoustically compact (i.e., $He<<1$). When considering the free-stream velocity of 45 m/s and slat chord length of 45 mm, then acoustic compactness can be assumed for frequencies below, let’s say 1000 Hz ($St_s \approx 1$). This aspect is observed when reviewing the polar directivity for different one-third octave bands of Strouhal numbers, especially in Fig. 5.22b of the swept-wing configuration. A dipole-like radiation pattern is observed for $St_s < 1$, whereas the radiation patterns become almost omnidirectional outside of this range. The unswept wing shows omnidirectional behavior starting sooner at $St_s > 0.5$. This could be due to contamination from other sound sources. Based on the beamforming results in Fig. 5.12 there could
5.6. Directivity Measurements

be contamination of noise from the main element’s cove region. It is also possible that
the low-frequency peaks (St < 1) seen in Fig. 5.17a are caused by a shear layer flapping
motion (explained in [Pascioni and Cattafesta, 2018]) which have a different radiation
pattern than the dipole axis on the slat trailing edge. A point dipole radiation pattern
is difficult to recover in small-scale wind tunnel tests considering the conditions required
for acoustic compactness. The relatively high Reynolds number needed to mimic the
full-scale application requires high flow velocities, resulting in an increase of the domi-
nant frequencies in the spectrum due to Strouhal number scaling. Acoustic compactness
may no longer be completely preserved in these types of situations.

![Figure 5.20: Geometric definition of the compact dipole axis on the slat trailing edge.](image)

![Figure 5.21: Bandwidth-limited St\(_s\)=0.2-10 directivity.](image)

![Figure 5.22: Polar slat noise directivity for different one-third octave bands of Strouhal number (Re\(_c\) = 0.9 \cdot 10^6).](image)

(a) \( \Lambda = 0^\circ \) and \( \alpha_g = 19.5^\circ \).

(b) \( \Lambda = 30^\circ \) and \( \alpha_g = 16.5^\circ \).
Chapter 5. The Sweep Angle Effect on Slat Noise Characteristics of the 30P30N High-Lift Model in an Open-Jet Wind Tunnel.

5.7 Conclusions

The effect of sweep angle on slat noise characteristics is investigated through open-jet measurements of the 30P30N high-lift common research model. The measurements are performed in an academic scale (0.9 m X 0.7 m) wind tunnel facility at the University of Twente. Aeroacoustic data is obtained of both an unswept and 30° wing swept configuration. A comparison of the noise characteristics is made based on an equivalent section lift condition.

Microphone phased array measurements with 64 GRAS 40 PH microphones were performed. In-house developed beamforming algorithms, benchmarked with the array benchmark database, allowed the localization and quantification of the slat noise. Noise treatment minimized the contamination of background noise and spurious sound sources such as the slat brackets and wall junctions. Integrated sound pressure levels based on the CBF and DAMAS beamforming algorithms show that the 30° wing sweep has a negligible effect on the far-field noise spectra. The broadband noise component, low-frequency tonal peaks, and high-frequency vortex shedding hump were found to be unaffected by the 30° wing sweep. It was found that the Mach number scaling is similar in both configurations, close to a M^4 power law. We, therefore, conclude that the unswept wing configuration can successfully mimic the swept wing configuration when considering similar section lift conditions.

Directivity measurements show a point dipole-like radiation pattern of the overall sound pressure level for both the unswept and swept configuration. However, the acoustic compactness condition, required to obtain a point dipole radiation pattern, may be challenging to obtain in small-scale wind tunnel testing. This results from the high free-stream velocities and consequently dominating higher frequencies involved.

Further efforts will examine the influence of the wind tunnel test section (hard wall and kevlar) on the slat noise characteristics. Comparisons with other dedicated wind tunnel tests of the 30P30N are also scheduled to investigate wind tunnel effects further. The data presented in this study will be made available to the database in category 7 of the BANC workshop for future benchmark purposes.
Figure 5.23: DAMAS output maps with $\alpha_{e,s} = 5.5^\circ$ ($\alpha_g = 19.5^\circ$) and $Re_c = 0.9 \cdot 10^6$. Contour levels correspond to third-octave band SPL levels. The levels are expressed at the array center location.
REFERENCES

Figure 5.24: DAMAS output maps with $\alpha_{e,s} = 5.5^\circ$ ($\alpha_g = 16.5^\circ$) and $Re_c = 0.9 \cdot 10^6$. Contour levels correspond to third-octave band SPL levels. The levels are expressed at the array center location.

References


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Chapter 6

Slat Noise Measurements in Open-Jet, Hard-Wall and Hybrid Wind Tunnel Test Sections.


Abstract

This study assesses the comparability of aeroacoustic slat noise measurement in an open-jet, a hard-wall, and a hybrid test section configuration. The 30P30N common research model was tested in the Aeroacoustic Wind Tunnel of the University of Twente at a chord-based Reynolds number up to $1 \cdot 10^6$ and free-stream Mach number 0.15. Pressure distribution measurements were used to determine the aerodynamic similar condition of the flow in the slat cove. These conditions are subsequently validated by time-averaged 2D PIV measurements. The experiments show that general aspects of the flow in the slat cove are similar when the $C_p$ distribution around the leading-edge slat element in each test section configuration is comparable. Overall, the shear layer path and reattachment point are in close agreement, showing that the aeroacoustic noise production mechanisms are comparable. Microphone phased array measurements were subsequently conducted to compare the far-field noise characteristics originating from the slat in each test section.
configuration. A framework is proposed to correct the acoustic measurements of each test section configuration to a standard free-field condition. Transmission loss corrections are required for the microphone measurements in the hard-wall and hybrid test section configurations, whereas a coherence loss model was validated and applied to correct the microphone measurements in the open-jet. In general, the measured noise spectra in each test section configuration are similar within approximately 5 dB. The largest differences are seen for small angles of attack with noise levels in the OTS showing considerably higher values than the CTS and HTS results. In addition, the behavior of the vortex shedding hump in the high frequency range is different in the open-jet measurements, which could be related to the larger noise level differences at small angles of attack.

6.1 Introduction

Aircraft noise is the main contributor to noise nuisance around airports. Various studies have shown a correlation between aircraft noise exposure and adverse health effects. Sleep deprivation, and an increased use of cardiovascular medication, are associated with exposure to aircraft noise [Franssen et al., 2004]. Moreover, aircraft noise annoyance is associated with depression, anxiety [Beutel et al., 2016] and an increased risk of hypertension [Baudin et al., December 2020; Eriksson et al., 2007]. The societal resistance against aircraft noise has therefore increased in recent years, compelling governments to impose strict noise regulations on airport traffic in populated areas.

One of the major contributors to aircraft noise in the approach condition is high-lift device noise, i.e., the noise generated by the wings [Dobrzynski and Pott-Pollenske, 2001]. This noise component is mainly generated by the leading-edge slat element and flap side-edge [Guo and Joshi, 2003]. In general, noise from the leading-edge slat noise contributes more compared to the flap side-edge [Dobrzynski and Pott-Pollenske, 2001]. Therefore, slat noise has been widely studied in the literature both numerically [Choudhari and Lockard, 2015] and experimentally [Pascioni and Cattafesta, September 2018; Murayama et al., 2018; Pagani et al., April 2017; Pott-Pollenske et al., 2003; Zhang et al., 2021; Jawahar et al., August 2020; Fleury et al., 2015].

The flow mechanisms responsible for the slat noise generation are presently well understood. A schematic representation of the flow is shown in Fig. 6.1. A free shear layer is formed at the slat cusp, which introduces turbulent structures to the flow. These structures are convected along the shear layer path and typically reattach to the slat
surface upstream of the slat’s trailing-edge. This yields strong pressure fluctuations at the reattachment point. A part of the turbulent structures flow into the re-circulation region of the slat cove, while most are accelerated towards the trailing-edge through the gap between the main element and slat element. The high-intensity turbulent structures subsequently scatter at the trailing-edge, thereby generating broadband noise. Additional noise is produced by the vortex shedding mechanism at the slat trailing-edge which yields a narrow-band hump in the high frequency range, depending on the trailing-edge thickness and Reynolds number [Makiya et al., 2010]. Moreover, narrow-band peaks in the low frequency range are associated with a feedback mechanism between the radiated acoustic waves and the amplification of coherent structures in the shear layer [Terracol et al., 2016]. The additional noise from the vortex shedding and slat cove resonance is widely considered as a consequence of low Reynolds number effects and is not present at flight Reynolds numbers [Ahlefeldt, 2013]. The overall flow topology, i.e., the shear layer path, mainly determines the overall noise generation from the slat [Guo, July 2012; Terracol et al., 2016; Henning et al., 2012] which consequently depends on the aerodynamic angle of attack and geometric slat parameters [do Amaral et al., 2017; Pott-Pollenske et al., 2003; Botero-Bolivar et al., 2020].

**Figure 6.1:** Schematic of slat noise source mechanisms.

Small-scale wind tunnel facilities are a powerful tool for extensive noise studies of high-lift devices. In general, aeroacoustic measurements of high-lift devices are preferred to be performed with a Reynolds number close to the real flight case because of the sensitivity of noise mechanisms to the Reynolds number [Ahlefeldt, 2017]. However,
such measurements are costly and complex, making them less beneficial for fast evaluation of low-noise wing configurations, noise reduction methods, or for the evaluation of noise prediction models. Moreover, computational fluid dynamics at realistic Reynolds numbers are still beyond reach due to limitations of computing power, although recent developments in the Lattice Boltzmann Method seem promising [Khorrami et al., 2014, 2018]. Therefore, small-scale measurements of high-lift devices are widely performed, e.g., for the evaluation of noise reducing concepts [Zhang et al., 2021; Jawahar et al., August 2020] or the validation of computational aeroacoustic simulations [Choudhari and Lockard, 2015] and noise prediction models [Molin et al., 2003; Perennes and Roger, 1998].

Small-scale wind tunnel tests are often chosen for the aeroacoustic noise reduction studies. However, the choice of wind tunnel test section configuration is less evident. Facility restrictions, instrumentation and specific measurement requirements typically play an important role in the choice of the test section type. The test section configurations presently available for aeroacoustic tests are the open-jet, closed (i.e., hard-walled), and hybrid test section.

In a hard-wall test section, i.e. closed test section (CTS) configuration, the flow is bounded by the solid walls, as illustrated in Fig 6.2. This arrangement is most favorable for aerodynamic measurements because it resembles the free-air condition the best. Aerodynamic boundary corrections to the free-air condition are typically small and widely available in the literature. However, the CTS is less favorable from an acoustic point of view. The solid walls reflect acoustic waves, affecting microphone measurements [Fischer and Doolan, December 2017; Sijtsma and Holthusen, 2003]. Moreover, microphones installed in the side walls of the CTS are contaminated with self-noise from the boundary layer on the walls, which severely hampers the signal-to-noise ratio in microphone array measurements [Jaeger et al., 2000].

The open-jet test section (OTS), illustrated in Fig. 6.2, does not have the acoustics limitations of a CTS because the side-walls are removed. Microphones can be placed out of the flow and an anechoic chamber encloses the open-jet test section, creating an acoustic free-field condition. Microphone measurements still need corrections to account for the propagation of the sound waves through the shear layer of the open-jet. Shear-layer refraction occurs as well as coherence loss between microphone pairs. Fortunately, the shear layer refraction [Amiet, 1978; Sarradj, 2017; Bahr et al., 2011] and coherence loss [Amiet, 1978; Biesheuvel et al., 2019; Ernst et al., 2015; Sijtsma et al., 2014] mechanisms have nowadays been widely studied and correction methodologies
have been proposed and validated. Aerodynamic corrections in the open-jet test section are typically more severe compared to the CTS, especially when the lift generated by a wind tunnel model is relatively large compared to the size of the wind tunnel test section [Brooks and Marcolini, 1984].

In recent years, the hybrid test section (HTS) has become increasingly popular for aeroacoustic research. A HTS is created by replacing the solid side-walls of the CTS with an acoustic transparent material, illustrated in Fig. 6.2. Side-walls of stretched Kevlar cloth panels are widely used for this purpose [Devenport et al., 2013]. The Kevlar cloth bounds the flow inside the test section whilst allowing sound waves to propagate out of the test section. Aerodynamic boundary corrections are more complex, but can typically be considered using simple approximations [Brown, 2016]. Nonetheless, angle of attack corrections are much smaller compared to the OTS, making the HTS more favorable in terms of aerodynamics. Moreover, acoustic corrections are simplified because the thick shear layer in the OTS is nonexistent, yielding improved microphone coherence [Bahr et al., 2021]. The self-noise of the acoustic transparent material remains an issue, limiting acoustic measurements in the high-frequency range. Recent research aims at optimizing the acoustic material used in the HTS to minimize this effect [Szoke et al., 2021].

Aeroacoustic measurements of high-lift devices performed in small and large-scale wind tunnels are already widely reported in the literature. However, little is known about the comparability of these measurements when performed in different test section configurations. This is mainly because systematic errors in aeroacoustic measurements are typically not straightforward to identify and reduce. These systematic errors arise from, e.g., the use of different wing models, microphone hardware, acoustic corrections or beamforming post-processing techniques [Spehr and Ahlefeldt, March 2019; Bahr et al., 2017; Sarradj et al., 2017; Merino-Martinez et al., March 2019]. Kroeber and Koop [Kröber and Koop, 2011] have studied the comparability of measurements of a generic 2D high-lift geometry in two different wind tunnels using an identical microphone array and high-lift model. They performed measurements in an open-jet and a closed test section configuration. Kroeber and Koop found that the differences in the integrated noise levels between the OTS and CTS are largest at low frequencies with OTS noise levels being higher. Moreover, coherence loss was found to reduce integrated noise levels by 4 dB in the OTS. A correction model for the coherence loss was not applied. Murayama et al. [Murayama et al., 2018] and Ito et al. [Ito et al., 2010] compared slat noise measurements from a 30P30N high-lift model in a HTS and CTS. They showed that acoustic
measurements were similar and highlighted the benefits of the HTS. A comparison of airframe noise measurements from a scaled aircraft model in an OTS and CTS is also given in [Oerlemans et al., 2007]. They found that absolute noise levels differences are within 3 dB for all frequencies whereas relative noise level differences agree within 1 dB. Larger differences were occasionally found at high-lift conditions, which were suggested to be due to slightly different flow conditions resulting from the boundary conditions imposed by the test section configurations. More recently, [Bahr, 2021] compared airframe noise measurements in an OTS and CTS of a scaled aircraft half-model. This study showed good agreement of noise spectra in the low to mid-frequency range. The CTS noise measurements suffered from low-frequency background noise contamination. Moreover, the open-jet suffered from coherence loss at high frequencies. A coherence loss correction technique was proposed to account for this effect.

This study aims to investigate the comparability of aeroacoustic measurements of a common 2D high-lift research model in an OTS, CTS, and HTS. Different from previous research, we use the same wind tunnel facility and wing model with all three test section configurations. The $C_p$ distribution was measured in each test section configuration to find the best angle of attack conditions for aerodynamic comparability of the slat noise mechanism. PIV measurements were performed to investigate the similarity of the flow topology in the slat cove. The far-field noise was measured at various angles of attack using the same hardware and data post-processing method. A framework is proposed to correct the microphone phased array measurements for test section dependent effects. Finally, the measurements presented in this study provide a dataset for the validation of wind tunnel correction models and contribute to the workshop on Benchmark Problems for Airframe Noise Computations (BANC) [Choudhari and Lockard, 2015] and the Hybrid Anechoic Wind-tunnel Technology Workshop (HAWT).

6.2 Experimental Facility and Model

6.2.1 Wind Tunnel Facility

The measurements described in this study were performed in the Aeroacoustic Wind Tunnel of the University of Twente. A description of the facility and test section can be found in Sec. 3.1.
The CTS measures 3.8 m x 0.9 m x 0.7 m (length x width x height) and is located directly downstream of the contraction. The side walls comprise transparent acrylic panels allowing for optical access. Acoustic measurements were performed with the microphone phased array placed behind an acoustic transparent window [van Bokhorst and Tuinstra, 2019]. The conventional flush-mounted configuration of the microphones in the CTS was not chosen because of the expected saturation of the microphones at high flow speed conditions, the proximity to the noise source, and the pressure doubling effect. Therefore, the microphones were recessed behind a 1.5 mm thick stainless steel perforated plate with an R3T5 perforation pattern, giving an open area ratio of 33%. The perforated plate was also covered with a 5 mm thick poly-ether foam layer (35 kg/m\(^3\)) to reduce self-noise generated by the grazing flow [Sijtsma and Holthusen, 1999; van Bokhorst and Tuinstra, 2019]. Melamine foam with 40 mm thickness covered the aluminum plate support of the microphones to reduce acoustic reflections. Figure 6.2c and Fig. 6.3 schematically show the implementation and design of the acoustic transparent window. Measurements of the \(C_p\) distribution of the 30P30N model showed that the acoustic window had a negligible effect on the aerodynamic behavior of the model in the CTS.

The HTS is created by replacing the side walls of the CTS with panels comprising stretched Kevlar cloth, shown schematically in Fig. 6.2b. The wall panels have an in-house developed tensioning frame. Plain weave Kevlar cloth with 0.12 mm thickness and a specific weight of 61 g/cm\(^2\) was used. The cloth was tensioned to approximately 1500 N/m in the center of the panel, measured with an STM-50 tension meter. Both the Kevlar cloth type and tension value were similar to [Devenport et al., 2013] and others [Bahr et al., 2021; Mayer et al., 2019]. Three separate Kevlar panels were used on one side of the test section, whereas a single panel was used on the other side of the test section, where the microphone array was located. Figure 6.2d and Fig. 6.2a schematically show the OTS, located downstream of the CTS. No side-walls are used in this test section, while the upper and lower walls of the test section comprise plates with a circular end for noise abatement. A sound-absorbing wall was placed between the microphone array and collector to reduce the background noise contamination from the collector. The sound-absorbing wall comprises flat plate absorbers, similar to those used for the walls surrounding the collector, shown in Fig. 6.2a. This wall was useful when the high-lift model was set at high geometric angles of attack, which resulted in a part of the flow to impinge on the collector walls, giving increased background noise generation.
6.2. Experimental Facility and Model

(A) Anechoic chamber with wind tunnel test sections. Hybrid test section (left) and open-jet test section (right).

(B) Hybrid (Kevlar-wall) test section.

(C) Hard-wall test section.

(D) Open-jet test section.

Figure 6.2: Schematic of the wind tunnel test section configurations for aeroacoustic measurements.

Figure 6.3: Schematic cross-section of the acoustic transparent window used in the CTS.
6.2.2 Microphone Phased Arrays

Sixty-four GRAS 40PH free-field microphones were used in the microphone phased arrays. The analog signals from the microphones were sampled by 4 NI PXIe-4499 Sound and Vibration Modules installed in a NI PXIe-1073 chassis. A sampling time of 45 s with a sampling rate of 102.4 kHz was used in all measurements. The sensitivity of each microphone was calibrated prior to the measurement campaigns in each test section configuration using a GRAS 40AG pistonphone. The individual microphone frequency response curves were included in the data processing. Wind caps (type GRAS AM0364) were used on the microphones during the OTS measurements. This was done because the microphones in the OTS setup were exposed to wind gusts caused by the flow re-circulation in the anechoic chamber from the deflected jet. The wind caps had no influence on the frequency response curves of the microphones within the measured frequency range.

Open-Jet and Hybrid Test Section

The microphone phased arrays used in the open-jet and hybrid test section were identical. A Vogel spiral arrangement was used, providing good mainlobe-to-sidelobe ratio in a wide frequency range [Sarradj, 2015]. The array had a diameter of 1 m and was located 1.45 m from the centerline of the wind tunnel in the OTS and 1.5 m in the HTS. The microphones were mounted to an open aluminum structure designed to minimize sound field interference [Sanders et al., 2020]. Figure 6.4a shows the position of the microphone phased array relative to the high-lift model.

Hard-Walled Test Section

A down-scaled version of the same microphone phased array arrangement was used in the CTS with a diameter of 0.3 m. This minimized the influence of the emission angle because the CTS array was closer to the high-lift model. Figure 6.4b gives a comparison of the beamwidth and mainlobe-to-sidelobe ratio for a 0.5 m x 0.5 m search grid. The slightly different performance of the microphone phased arrays requires appropriate selection of the parameters in the data post-processing of the beamforming. This will be discussed in Sec. 6.3.2.1.
6.3. Experimental Methodology

(A) Microphone array arrangement and the Region of Interest (ROI) relative to the high-lift model.

(B) Array performance for a 0.5 m x 0.5 m search grid in the experimental setups.

Figure 6.4: Microphone Phased Arrays.

6.2.3 The 30P30N High-Lift Model

A description of the 30P30N high-lift model is given in Sec. 3.2.4. The static pressure was sampled with 6 pressure scanners comprising a combination of the Model 9116 and the Model 9216 by Netscanner Systems. These pressure scanners have a system accuracy of 0.05\% relative to the full-scale. The static pressure was sampled for 10 s at 50 Hz. A Pitot-static tube was used to measure the free-stream velocity and was located approximately 2\(c_{\text{stowed}}\) upstream of the model in each test section configuration. The free-stream velocity was calibrated at each geometric angle of attack to account for the flow blockage effect. No tripping devices were applied to the model. The noise treatment, described in Sec. 3.2.4 was applied to reduce the influence of spurious noise sources.

6.3 Experimental Methodology

6.3.1 PIV Setup

Time-averaged PIV measurements of the slat cove region were performed in a 2D plane, capturing two velocity components. A description of the hardware can be found in
Sec. 3.4. Images were acquired using a LaVision Imager SX 9M camera with a 105 mm lens (SIGMA 105 mm F/2.8 DG Macro) and a green light filter. This yielded an image resolution of $\approx 60$ pixels/mm or 2700 pixels per slat chord length. Five hundred image pairs were taken for each measurement, which yielded acceptable statistical convergence of the velocity components. PIV images were acquired in two separate fields of views, namely in the slat cove (FOV1) and the slat wake (FOV2), shown schematically in Fig. 6.5. Unfortunately, no measurements were performed of FOV2 in the HTS due to set-up limitations. The images were acquired in a plane perpendicular to the model surface at $y/b = 0$ for FOV1 and $y/b = -0.25$ for FOV2. The latter was necessary because of visual obstruction from the slat brackets. The flow along the span was found to be sufficiently two-dimensional within the center-span region, so that the individual velocity fields from FOV1 and FOV2 can be overlapped. Each field of view was calibrated with a LaVision 106-10 calibration plate measuring 106 mm x 106 mm (shown in Fig. 3.42). The camera was positioned below the test section and mounted to the turntable of the airfoil model (shown in Fig. 3.42). This enabled the simultaneous rotation of the camera and the high-lift model. Consequently, the velocity components are shown in the reference frame of the model.

Several post-processing steps were performed to compute the final vector fields. (1) Image pairs were acquired using the time delays $Dt$ shown in Tab. 6.1. (2) A sub-pixel shift correction was applied to account for camera vibrations. The illuminated contour of the model served as the reference. (3) A sliding background subtraction was applied, reducing high intensity regions. (4) A geometric mask was applied to define the flow domain of interest. (5) The standard PIV operation was performed to compute the vector fields using a $32 \times 32$ window size with 75% overlap. The resulting vector resolution in each field of view is given in Tab. 6.1. (6) Vector post-processing was applied to remove vectors with a correlation peak below 0.2. Moreover, an outlier removal technique was applied. (7) The 500 image pairs were averaged and the mean quantities were calculated.

The HTS and CTS were slightly modified to allow the laser sheet to enter the test section. The three separate panels in the HTS, shown in Fig. 6.2b, were interchanged with the single large panel on the lower side of the model. The center panel was then replaced with an acrylic panel. Similarly, the acoustic window in the CTS, shown in Fig. 6.2c was replaced by an acrylic panel. Both modifications had a negligible effect on the aerodynamic behavior of the high-lift model in the test section configurations.
6.3. Experimental Methodology

Figure 6.5: Schematic of the PIV setup.

<table>
<thead>
<tr>
<th></th>
<th>Dt FOV1</th>
<th>Dt FOV2</th>
<th>PIV window</th>
<th>vector/mm FOV1</th>
<th>vector/mm FOV2</th>
<th>Lens focal FOV1</th>
<th>Lens focal FOV2</th>
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<td>7.7</td>
<td>100 mm</td>
<td>60 mm</td>
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<tr>
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<td>2.08</td>
<td>32x32 (75% overlap)</td>
<td>7.1</td>
<td>9.3</td>
<td>100 mm</td>
<td>100 mm</td>
</tr>
<tr>
<td>HTS</td>
<td>2.08</td>
<td>-</td>
<td>32x32 (75% overlap)</td>
<td>7.3</td>
<td>-</td>
<td>100 mm</td>
<td>-</td>
</tr>
</tbody>
</table>

Table 6.1: PIV parameters

6.3.2 Acoustic Beamforming

6.3.2.1 Data Post-Processing

The microphone phased array data was processed with our in-house beamforming code. Conventional beamforming in the frequency domain (CBF) was used to generate the results of this study. The cross spectral matrix (CSM) was calculated using the Welch-Overlapped Segmented Averaging (WOSA) method, where the time signal is divided into segments of 8192 samples and the Hanning window is applied to each segment with a 50% overlap. The average over all blocks was subsequently taken. Diagonal removal of the CSM was applied to remove the contribution of microphone self-noise. No microphone weighting functions were applied. The search grid extends from $x = -0.3$ m to 0.4 m and $y = -0.5$ m to 0.5 m relative to the centers of the microphone arrays. The search grid had a spatial resolution of 0.01 m. This dimension is smaller than the smallest beamwidth at the highest frequency of interest, 25 kHz. Slat noise was isolated from other sources in the beamforming maps using the Source Power Integration (SPI) technique [Brooks and Hodgson, 1981]. The region of interest (ROI) was located on the center span of the model and covers 0.3 m along the spanwise direction and 0.3 m in the chordwise direction, illustrated in Fig. 6.4a. The integrated spectral levels are
normalized to an observer distance of 1 m and a span length of 1 m. Values below 10 dB of the peak value in the beamforming maps were removed to reduce the influence of background noise and sidelobes. Moreover, the integrated noise levels in the ROI are not considered when the peak value in the beamforming maps is not on the leading-edge slat element. This calculation step was necessary at low frequencies where the noise from the flap cove was dominant over the slat. The microphone frequency response curves were applied in the spectral computation. Furthermore, it was assumed that the source directivity is negligible considering the relatively small emission angles of the microphones in the array compared to the typical directivity pattern of slat noise. Test section dependent acoustic corrections were applied according to the procedure described in the following sections.

6.3.2.2 Open-jet Test Section Corrections

In the open-jet, sound waves are convected with the flow and are refracted at the shear layer interface [Amiet, 1978]. This requires the consideration of two effects. First, the shear layer refraction of the sound waves needs to be corrected for which the shear layer refraction correction method described in [Bahr et al., 2011] was used. This method assumes a different speed of sound inside the flow compared to the quiescent air outside the jet. Second, the shear-layer induced coherence loss effect on the microphone phased array measurements was considered [Kröber and Koop, 2011; Bahr et al., 2021]. Slat noise is generated in a broad frequency range while measurements are performed at high Reynolds number, i.e. high flow velocities. Therefore, we performed coherence loss measurements to verify this effect and to validate a suitable coherence loss model. Appendix 6.6 gives a comprehensive description of the coherence loss measurements and modeling.

In summary, the coherence loss correction is applied as follows. First, the free shear layer in the OTS was characterized using hot-wire anemometry. Second, the von Karman coherence loss model was validated experimentally. Third, a beamforming simulation without coherence loss is performed with a monopole point source of unit strength in the center of the ROI, defined specifically for the high-lift model measurement, shown in Fig. 6.4a. Fourth, another beamforming simulation is performed, including the coherence loss effect using Eq. 6.1 and the parametrization of the open-jet setup, shown in Appendix 6.6. Fifth, source power integration is performed on both beamforming maps from which the reduction of the integrated source power due to the coherence loss effect
can be determined. This procedure provides a correction curve, which is then finally applied to the integrated noise levels obtained from the slat element of the high-lift model in the OTS. Alternatively, the coherence loss can also be modeled in each search grid point individually to correct the beamforming maps directly. However, this method is computationally intensive and is only beneficial if the sound source extends along the streamwise direction in the test section. Figure 6.6a shows the modeled and measured correction curves from a monopole loudspeaker source with \( M_0 = U_0/c_0 = 0.09 \) and 0.15 where \( U_0 \) is the free-stream velocity and \( c_0 \) the speed of sound. Frequencies below 8 kHz were not considered because these measurements were contaminated by background noise from the wind tunnel and from the airfoil used to mount the calibration source. Figure 6.6a shows that the modeled coherence loss effect on the integrated noise levels is in close agreement with the measurements. The modeled and measured coherence loss at \( M_0 = 0.09 \) appears quantitatively similar to the estimated coherence loss in a similarly sized open-jet wind tunnel in [Kröber and Koop, 2011]. However, these results should be carefully comparable because the coherence loss depends on shear layer thickness, microphone phased array arrangement, and microphone phased array location which are not exactly the same as in [Kröber and Koop, 2011]. For a free-stream velocity of \( M_0 = 0.15 \) we find a coherence loss up to 6 dB at 25 kHz. This free-stream condition corresponds to a slat noise measurement at the highest Reynolds number in this study, namely \( Re_c = 1,000,000 \).

### 6.3.2.3 Hybrid Test Section Corrections

In the HTS, the sound waves pass through the stretched Kevlar cloth, which leads to transmission loss. The transmission loss was therefore measured, following the procedure described in [Devenport et al., 2013]. A speaker was placed on one side of the test section and a microphone was located on the opposite side. The transmission loss was then determined with and without the Kevlar panel installed and the results are shown in Fig. 6.6b. A quadratic equation is fitted to the measured transmission loss, as proposed in [Devenport et al., 2013]. Flow-induced transmission loss was applied using the model in [Devenport et al., 2013]. Moreover, a refraction correction was applied following the procedure in [Bahr et al., 2011].
6.3.2.4 Hard-walled Test Section Corrections

The microphones in the CTS were placed behind an acoustic transparent window, thereby yielding acoustic transmission loss. The transmission loss model for a perforated plate proposed in [Phong and Papamoschou, 2013] is used and validated with measurements in Fig. 6.6b. The transmission loss from the 5 mm foam layer was negligible within the measured frequency range. A pressure field response correction, typically necessary in a CTS, was not required since the microphones were not flush mounted. Furthermore, coherence loss effects and boundary layer absorption were assumed to be negligible due to the small size of the boundary layer on the walls of the test section. De-reverberation techniques were not applied. The refraction correction from [Bahr et al., 2011] was also applied here.

![Graphs showing modeled and measured coherence loss effect and transmission loss](image)

Figure 6.6: Acoustic corrections.

6.4 Aerodynamic Results

6.4.1 Static Pressure

The aerodynamic comparability between test section configurations is first analyzed in view of the pressure distribution measured on the model. The geometric angle of attack $\alpha_g$ was varied from $8^\circ$ to $23^\circ$ in the OTS, from $0^\circ$ to $16^\circ$ in the CTS and from $0^\circ$ to $16^\circ$ in the HTS. The effective angle of attack of $3^\circ$, $5.5^\circ$ and $8^\circ$ will be specifically addressed.
to analyze the comparability. These angles of attack are commonly addressed, e.g., in the BANC workshop and therefore allow for direct comparison to experimental and numerical results from the literature. $C_p$ distribution measurements are compared to data from free-air RANS simulations with a Reynolds number of $Re_c = 1,700,000$ using the Spalart–Allmaras turbulence model [Choudhari and Lockard, 2015]. This is the common reference condition used in the literature and it is therefore also used here.

Figure 6.7 shows the measured $C_p$ distribution in the different test section configurations for $\alpha_e \approx 1.5, 3, 5.5$ and $8^\circ$. Overall, the main characteristics of the $C_p$ distribution match well for the different test section configurations. However, distinct differences can also be seen. Closest to the reference $C_p$ distribution is the CTS configuration. The pressure recovery on the upper side of the main element and suction peak on the flap element differ slightly from the reference. This is attributed to the trailing-edge modification on the main element, discussed in section 6.2.3. A comparison with $C_p$ measurements from the dataset from [Pascioni and Cattafesta, 2016] at the same Reynolds number indicated that this was not an effect related to the Reynolds number. This comparison can be found in Fig. 6.28 in Appendix 6.6. The $C_p$ distribution measured in the OTS shows the largest difference compared to the reference. Especially the $C_p$ distribution on the pressure side of the main element is notably lower. This can be attributed to the streamline curvature effect, also observed in [Kröber and Koop, 2011]. Nevertheless, a satisfactory match of the $C_p$ distribution of the slat element in the OTS can still be found, although the distribution deviates on the upper side at the highest effective angle of attack of $8^\circ$ in Fig. 6.7d. The $C_p$ measured in the HTS shows a distribution in between the OTS and CTS. The pressure recovery on the pressure side of the main element is close to that in the OTS. Note the higher $C_p$ measured on the pressure side in the HTS. It is presumed that this is caused by a streamwise pressure gradient in the HTS, induced by the porous Kevlar walls. Unfortunately, streamwise pressure gradient measurements could not be conducted in the current study to correct for this effect and investigate it further. Nevertheless, we can still find a close agreement of the $C_p$ distribution around the slat region. Subsequently, PIV measurements can be used to investigate if the comparable $C_p$ distribution still yields comparable flow topology and, consequently, comparable noise characteristics.

The comparative aerodynamic parameter used in this study is the slat lift, i.e., the $C_p$ distribution around the slat cove region. Kröber et al. [Kröber and Koop, 2011] showed that this parameter is the best comparative aerodynamic parameter when investigating slat noise. Their measurements showed that an exact match of the global $C_p$ distribution
could not be found. This result was confirmed with our $C_p$ measurements. Therefore, in this study, we have defined the effective angle of attack as the angle for which the measured $C_p$ distribution on the slat element best fits with the reference $C_p$ distribution from simulations. The angle of attack resolution of the reference $C_p$ distribution was $0.5^\circ$. The relation between the geometric and effective angle of attack is approximated to be linear in each test section configuration and was found using the least square fit of the measured $C_p$ distribution on the slat with the reference simulations. Figure 6.8 shows the resulting relation between the effective angle of attack $\alpha_e$ and the geometric angle of attack $\alpha_g$. It is found that the pressure distributions in the HTS and CTS match when $\alpha_{e,CTS} \approx \alpha_{e,HTS} - 2^\circ$. This is in close agreement with the results found in [Pascioni and Cattafesta, 2016] in a similarly sized test section. The $C_p$ distribution in the OTS requires a larger angle of attack correction due to the jet deflection. Also, the slope between $\alpha_e$ and $\alpha_g$ in the OTS is lower because the flow deflection in the open-jet depends on the total lift generated by the airfoil, which increases with angle of attack [Brooks and Marcolini, 1984]. The linear curve fit is used in this study to correct the geometric angles of attack to an effective angle of attack to investigate the slat noise comparability.
6.4. Aerodynamic Results

(a) $\alpha_e = 1.5^\circ$.
(b) $\alpha_e = 3^\circ$.
(c) $\alpha_e = 5.5^\circ$.
(d) $\alpha_e = 8^\circ$.

Figure 6.7: Comparison of the measured $C_p$ distributions ($Re_c = 1 \cdot 10^6$ and $M_0 = 0.15$) of the 30P30N with the reference condition from RANS-SA simulations ($Re_c = 1.7 \cdot 10^6$ and $M_0 = 0.17$).

Figure 6.8: Relation between the effective angle of attack $\alpha_e$ and the geometric angle of attack $\alpha_g$ in each test section configuration based on matching the $C_p$ distribution on the slat element to a reference numerical solution.

6.4.2 PIV Measurements

The local flow field in the slat cove region for a comparable $C_p$ distribution over the slat element was further investigated with PIV measurements. All flow quantities are represented in dimensionless form. The comparison will focus on general aspects of the slat cove flow, such as the shear layer trajectory and the reattachment location. In addition, the velocity magnitude, the spanwise vorticity and the two-dimensional turbulent kinetic energy along the shear layer path are examined. These metrics have shown to be most relevant to the generation of slat noise, although they cannot be directly related to slat noise using, e.g., existing slat noise prediction models [Choudhari and Lockard, 2015]. The time-averaged velocity magnitude contours are shown in Fig. 6.9 for each of the test section configurations and two angles of attack. The velocity contours show the apparent similarity in flow topology in each test section configuration. This can also be seen in the visual comparison of the spanwise vorticity ($\omega_z$) and two-dimensional turbulent kinetic energy ($\text{TKE}_{2D} = \frac{1}{2}(\langle u'u' \rangle + \langle w'w' \rangle)/U_0^2$) in Fig. 6.10 and Fig. 6.11 respectively. The spanwise vorticity and $\text{TKE}_{2D}$ show similar values along the shear layer path. Minor differences are visible. The $\text{TKE}_{2D}$ near the slat wake is slightly lower in the OTS compared to the CTS.

The comparability of the slat cove flow is further quantified. The path of the shear layer was extracted by applying a Canny edge detection algorithm to the velocity contour maps. This yielded a reliable and repeatable procedure which corresponds well with the path otherwise approximated by tracking the highest level of $\text{TKE}_{2D}$ along the shear layer, e.g. in [Pascioni and Cattafesta, September 2018]. Figure 6.12 shows the extracted shear layer paths for $\alpha_e = 3, 5.5$ and $8^\circ$. It is found that the shear layer paths are in close agreement between each test section configuration when the $C_p$ distribution of the slat cove region is similar. The shear layer path near the reattachment point is not shown because it is difficult to define in this region. Instead, we examine the reattachment location, which can be approximated using the stagnation point of the streamlines from the flow field. The distance from the slat cusp to the reattachment location is given in Fig.6.13, normalized with the slat chord length $c_s$. It is found that the reattachment location is in close agreement between the test section configurations and follows a similar trend with the effective angle of attack.
6.4. Aerodynamic Results

Figure 6.9: Time-averaged velocity magnitude contours \( \sqrt{\langle u \rangle^2 + \langle w \rangle^2 / U_0} \) obtained from the PIV measurements for (a) \( \alpha_e = 3^\circ \) and (b) \( \alpha_e = 5.5^\circ \) with \( Re_e = 1 \cdot 10^6 \) and \( M_0 = 0.15 \). OTS (left), CTS (middle), HTS (right).

\[ \alpha_e = 3^\circ. \]

\[ \alpha_e = 5.5^\circ. \]

**Figure 6.10**: Time-averaged spanwise vorticity contours \( (\omega_y c/U_0) \) obtained from the PIV measurements for (a) \( \alpha_e = 3^\circ \) and (b) \( \alpha_e = 5.5^\circ \) with Re\(_c = 1 \cdot 10^6 \) and M\(_0 = 0.15 \). OTS (left), CTS (middle), HTS (right).
6.4. Aerodynamic Results

Figure 6.11: Two-dimensional turbulent kinetic energy contour maps (TKE$_{2D} = \frac{1}{2}(\langle u'^2 \rangle + \langle w'^2 \rangle)/U_0^2$) obtained from the PIV measurements for (a) $\alpha_e = 3^\circ$ and (b) $\alpha_e = 5.5^\circ$ with Re$_e = 1 \cdot 10^6$ and M$_0 = 0.15$. OTS (left), CTS (middle), HTS (right).

Figure 6.12: Extracted shear layer path at various angle of attack (Re$_e = 1 \cdot 10^6$ and M$_0 = 0.15$).
The flow quantities along the shear layer path are compared in more detail. Line profiles are defined along the shear layer path, normal to the shear layer path curve. Figure 6.12a shows these line profiles for different location along the shear layer path $s$, normalized with the slat chord $c_s$. The distance normal to the shear layer path is defined as $d$, which is positive in the direction outwards of the cove and negative in the inwards direction. Figure 6.14, Fig.6.15 and Fig. 6.16 show the line profiles of velocity magnitude, spanwise vorticity and TKE$_{2D}$ at various positions along the shear layer path $s$. The x-axis range decreases with $s/c_s$ to accentuate the differences. Overall, the flow quantities are generally similar. The velocity magnitude outwards of the shear layer path in the OTS shows slightly higher values, up to approximately 5%. The most significant differences occur at the beginning of the shear layer path ($s/c_s = 0.1$) which is attributed to a combination of PIV measurement uncertainty and uncertainty in the location of the shear layer path. These differences also show to be more random and not systematic. Excluding the beginning of the shear layer, it is found that the spanwise vorticity and TKE$_{2D}$ are in close agreement between the test section configurations.
6.4. Aerodynamic Results

Figure 6.14: Time-averaged velocity magnitude profiles \(\sqrt{\langle u^2 \rangle + \langle w^2 \rangle / U_0}\) normal to the shear layer at different locations along the shear layer path, indicated in Fig. 6.12a \((Re_c = 1 \cdot 10^6\) and \(M_0 = 0.15)\).

Figure 6.15: Time-averaged spanwise vorticity \(\omega_y c / U_0\) normal to the shear layer at different locations along the shear layer path, indicated in Fig. 6.12a \((Re_c = 1 \cdot 10^6\) and \(M_0 = 0.15)\).
Figure 6.16: Two-dimensional turbulent kinetic energy profiles normal to the shear layer at different locations along the shear layer path
\[
\text{TKE}_{2D} = \frac{1}{2} (\langle u' u' \rangle + \langle w' w' \rangle) / U_0^2,
\]
indicated in Fig. 6.12a (Re$_c$ = $1 \cdot 10^6$ and $M_0 = 0.15$).

Figure 6.17 shows the time-averaged velocity fluctuations along the shear layer path for $\alpha_e = 3^\circ$ and $\alpha_e = 5.5^\circ$. Note the initial increase of the velocity fluctuations at the beginning of the shear layer, i.e., $s/c_s < 0.15$. This increase is mostly because of an increment of the $u'$ velocity fluctuation. However, the velocity fluctuations are seen to converge to similar values along the shear layer path. A comparison of the measured $u'$ and $v'$ velocity fluctuation is made with simulation results in [Choudhari and Khorrami, 2007] in Fig 6.17d showing good agreement and validation of the results.
6.5 Acoustic Results

This section addresses the comparability of the acoustic measurements obtained from a microphone array phased array in the different test section configurations. First, a comparison is made between the single microphone spectra. Second, the beamforming maps and integrated noise levels are compared and the differences are discussed.

6.5.1 Single Microphone Spectra

Figure 6.18 shows the 30P30N noise spectra obtained from the microphone in the center of the phased array used in each test section configuration with $\text{Re}_c = 1 \cdot 10^6$ and $M_0 = 0.15$. Figure 6.18a also shows the background noise spectra of the empty tunnel configuration. All spectra are normalized to a reference distance of 1 m assuming $\text{Re}_c = 1.7 \cdot 10^6$. 
spherical spreading. The signal-to-noise ratio is approximately 10 to 20 dB across the entire frequency range in all test section configurations. The HTS shows increased background noise levels at higher frequencies due to the self-noise of the Kevlar wall panels. Low-frequency background noise in the CTS is not present because the microphone phased array is recessed behind the acoustic transparent window. The single microphone spectra show good agreement of the noise level measured in the different test section configurations. The closest agreement is found between the CTS and HTS spectra both in terms of noise level as well as in spectral shape. The OTS spectrum shows a similar spectral shape as in the CTS and HTS in the low and mid-frequency range, but the noise levels from the OTS in these ranges are slightly higher, especially at a lower angle of attack. The largest discrepancy is seen in the high frequency range, where the OTS noise spectra differ from the CTS and HTS results. This may be due to higher background noise levels in the CTS and HTS because of the influence of the 30P30N model on the flow at the side walls of the test section.
6.5. Acoustic Results

Figure 6.18: Single microphone spectra as a function of the Strouhal number based on the slat chord from the center microphone in each phased array for various effective angles of attack. The noise levels are normalized to a reference distance of 1 m ($Re_c = 1 \cdot 10^6$ and $M_0 = 0.15$). The dashed lines (---) in (a) indicate the background noise spectra of the empty test section.

6.5.2 Beamforming Maps

Figure 6.19 and 6.20 compare the one-third octave band source maps for the different test sections at the effective angles of attack of 3° and 5.5°. All output maps are normalized to a reference distance of 1 m. We see that the leading-edge slat element is the dominant noise source in a broad frequency range. The noise treatment at the wall junction and on the slat brackets shows to be efficient, yielding a clean slat noise source in the center span region of the model. A noise source originating from the flap cove on the main element appears at 2 kHz in the HTS and CTS. It is found that the flap noise contribution at low frequencies $<2$ kHz increases with angle of attack and is present in all test section configurations, although not so clearly visible in the OTS source maps in

Fig. 6.19 and 6.20. At high frequencies (>16 kHz) we see that the vortex shedding noise originating from the leading-edge slat is still dominant in all test section configurations. However, the slat noise is less visible at high frequencies in the HTS, which is attributed to increased background noise levels originating from the self-noise of the Kevlar panels. The comparison of the beamforming maps shows that the source characteristics and visibility are similar, even though the performance of the microphone array in the CTS is slightly different (see Fig. 6.4b).

The noise level differences in the beamforming maps are further quantified using source power integration. Far-field integrated spectral noise levels within the ROI area, defined in Fig. 6.4a, are shown in Fig. 6.21 for $\alpha_e = 1.5, 3, 5.5$ and 8°. The SPL is shown as a function of the Strouhal number $St_s$ based on the slat chord. The integrated spectral levels are also normalized to a reference distance and airfoil span of 1 m. Overall, we find a good agreement between the spectral shapes. The tonal peaks are more clearly distinguishable compared to the single microphone spectra in Fig. 6.18. Most importantly, the Strouhal number of the tonal peaks in the spectrum agrees well between the different test section configurations. These tonal peaks are caused by the slat cove resonance which is sensitive to the effective angle of attack [Terracol et al., 2016]. We find that the noise spectra in the CTS and HTS are the most comparable. The noise levels measured in the OTS are seen to be notably higher compared to the other test section configurations, up to approximately 5 dB. The difference is seen to decrease with increasing angle of attack. Higher noise levels in the OTS compared to a CTS were also found in [Kröber and Koop, 2011]. In the high frequency range, the vortex shedding noise is in close agreement when comparing the CTS and HTS results. However, the OTS shows a different behavior of the vortex shedding hump. This could be correlated to the differences observed in the flow in the slat wake shown by the PIV results in Fig. 6.11. This difference in the flow of the slat wake could also be responsible for the increased noise levels in the OTS. Unfortunately, a causal link could not be determined with the current dataset. When comparing the integrated noise levels to the single microphone spectra in Fig 6.18 we see that the differences between the noise spectra are similar for each of the test sections. For reference, we also compare our data with the normalized integrated spectrum from [Pascioni and Cattafesta, 2016] for $\alpha_e = 5.5°$ with $Re_e = 1.71 \cdot 10^6$ in Fig. 6.21c. A 4.5th power law scaling correction is applied to the reference data to correct for the difference in Mach number. A good agreement is found with the noise spectra from Pascioni et al. both in terms of noise level and spectral shape.
6.5. Acoustic Results

Figure 6.19: Normalized one-third octave beamforming maps for \( \alpha_e = 3^\circ \). The output maps are normalized to a reference distance of 1 m for comparison (\( \text{Re}_{e} = 1 \cdot 10^6 \) and \( \text{M}_0 = 0.15 \)).

![Normalized one-third octave beamforming maps for $\alpha_e = 5.5^\circ$. The output maps are normalized to a reference distance of 1 m for comparison ($Re_c = 1 \cdot 10^6$ and $M_0 = 0.15$).](image)

Figure 6.20: Normalized one-third octave beamforming maps for $\alpha_e = 5.5^\circ$. The output maps are normalized to a reference distance of 1 m for comparison ($Re_c = 1 \cdot 10^6$ and $M_0 = 0.15$).

Figure 6.22 shows the differences between the far-field integrated spectra between each possible pair of test section configuration for various Reynolds numbers with $\alpha_e = 3^\circ$ and $5.5^\circ$. Note that flow velocity dependent corrections are made to the far-field noise.
spectra obtained in the OTS and HTS (see Sec. 6.3.2.3 and 6.3.2.2). Overall, we see that the level differences between the integrated spectra are consistent within the Reynolds number range. The largest differences are found in the high-frequency range \( f > 10 \text{ kHz} \), where the vortex shedding noise is dominant. This is due to the different behavior of the vortex shedding mechanism in the OTS. Larger differences are also seen in the low-frequency range \( f < 2 \text{ kHz} \). The tonal peaks caused by the slat cove resonance mechanism are dominant in this range. Therefore, minor differences in the amplitude and Strouhal number associated with these peaks can lead to larger differences when comparing the far-field noise spectra. The difference between the integrated spectra from the OTS and HTS is smooth because the microphone phased array that was used in these configurations is identical.

Figure 6.21: Power spectral density (PSD) of the integrated noise level spectra from the ROI, normalized to a reference distance of 1 m \( (Re_c = 1 \cdot 10^6 \text{ and } M_0 = 0.15) \).

6.6 Conclusions

The comparability of aeroacoustic slat noise measurements from a generic high-lift research model was investigated in an open-jet, a hard-wall and a hybrid test section configuration of a closed circuit wind tunnel facility. Systematic measurement errors, present in aeroacoustic tests, were reduced by using the same wind tunnel model, microphone phased array hardware and beamforming algorithms. Aeroacoustic comparability was obtained by matching the $C_p$ distribution on the leading-edge slat element to the $C_p$ distribution from free-flight RANS simulations. An overall match of the $C_p$ distribution in all test section configurations can not be achieved. Nevertheless, a satisfactory match of the $C_p$ distribution around the leading-edge slat element can still be found. Large angle of attack corrections were required in the OTS to match the $C_p$ distribution over the slat element to the reference $C_p$ distribution, varying from $9^\circ$ to $15^\circ$. Using a HTS...
6.6. Conclusions

greatly reduces the required angle of attack correction, showing a geometric angle of attack offset of approximately 2° compared to the CTS. However, the $C_p$ distribution in the HTS shows higher pressure values on the pressure side of the model. This is most likely due to a streamwise pressure gradient in the test section configuration, which is unaccounted for.

Time-averaged 2D PIV measurements were conducted to investigate the comparability of the flow topology in the slat cove in these similar aerodynamic conditions. The results show that general aspects of the slat cove flow are in close agreement between the OTS, CTS and HTS. The shear layer path was found to follow the same trajectory when a similar $C_p$ distribution around the leading-edge slat element was set in each test section configuration. In addition, the location of the reattachment point of the shear layer was found to be in close agreement and follows the same trend with the effective angle of attack. The velocity magnitude, spanwise vorticity and $TKE_{2D}$ along the shear layer path do not show any major differences in the slat cove flow. Notable differences were observed when comparing the $TKE_{2D}$ in the slat wake between the OTS and CTS which needs further investigation in future work.

Single microphone spectra show good comparability of the noise levels and spectral shape in the low and mid-frequency range. However, the noise levels measured in the OTS are consistently higher than measured in the CTS and HTS, especially at small effective angle of attack. These noise level differences reduce with increasing angle of attack. The single microphone spectra from the CTS and HTS are contaminated in the high-frequency range with self-noise, which prohibits a quantitative comparison of the noise spectra. Acoustic beamforming was therefore performed, showing a clean noise radiation from the leading-edge slat in a broad frequency range in all test section configurations. Noise originating from the flap cove still occasionally contaminates the source maps in the low frequency range ($<\approx2\text{ kHz}$). In addition, measurements showed that the coherence loss effect on the integrated noise spectra in the high frequency range could not be neglected. A coherence loss model was therefore validated and used to correct the OTS measurements.

The far-field noise spectra obtained in the OTS in general show higher noise levels compared to the CTS and HTS results, especially at lower angles of attack. In addition, the vortex shedding hump in the far-field noise spectra from the OTS shows different characteristics than the CTS and HTS results. This suggests that the physical flow mechanism at the slat trailing-edge in the OTS is different and may also result in the higher noise levels which were observed. Maximum noise level differences between the test section
configurations are approximately 5 dB or less. Maximum noise level differences up to approximately 5 dB are found, which become smaller with an increasing angle of attack. Further research should be conducted, focusing on the unsteady components of the flow. Such measurements should include either time-resolved PIV measurements of the slat wake or fluctuating surface pressure measurements on the upper and lower side near the trailing-edge of the slat.

Appendix

6.A Coherence Loss Modeling and Measurements

The coherence loss $\Gamma$ between microphone pairs can be modeled using the procedure described in [Biesheuvel et al., 2019; Ernst et al., 2015]

$$
\Gamma(\Delta r, x, k) = \exp \left( -\frac{\pi k^2 Z}{4} D(\Delta r, x) \right)
$$

(6.1)

where $\Delta r$ is the distance between a microphone pair, $x$ the streamwise position, $k$ the wavenumber, $Z$ the acoustic path length through the shear layer and $D(\Delta r, x)$ the structure function which describes the turbulence in the shear layer

$$
D(\Delta r, x) = 2 \left[ b_{||}(\Delta r, x) - b_{||}(0, x) \right].
$$

(6.2)

The structure function comprises the two-dimensional correlation function $b_{||}$ which characterizes the velocity perturbations in the shear layer and can be modeled using, e.g., the Gaussian

$$
b_{||}(\Delta r, x) = \frac{\sigma^2 l(x)}{2\sqrt{\pi}} \left( 1 - \frac{\Delta r^2}{l(x)^2} \right) \exp \left( -\frac{\Delta r^2}{l(x)^2} \right)
$$

(6.3)

or the von Karman model [Wilson, 1998]

$$
b_{||}(\Delta r, x) = \frac{2\sigma^2 l(x)}{\pi \Gamma(1/3)} \left( \frac{\Delta r}{2l(x)} \right)^{5/6} \left[ K_{5/6} \left( \frac{\Delta r}{l(x)} \right) - \frac{\Delta t}{2l(x)} K_{1/6} \left( \frac{\Delta r}{l(x)} \right) \right].
$$

(6.4)

These models have been found to be most consistent with measurements in the literature [Ernst et al., 2015; Biesheuvel et al., 2019]. The coherence loss model in Eq. 6.1 requires the determination and parametrization of wind tunnel parameters [Biesheuvel
et al., 2019], namely the shear layer thickness \( l(x) \), the turbulent velocity profile in the direction transverse of the jet flow \( u'_t \), and the turbulence intensity \( \sigma \), normalized with the speed of sound. These parameters were determined from hot-wire anemometry measurements. The HWA system comprises a Dantec Streamline Pro Frame with a single wire (55P11) and an x-wire (55P61) probes. The probes are traversed using a 3D traverse system.

Figure 3.14a shows the measured shear layer thickness to have a spreading rate \( \sigma_{sl} \) of 14. The thickness was determined from the location in the velocity profile where \( u = 95\% U_0 \). The parameter \( h \) in Fig. 3.14 is the wind tunnel width. The (effective) acoustic path length \( Z \) is determined from the ray paths calculated with Amiet’s shear layer refraction model and their intersection points with the effective thickness shear layer. The effective shear layer thickness was determined following the scaling procedure in [Biesheuvel et al., 2019] by using the measured turbulent velocity profile \( (w') \), as shown in Fig. 3.14b. Because coherence loss increases shear layer thickness \( l(x) \) and acoustic path length \( z \), the maximum of these values is taken when computing the coherence loss between microphone pairs [Ernst et al., 2015].

The coherence loss model described above is verified with measurements of the shear-layer induced coherence loss. Figure 6.23a schematically shows a special calibration sound source designed for this purpose. The sound source comprises a speaker (Wavecor TW013WA01) placed behind a converging (brass) duct such that a monopole point source is created, following the design in [Biesheuvel et al., 2019; Lauterbach et al., 2009]. The converging duct has an opening size of 3.1 mm and is enclosed in an aerodynamic fairing which is mounted to a small NACA-0018 airfoil profile, shown in Fig. 6.23b. The
airfoil generated trailing-edge noise which was dominant over the loudspeaker in the low-frequency range. However, because coherence loss occurs mostly at high frequencies, this did not influence the results.

For the coherence loss measurements, the microphone signals from the array were sampled at 50 kHz for 120 s. The cross-spectrum between each microphone pair was calculated using a segment size of 2048 samples. Measurements were performed with a free-stream velocity $U_0 = 10, 20, 30, 40$ and 50 m/s. Frequencies higher than 25 kHz were not considered mainly because the slat noise emitted by the 30P30N could not be distinguished from microphone phased array measurements at these frequencies. Measurements of coherence loss at higher frequencies would also require an accurate calibration of the microphone locations, using, e.g., an ‘acoustic GPS’ system [Ernst et al., 2017], which was not available.

The measured coherence loss as a function of the distance $\Delta r$ between microphone pairs with the most downstream microphone in the array is shown in Fig. 6.24 for $M_0 = 0.09$ ($U_0=30$ m/s) and 0.15 ($U_0=50$ m/s) and a frequency of 22 kHz. A good fit between the measured and modeled coherence loss is found at low flow velocities $<30$ m/s. The von Karman model is most comparable to the measured coherence loss similar to what was found in [Biesheuvel et al., 2019]. At higher flow velocities, the models seem to over-predict the coherence loss effect. This was also observed in [Biesheuvel et al., 2019] and [Ernst et al., 2015]. This can be attributed to an insufficient signal-to-noise ratio, as the background noise of the wind tunnel becomes too high compared to the noise generated by the calibration source. A more powerful speaker would solve this problem. Nonetheless, the low velocity data shows a good validation of the coherence loss model. Figure 6.25 shows the modeled and measured coherence loss for the most downstream and center microphone in the array, with $M_0=0.15$ and 22 kHz in a scatterplot. Figure 6.25c and 6.25f demonstrate the effect of having a thicker shear layer downstream and thus a more severe coherence loss downstream compared to the upstream direction.
6.A. Coherence Loss Modeling and Measurements

(A) $M_0 = 0.09 \ (U_0=30 \text{ m/s}).$

(B) $M_0 = 0.15 \ (U_0=50 \text{ m/s}).$

Figure 6.24: Measured and predicted coherence loss (Gaussian & von Karman) at 22 kHz for the most downstream microphone in the microphone array.

(A) Von Karman model for the most downstream microphone, $M_0 = 0.09.$

(B) Von Karman model for the most downstream microphone, $M_0 = 0.15.$

(C) Von Karman model for the center microphone, $M_0 = 0.15.$

(D) Measurement of the most downstream microphone, $M_0 = 0.09.$

(E) Measurement of the most downstream microphone, $M_0 = 0.15.$

(F) Measurement of the center microphone, $M_0 = 0.15.$

Figure 6.25: Modeled (top) and measured (bottom) coherence loss between microphone pairs at 22 kHz.

Figure 6.26: One-third octave band beamforming map at $f_c = 10$ kHz showing the effect of coherence loss for $M_0 = 0.15$. SPL levels are expressed at the microphone array center.

Figure 6.27: One-third octave band beamforming map at $f_c = 20$ kHz showing the effect of coherence loss for $M_0 = 0.15$. SPL levels are expressed at the microphone array center.

6.8 $C_p$ Distribution Comparison

Figure 6.28 compares the measured $C_p$ distribution with the measured $C_p$ distribution with $Re_c = 1 \cdot 10^6$ in [Pascioni and Cattafesta, 2016]. Note that this data is not given in the publication but was made available upon request.
REFERENCES

Figure 6.28: Comparison of the measured Cp distributions for different angles of attack (Re = 1 \cdot 10^6).

References


REFERENCES


REFERENCES


Chapter 7

Conclusions and Outlook

This work aims at understanding and reducing the uncertainty in wind tunnel testing of airframe noise components. The majority of this work consists of experimental work performed in a small-scale wind tunnel facility. The main conclusions are summarized here and recommendations for further research are discussed.

Conclusions

A considerable amount of work went into the experimental facility’s design, assembly, and characterization of the setups and methods. Chapter 3 gives a comprehensive description of this part of the work. A hard-wall, an open-jet, and a hybrid test section configuration were designed to accommodate the experiments performed in this thesis and state-of-the-art aeroacoustic measurements in general. Several airfoil models were instrumented to determine their aerodynamic and aeroacoustic behavior in each test section configuration. Moreover, these airfoil models accommodated the understanding of wind-tunnel-induced effects on the aeroacoustic measurements. A framework for performing and correcting microphone phased array measurements to a standard free-field condition was discussed. Finally, several beamforming algorithms were developed and successfully benchmarked with an array benchmark dataset.

In chapter 4, the comparability of trailing-edge noise measurements is investigated using a set of single-element airfoil models at low angles of attack in an open-jet, hard-wall, and hybrid test section. A DU97W300 airfoil model validated basic 2D aerodynamic corrections focusing on the $C_p$ distribution. The $C_p$ distribution and overall lift are
essential to the trailing-edge noise mechanism because the $C_p$ distribution influences how the turbulence inside the boundary layer on the airfoil develops towards the trailing edge. The DU97W300 measurements showed that the $C_l - \alpha$ curves from each test section configuration collapse generally well when the corrections are applied. The most significant offset between the effective and geometric angle of attack was found in the OTS and showed a linear increase with $C_l$ as expected. A much smaller angle of attack correction is required in the HTS, whereas the smallest angle of attack correction is needed in the CTS. Towards higher angles of attack and stall, notable differences were observed between the lift curves and the $C_p$ distributions from the different test section configurations. The $C_p$ distribution from the OTS showed the most significant change of the $C_p$ distribution at a high angle of attack w.r.t. to the reference and other test section configurations. However, the boundary interference effects could play a role that was not corrected. Aeroacoustic measurements were performed on the NACA-0012 and NACA-0018 airfoil models at $0^\circ$ angle of attack and the NACA-63018 airfoil model at an effective angle of attack between $0^\circ$ and $8^\circ$. Unsteady wall pressure measurements, taken near the trailing edge, showed a negligible difference in wall pressure statistics between each test section configuration. This important result advocates that the sound source generation in each test section should be approximately equivalent. Microphone phased array measurements were used to confirm whether the far-field noise levels were comparable. The mean difference between the integrated spectra of the absolute noise levels for all measurements was $\approx 1$-$3$ dB across the entire frequency range. A larger deviation was found at the lowest frequency (500 Hz), presumably caused by the poorer performance of the microphone array in the CTS and/or reverberation effects. The noise reduction obtained with trailing-edge serrations was also measured in each test section configuration. The mean difference between the integrated spectra of the relative noise (i.e., noise reduction) levels for all measurements was again $\approx 1$-$3$ dB across the entire frequency range. Individual measurements can still show more considerable differences, which are attributed to contamination of the beamforming maps from background noise or spurious sources unrelated to the test section configuration.

The effect of sweep angle on slat noise measurements from a multi-element high-lift model is investigated in Chapter 5. Aeroacoustic measurements were performed with the 30P30N high-lift model at $0^\circ$ and $30^\circ$ sweep angle. The noise characteristics were compared based on the same sectional aerodynamic condition, which required a higher free-stream velocity for the swept configuration to preserve the same chord-wise velocity component. The microphone phased array measurements showed that the $30^\circ$ sweep angle has a negligible effect on the far-field noise spectra and their specific characteristics.
The tonal peaks, associated with a slat cove resonance, remained unaffected. In addition, no noticeable change was observed in the high-frequency hump in the far-field spectra, related to the vortex shedding at the slat’s trailing edge. Directivity measurements also showed a similar radiation pattern, equivalent to a point dipole-like radiation pattern. We conclude that the unswept wing configuration can successfully mimic the swept wing configuration when considering similar section lift conditions.

In Chapter 6, aeroacoustic measurements were performed to investigate the comparability of slat noise measurements of a multi-element high-lift model in an open-jet, a hard-wall, and a hybrid test section. $C_p$ distribution measurements demonstrated the effect of the test section configuration on the aerodynamic behavior of the high-lift model. A significant angle of attack correction was required in the OTS. Nevertheless, it was still possible to find a good match of the $C_p$ distribution around the slat element which should result in comparable noise radiation. The angle of attack correction in the HTS was significantly smaller than in the OTS. However, the $C_p$ distribution measured in the HTS showed unique results. A higher $C_p$ was measured on the pressure side of the model. This is presumably caused by a streamwise pressure gradient induced by the porous Kevlar walls. Unfortunately, streamwise pressure gradient measurements could not be conducted to correct this effect. Time-averaged 2D PIV measurements were performed to capture the mean flow field in the slat cove region. This allowed for the extraction of global flow features, which are considered to be primarily important to the noise generation mechanism. The PIV measurements showed that the shear layer path followed the same trajectory in each test section configuration when a similar $C_p$ distribution around the leading-edge slat element was set. The location of the reattachment point of the shear layer was found to be in close agreement with the test section configurations and follows the same trend with the effective angle of attack. No significant differences were found when comparing the velocity magnitude, spanwise vorticity, and $TKE_{2D}$ along the shear layer trajectory. The PIV results suggest that notable differences appear in the slat wake. Unfortunately, this area was not of primary concern in these measurements. Additional measurements are required to investigate this further. Far-field measurements using microphone phased arrays were subsequently performed. The microphone phased arrays used in the OTS and HTS were identical, whereas the CTS required a modification of the microphone phased array setup. A framework was proposed to correct the integrated spectra from the beamforming source maps from each test section configuration to a standard free-field condition. Special attention was paid to validating and implementing a coherence loss model for the OTS measurements, which showed promising results. It was found that the coherence loss
in the OTS reduced integrated spectral levels from the microphone phased array measurement by 6 dB at a free-stream velocity of 50 m/s and 25 kHz. In general, higher noise levels were found in the OTS compared to the CTS and HTS. The noise level in the OTS was up to 5 dB higher at low angles of attack. The difference is reduced by increasing the angle of attack. Moreover, it was found that the vortex shedding hump in the high-frequency range showed different characteristics in the OTS. These results suggest that the physical flow mechanism at the slat trailing edge in the OTS is slightly different and more sensitive to the boundary conditions imposed by this test section configuration. The far-field spectra from the CTS and HTS were in close agreement even though the microphone phased array setup was different.
Chapter 7. Conclusions and Outlook

**Recommendations for Future Work**

The test section configurations and experimental setups built for this work were developed with a strict budget. Therefore, several modifications could be made to improve the capabilities of the facility further:

1. The background noise level of the wind tunnel is dominated by noise originating from the turning vanes downstream of the collector. Therefore, replacing these turning vanes with turning vanes that produce less noise would be highly beneficial.

2. The collector could be covered with a soft material, such as faux fur, to reduce noise reflections and noise generated by the impingement of the unsteady jet.

3. The hybrid test section should be improved so that both sidewalls comprise a single stretched Kevlar panel.

4. The sandwich plates on the top and bottom of the OTS could be replaced with flat plate absorbers for noise abatement.

The microphone phased array system and beamforming software should be developed further to improve the measurement accuracy:

1. An acoustic GPS system should be developed so that the error in microphone locations can be accurately determined for array calibration.

2. Dedicated wind tunnel measurements should be performed to determine the statistics of the variation of the input parameters used in beamforming. These statistics can be used to estimate the uncertainty of CBF results, e.g., by using a Monte Carlo simulation approach.

3. The validation and accuracy of the coherence loss model could be improved by using a loudspeaker that generates a higher signal-to-noise ratio.

4. Other beamforming algorithms, which are more beneficial for particular noise sources, could be implemented and benchmarked.

5. Regularization should be applied to the beamforming algorithms in order to reduce their uncertainty and to obtain a stable reconstruction of source maps, especially if the signal-to-noise ratio in a measurement is low.
The comparability of trailing-edge noise measurements in different test section configurations should be extended to include other, more complex, single-element airfoil geometries. Firstly, aeroacoustic measurements should be performed on the DU97W300 airfoil model with unsteady wall pressure instrumentation. However, this geometry represents the inner part of the rotor blade from a wind turbine. Therefore, this geometry is less relevant to studying airfoil trailing-edge noise when considering wind turbine noise in the standard operating condition. On the other hand, the DU97W300 airfoil is still helpful in analyzing the separation or stall noise mechanism in the pre-stall and stall condition. Secondly, aeroacoustic measurements of a DU97W180 geometry should be performed, more representative of the outer part of a wind turbine rotor blade. Therefore, this geometry is more suitable for studying the trailing-edge noise mechanism. Lastly, a controlled diffusion airfoil geometry should be used for aeroacoustic characterization. This type of airfoil is widely used in compressors or other cascade applications.

The aeroacoustic data presented in Chapter 4 is a valuable dataset for cross-facility benchmark efforts. The NACA-0012, NACA-0018, and NACA-63018 are common airfoil geometries widely used for benchmark purposes in the scientific community, e.g., in the Hybrid Anechoic Wind Tunnel Workshop. Especially the NACA-63018 dataset is particularly interesting because the same airfoil model was tested by TU Delft and DTU in their aeroacoustic facilities. Therefore, a cross-facility comparison of the data from this airfoil could provide a better understanding of the sources of uncertainty in aeroacoustic wind tunnel testing. A common beamforming framework or software package should be used in this case, such as the python-based ACOUNLAR software.

The high-lift model study conducted in Chapter 6 needs further investigation:

1. The 30P30N measurements in the HTS show a significant difference of the $C_p$ distribution in this test section configuration. Static pressure taps should be installed in the Kevlar wall panels to measure the pressure distribution on the side walls.

2. The different slat noise characteristics found in the open-jet test section need further study to understand the responsible physical mechanism. Time-resolved stereoscopic PIV measurements should be performed, focusing on the trailing-edge region of the slat.

3. The experimental dataset should be used to perform a cross-facility comparison of 30P30N wind tunnel measurements found in the literature. This could improve the understanding of facility-dependent wind tunnel uncertainties, especially in combination with the NACA-63018 dataset.
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