Two wing self-noise modeling methods were used to predict the effect of airfoil camber and thickness on wing self-noise and their relationship with lift. Both a semi-empirical airfoil self-noise prediction code called NAFNoise, and a noise metric which uses steady RANS models were used for the investigation. Development and previous validations of the NAFNoise code with experimental measurements are summarized to understand the limitations of the code. Predictions were made at low speed and moderate Reynolds number similar to the environment of a small unmanned aerial system. Self-noise predictions are plotted with lift coefficient for a series of camber and thickness changes. The boundary layer was tripped and left untripped to understand the effect boundary layer transition has on airfoil noise versus airfoil aerodynamic performance. Analysis indicated that increasing airfoil camber leads to higher overall sound level at lower angles of attack. An increase in airfoil camber increases lift at lower angles of attack. NAFnoise models with untripped boundary layers predict that increasing airfoil thickness leads to higher overall sound level, with largest increase in overall sound level at lower angles of attack. Results were inconclusive as to whether increasing camber is an effective way to increase lift coefficient and lift over drag without significantly increasing airfoil noise, because the conclusion was dependent on modeling methodology. NAFNoise models with untripped boundary layers indicated that increasing camber would result in a beneficial increase in section lift coefficient & lift over drag (up to about 8% camber) with minimal increase in noise production. RANS based noise metric and exclusion of the laminar boundary layer vortex shedding model from NAFNoise predictions indicated an increase in noise with section lift coefficient independent of airfoil camber.

**Nomenclature**

\[
\begin{align*}
\alpha & = \text{angle of attack} \\
\text{a} & = \text{speed of sound} \\
b & = \text{span} \\
c & = \text{chord} \\
C_d & = \text{section drag coefficient} \\
C_l & = \text{section lift coefficient} \\
\delta & = \text{boundary layer thickness} \\
\delta^* & = \text{displacement thickness} \\
H & = \text{distance to far field observer} \\
l_0 & = \text{characteristic turbulence length scale} \\
L & = \text{spanwise extend wetted by the flow}
\end{align*}
\]

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I. Introduction

High-fidelity aeroacoustic computational modeling methods that solve the Navier-Stokes equations around a wing geometry remain computationally expensive. Analysis time is too long to incorporate into a complex aircraft vehicle design optimization framework that may analyze hundreds to thousands of configurations. Rapid airfoil self-noise modeling options are under investigation for use in multidisciplinary aircraft design computational systems being developed at the U.S. Air Force Research Laboratory [1]-[3]. Here we consider vehicles operating at low speed and moderate Reynolds number similar to the environment of a small unmanned aerial system. Critical to incorporating airframe noise into a multidisciplinary vehicle design framework is an adequate understanding of the accuracy and limitations of the modeling method. An ideal modeling tool will provide accurate noise magnitude across the hearing spectrum, rather than overall sound pressure level (OASPL), so that an accurate prediction of a perceived noise level can be determined. Accurate noise spectrum data is also critical when comparing airframe noise sources with other vehicle noise generators in an automated full vehicle design framework. Self-noise codes that predict wing noise trends, but do not accurately capture the magnitude or frequency of peak noise run the risk of driving a vehicle design in the wrong direction (e.g. toward an airframe noise dominated design, rather than engine noise dominated) [1]. Nonetheless, OASPL may be adequate when comparing one component design to another, or if the noise spectrum is broadband without dominating narrowband peaks. We consider two methods of wing noise prediction, self-noise semi-emperical models packaged in the code NAFNoise [4], and the RANS noise metric methodology developed by Hosder et al. [5]. The former method yields airfoil noise predictions at the 1/3 octave, whereas the latter produces only an OASPL.

NAFNoise was developed by NREL for the design of wind turbines. NAFNoise incorporates many of the noise models developed in the NASA self-noise modeling report by Brooks, Pope, and Marcolini (BPM) [6], with some additional modeling options for several airfoil noise generation mechanisms. The BPM modeling approach represents airfoil self-noise as the combination of turbulent boundary layer trailing edge noise, separated flow noise, trailing edge bluntness noise, tip vortex formation noise, and laminar boundary layer vortex shedding noise [6]. The BPM model and code is based on experimental measurements of the NACA 0012 section profiles. While the models generally reproduce the originally experimental data, there is concern about applying the models at flow conditions outside of the original tests [7], and to non-NACA 0012 airfoils [23]. The NAFNoise code includes the option to replace critical scaling parameters (boundary layer parameters) used in the BPM model with values computationally calculated with the aerodynamic modeling program XFOIL [9].

A recently proposed noise metric developed by Hosder et al. [5] is investigated and compared with NAFNoise predictions. The Hosder method uses steady RANS CFD models to calculate a noise metric that is not the exact noise generated by the wing, but they propose it is a good relative noise indicator. A key difference between Hosder noise metric and the BPM modeling approach is the use of turbulence parameters (characteristic turbulence length scale and characteristic turbulent velocity) to scale noise rather than various boundary layer thicknesses. The Hosder method has the benefit of being relatively simple to implement, and the CFD environment allows more accurate physical representation of wing geometry, however; method development described in [5] only considered small angle of attack configurations in which turbulent trailing edge noise was expected to dominate. The Hosder method warrants further validation so that limits to applicability can be defined.
This paper gives a background on the development and validation of NAFNoise available in open literature, and a description of a new RANS based noise metric developed by Hosder et al. [5]. A summary of aeroacoustic measurements on the NACA 0012 and other airfoil geometry plotted by angle of attack, Reynolds number, and Mach number provide an understanding of what airfoil noise measurement data is available, and what additional data points would be useful for further confidence in the ability of NAFNoise and other methods to predict profiles other than the NACA 0012. Finally, a prediction of the effect of airfoil camber and thickness on airfoil self-noise is completed using the NAFNoise code and RANS based noise metric.

II. Noise Model Background

A. NAFNoise

Turbulent boundary layer and separated flow noise models follow traditional TE noise scaling methods based on Ffowcs Williams and Hall [8]. For the problem of turbulence convecting at low subsonic velocity $U_c$ above a large plate, and past the trailing edge:

$$\langle p^2 \rangle \propto \rho_b v^2 U_c^2 \left( \frac{L}{r^2} \right) D. \tag{1}$$

With the assumption that $v' \propto U_c$, and $L \propto \delta$ or $\delta'$ sound pressure level amplitude scales primarily by $M^2$ (TE bluntness, which scales by $M^2$), and a boundary layer parameter (either displacement thickness or boundary layer thickness). Spectral shape and Strouhal dependence is modeled by boundary layer parameter, Mach number, angle of attack, and Reynolds number, with relations unique to each noise component. TE bluntness noise is also strongly dependent on trailing edge thickness ratio $(h/\delta_{avg})$ and a trailing edge angle parameter. A directivity function $D$ and factor of $1/r^2$ is applied to the calculation to estimate the noise at a specified location in the farfield. An example model equation for turbulent boundary layer trailing edge noise, for the pressure and suction sides, subscript $p$ or $s$,

$$SPL_p = 10 \log \left( M^5 \frac{\delta^*}{r_e^2} D_h \right) + A \left( \frac{St_p}{St_t} \right) + (K_1 - 3) + \Delta K_1 \tag{2}$$

$$SPL_s = 10 \log \left( M^5 \frac{\delta^*}{r_e^2} D_h \right) + A \left( \frac{St_s}{St_t} \right) + (K_1 - 3) \tag{3}$$

where $St_t = f \delta^*/U$, $St_t$ approximates $St_{peak}$, which is dependent on Mach number, $A$ is a spectrum shape function representative of the 1/3-octave spectral shape of the turbulent boundary layer noise mechanism dependent only on the ratio of $St$ to peak $St$, $K_1$ is a $Re_c$ dependent empirical value, and $\Delta K_1$ is an adjustment pressure side noise level as a function of $\alpha$ and $Re_c$. $\Delta K_1$ diminishes the pressure side noise contribution as the angle increases and velocity decreases [6]. The total turbulent boundary layer trailing edge noise at zero angle of attack is the combination of both the pressure and suction side noise. At angle of attack dependent noise is labeled “separation noise” and is modeled with

$$SPL_{\alpha} = 10 \log \left( M^5 \frac{\delta^*}{r_e^2} D_h \right) + B \left( \frac{St_s}{St_t} \right) + K_2 \tag{4}$$

where function $B$ is a spectral shape function similar to $A$, whose width is a function of $Re_c$. Even though the above equation is referred to as the separation noise model, it is meant to be valid for conditions in which the airfoil has not stalled [6]. A different separation noise model equation is used at higher angles of attack. The total noise is the combination of trailing edge turbulent boundary layer noise and angle dependent noise $SPL_{\alpha}$. At conditions in which laminar boundary layer vortex shedding noise $SPL_{LBLS}$ contributes to the overall airfoil noise (i.e. untripped cases), Brooks et al. [6] found that the combined contributions of $SPL_{LBLS}$, $SPL_p$, $SPL_s$, and $SPL_{\alpha}$ are important for total noise predictions. The reader is referred to Brooks et al. [6] for complete airfoil self-noise (referred to BPM) model details.

NAFNoise includes the same noise mechanisms as Brooks et al., with the exception of tip vortex formation noise, and the addition of a turbulent inflow noise model. The original Brooks et al. model is based on boundary layer thickness parameters that were obtained by experiments on the NACA 0012, which makes predictions questionable on airfoils that have a significantly different shape than the symmetric NACA 0012. The NAFNoise code includes the option of using the original BPM airfoil self-noise model empirical boundary layer parameters, or using boundary layer thickness predictions from the 2D code XFOIL [9]. An additional turbulent boundary layer trailing edge noise modeling option referred to as TNO is also available in NAFNoise. The TNO model uses the wave-number spectrum of unsteady surface pressures to estimate far field noise. The model is based on Blake [10],
and uses the following equation to calculate the far field noise assuming that the finite thickness of the trailing edge is negligible and the diffraction is similar to that of an idealized semi-infinite flat plate [11]

\[ S(\omega) = \frac{D}{4\pi R^2} \int_0^\infty \frac{\omega}{c_0 k_1} P(k_1, 0, \omega) \, dk_1 \]

with the wave number spectrum \( P(k_1, k_3, \omega) \) a function of boundary layer quantities. XFOIL is used to calculate boundary layer parameters along with several assumptions regarding the structure of the turbulent boundary layer which allows the calculation of the wave number spectrum of surface pressure fluctuations. Full details of the model implementation are included in [11].

One can conclude from comparisons between predictive models and experimental data in [6] that the BPM models work well for predicting noise of the NACA 0012 in the Reynolds number, Mach number, and angle of attack range used to build the models. Other researchers have raised concern that the BPM noise models are not applicable outside of the empirical data’s original range [7]. Moriarty [23] raised concerns about the applicability of SPL+LBL-VS model to airfoils with camber, and Tam and Ju [12] have pointed out major differences in experimental observation of tones considered to be associated with laminar boundary layer vortex shedding, with strong potential for measurements to be contaminated by facility-related tones, and notes the difficulty in accurately measuring and modeling combined airfoil noise environments. The uncertainty associated with using models based on the BPM model to predict airfoil self-noise should certainly be addressed through additional experiments and high fidelity computational modeling of cambered airfoils beyond those that currently exist in literature.

B. Concept of Hosder Noise Metric

The broadband noise originating from the interaction of the turbulent flow with the trailing-edge (TE) of a sharp-edged body such as a wing has been one of the main research areas of aeroacoustics for decades since this is the main noise mechanism of a clean wing [13]. The TE noise of a conventional wing at high lift can be thought of as a lower bound value of the airframe noise on approach [14] and its value can also be used as a measure of merit in noise-reduction studies. Most theories on TE noise use Lighthill’s acoustic analogy [15] and show that the noise intensity varies approximately with the fifth power of the freestream velocity and is also proportional to the TE length along the span and a characteristic length scale for turbulence [13], [16]. Based on these observations, Hosder et al. [5] proposed a noise metric (NM) as a relative indicator of the clean-wing airframe noise which does not necessarily provide the magnitude of the actual noise signature but is suitable for design trade-off studies. The following gives a quick derivation of their proposed noise metric. The starting point is the far-field noise intensity per unit volume, \( I \), of acoustic TE sources which Goldstein [17] obtained by rewriting the Ffowcs Williams and Hall equation [8]:

\[ I \approx \frac{\rho}{2\pi a^2 H^2} \omega_0 U_0^4 \]

where \( \rho \) is the freestream density, \( a \) is the freestream speed of sound, \( \omega_0 \) is the characteristic source frequency, \( u_0 \) is the characteristic velocity scale for turbulence, and \( H \) is the distance to the far-field observer. Equation (6) does not contain the dependency of the noise intensity on the directivity and the trailing-edge sweep angle, \( \beta \). These dependencies can be included as follows [13]

\[ I \approx \frac{\rho}{2\pi a^2 H^2} \omega_0 U_0^4 \cos^3 \beta \frac{D(\theta, \psi)}{H^2} \]

with the directivity term, \( D \), given by [8]

\[ D(\theta, \psi) = 2 \sin^2 \left( \frac{\theta}{2} \right) \sin \psi \]

where \( \theta \) is the polar directivity angle and \( \psi \) is the azimuthal directivity angle as defined in Figure 1.
Figure 1. Directivity angles definition (from Hosder et al.[5]). Here, the TE sweep angle $\beta$ is zero.

Note that Doppler factors are not included in equation (7), because the focus of the current study is on flows with low Mach numbers where the relative velocity between the sources and the observer is small. Using the Strouhal relation for turbulent flow [18], $\frac{\alpha_{0}l_0}{u_0} \approx \text{const.}$, where $l_0$ is a characteristic length scale for the turbulence, one can rewrite equation (7):

$$I \approx \frac{\rho}{2\pi^2 a^2} u_0^5 l_0^{-1} \cos^3 \beta \frac{D(\theta, \psi)}{H^2}$$

(9)

Accounting for the spanwise variation of the characteristic velocity and length scales, the trailing-edge sweep, and the directivity angles and assuming a correlation volume per unit span at the trailing edge as $dV = l_0^2 dy$ equation (9) can be integrated over the span $b$ to obtain

$$I_{NM} = \int_0^b \frac{\rho}{2\pi^2 a^2} u_0^5 l_0 \cos^3 \beta \frac{D(\theta, \psi)}{H^2} dy$$

(10)

where $I_{NM}$ is a noise-intensity indicator that can be evaluated on the upper or the lower surface of the wing. Once again note that $I_{NM}$ is not the exact value of the noise intensity; however, it is expected to be an accurate relative noise measure. Scaling with the reference noise intensity of $10^{-12}$ W/m$^2$ leads to a noise metric for the TE noise (in decibels) on the upper or lower surface [5]

$$NM_{upper, lower} = 120 + 10\log(I_{NM,upper,lower})$$

(11)

To obtain the total noise metric NM for a wing these can be added to yield

$$NM = 10\log \left(10 \frac{NM_{upper}}{10} + 10 \frac{NM_{lower}}{10} \right)$$

(12)

Hosder et al.[5] chose the characteristic turbulent velocity at a spanwise location of the wing trailing-edge, $y$, as the maximum value of the turbulent kinetic energy ($TKE$) profile along a direction normal to the wing surface $z_n$

$$u_0(y) = \max \left(\sqrt{TKE(z_n)}\right) = \sqrt{TKE(z_{max})}$$

(13)

They also proposed that the characteristic turbulence length scale can be expressed as

$$l_0(y) = \frac{\max(\sqrt{TKE(z_n)})}{\omega(z_{max})} = \frac{u_0(y)}{\omega(z_{max})}$$

(14)
where $\omega(z_{\text{max}})$ is the turbulence dissipation rate observed at the maximum $TKE$ location. They viewed this choice as more physics-based than other suggestions in the literature such as the various boundary-layer thicknesses. The $TKE$ and the turbulence frequency $\omega$ can be easily obtained from standard turbulence model equations used in Reynolds-averaged Navier-Stokes (RANS) solvers.

III. BPM model & NAFNoise prediction code compared with experiments

A large amount of experimental data has been collected as part of airfoil self-noise research. The data summarized here is available in open literature and was measured in one of three locations: UTRC Acoustic Research Tunnel [6], NRL (Netherlands) [20], and Virginia Polytechnic University [21][7]. There are certainly differences in measurements made at different facilities due to variation in the wind tunnels, model disparity, and differences in data acquisition and analysis methods.

A. NACA0012 Experiments

Figures 2 and 3 show plots of the data from each of testing location summarizing the conditions at which data is available for the NACA0012 airfoil profile, both natural boundary layer transition, and artificially tripped.

Both Devenport et al. [21]and Oerlemans [20] compared their experimental data to Brooks et al. [6]. For valid comparisons they had to correct their results for losses, the shear layer, microphone distance to the model, model span, and other details. Results obtained by Devenport et al. for a 20-cm-chord airfoil, at angle of attack of $0^\circ$ and $5.3^\circ$ were in good agreement with those obtained by Brooks et al., particularly in the 1,000 Hz to 4,000 Hz range. However, they observed differences of up to about 6 dB below 1,000 Hz. Oerlemans [20] described the comparison as follows: "The comparison figures showed a consistent picture for the different speeds: the general character of the spectra was well reproduced (i.e., broadband spectra for the tripped cases and spectral humps or tones for a number of untripped cases, depending on angle of attack). The frequencies of these humps corresponded quite well, although differences in level occurred, probably as a result of the sensitivity of the source mechanism to small geometric details."

Figure 2. Summary of NACA 0012 (Natural Transition) Aeroacoustic Data.
B. Cambered airfoil experiments

Less experimental data was found for airfoils other than the NACA 0012. VPU recorded measurements of three section profiles that were thicker and a different shape than the NACA 0012. One profile was proprietary, with limited information about the geometry. The other two profiles are the DU96, an 18\% thick profile from Delft University, and the S831, also with 18\% maximum thickness. NLR recorded measurements of six airfoils of various camber and thickness which are shown in Figure 4a. Several of the NLR tested airfoils were used by Moriarty for comparison of noise predictions from NAFNoise [22],[11].

Figure 3. Summary of NACA 0012 (Tripped) Aeroacoustic Data

Figure 4. Cambered Airfoil Experiments with BL tripped
A series of papers by Moriarty [11][22]-[23] compared predictions from different developmental versions of NAFNoise to the experimental data taken on non-NACA 0012 airfoils at NRL [20]. Comparison of cambered airfoil experimental data and the prediction tools raised serious concern about applying some of the BPM airfoil noise prediction models to non-NACA 0012 profiles. Moriarty [23] suggested that the “physical basis of the semi-empirical LBL-VS model is not valid for higher camber airfoils”. Domination of the noise by the separated flow model was also observed as AoA increased. The model may needs refining if used with non-NACA0012 airfoils. An alternate turbulent trailing edge noise model was implemented by Moriarty [11] that was developed at TNO-TPD in the Netherlands [24]. The model uses the unsteady surface pressures to estimate the far field noise and is an attempt to model the turbulent boundary layer trailing edge noise using a more physical model than that of Brooks et al. Comparison between predictions using the TNO model compared to the original turbulent boundary layer trailing edge noise of Brooks et al., showed some improvement with the TNO model, but it still remains unclear whether the model is sensitive enough to predict differences between airfoil shapes [11].

IV. Hosder Noise-Metric and TNO Validation

A noise-metric validation was performed with the same seven test cases used in Hosder et al [5], shown in Table 1, which cover a range of speeds at different small angles of attack. It should be noted that the flow conditions in the validation cases are not broad. All angles of attack are near zero and the largest variation of parameters is in Reynolds number which varies from 0.5x10^6 to 1.5x10^6. They selected these test cases from experimental results obtained by Brooks et al. [6] for which the one-third octave sound pressure level (SPL) spectrum was measured at a point H = 1.22m away from the mid-span TE (i.e. both directivity angles \( \theta \) and \( \psi \) are 90 degrees). The main noise mechanism of all the cases used in this validation study is believed to be the TE noise generated by the scattering of turbulent pressure fluctuations over the trailing edge.

<table>
<thead>
<tr>
<th>Case</th>
<th>( \alpha ) (deg)</th>
<th>( c ) (m)</th>
<th>Mach</th>
<th>( Re ) (Million)</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>0.0</td>
<td>0.3048</td>
<td>0.208</td>
<td>1.497</td>
</tr>
<tr>
<td>2</td>
<td>0.0</td>
<td>0.3048</td>
<td>0.092</td>
<td>0.665</td>
</tr>
<tr>
<td>3</td>
<td>2.0</td>
<td>0.2286</td>
<td>0.092</td>
<td>0.499</td>
</tr>
<tr>
<td>4</td>
<td>1.5</td>
<td>0.3048</td>
<td>0.116</td>
<td>0.831</td>
</tr>
<tr>
<td>5</td>
<td>0.0</td>
<td>0.3048</td>
<td>0.162</td>
<td>1.164</td>
</tr>
<tr>
<td>6</td>
<td>2.0</td>
<td>0.2286</td>
<td>0.208</td>
<td>1.122</td>
</tr>
<tr>
<td>7</td>
<td>1.5</td>
<td>0.3048</td>
<td>0.208</td>
<td>1.497</td>
</tr>
</tbody>
</table>

NASA’s OVERFLOW [19] solver is used to produce RANS results for all these test cases. This code uses structured overset grid systems and algebraic, one-equation, and two-equation turbulence models as well as low speed preconditioners are available. It also allows the user to discretize inviscid fluxes with up to sixth order accurate schemes which helps to keep the artificial dissipation error low. The total noise metric for each case, \( NM_i \), was calculated using the procedure described above and for the same cases, the overall sound pressure levels \( OASPL_i \) were calculated from the available experimental data. Since the noise metrics are not the exact values for the overall sound pressure levels, Hosder et al. [5] scaled the noise metric for each case with the value obtained for case 1 using the following equation

\[
NM_{si} = 10^{0.1(NM_i - NM_1)}
\]

and they used the same scaling for the \( OASPL_{si} \) values as well. Figure 5 shows the comparison of \( NM_{si} \) and \( OASPL_{si} \) calculated with the Brooks et al. experimental data and TNO model coded in NAFNoise for each of the seven cases. In general, the TNO model followed the trend of the experimental data, with a small underprediction in relative noise for case 2, and case 5, and overprediction for case 6 and case 7. The RANS produced noise-metric calculation resulted in a slight overprediction for each of the test cases, with overall better agreement with experiment than the TNO model for case 6 and 7. It can be inferred that the agreement in trends between the experiment and the numerical predictions are very good which implies that the noise metric is capable of capturing the variations in the TE noise as a relative noise measure when considering the small spread in flow conditions and parameters of the test cases.
V. Camber & Thickness Study

A. NAFNoise Total Noise Prediction

NAFNoise was used to predict the effect of airfoil camber and thickness on airfoil self-noise. The environmental and modeling parameters used in the study are listed in Table 2. Two different Reynolds numbers corresponding to two different freestream velocity and Mach numbers, and angles of attack in the range used for cruising or loitering (-2° to +8°) were considered. The noise levels presented in this section were calculated using the turbulent boundary layer noise, separation noise, and laminar boundary vortex shedding noise models in NAFNoise. Artificial tripping of the boundary layers was not implemented except for the cases in which tripped boundary layers were compared to naturally transitioning boundary layers.

<table>
<thead>
<tr>
<th>Study Parameters</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>$c$</td>
<td>1 meter</td>
</tr>
<tr>
<td>$b$</td>
<td>1 meter</td>
</tr>
<tr>
<td>$V$</td>
<td>25 m/s (48.6 KTAS) &amp; 35 m/s (68 KTAS)</td>
</tr>
<tr>
<td>$Re_c$</td>
<td>$1.7 \times 10^6$ &amp; $2.4 \times 10^6$</td>
</tr>
<tr>
<td>Mach</td>
<td>0.07 &amp; 0.10</td>
</tr>
<tr>
<td>$\alpha$</td>
<td>-2° to +8°</td>
</tr>
<tr>
<td>Turbulent BL</td>
<td>TNO w/XFOIL BL parameters</td>
</tr>
<tr>
<td>Noise Model</td>
<td></td>
</tr>
</tbody>
</table>

The series of airfoils used in the airfoil camber comparison, and airfoil thickness comparisons, are shown in Figure 6 a and b, respectively. Figure 7 shows the section lift and drag coefficient predictions from XFOIL for the series of airfoil profiles used in the two studies. It is well known that an increase in camber will shift the linear portion of section lift slope towards lower angles of attack. This results in an increase in lift coefficient as camber increases, for a given angle of attack. Increasing camber also has the effect of increasing the drag coefficient at angles of attack near zero. Increasing thickness will increase $C_{d,0}$ and effect $C_{l,\text{max}}$. Thickness also effects wing structural design and therefore can impact vehicle weight.
Figure 6. Airfoil series profiles used in study: a.) camber series; b.) thickness series

Figure 7. Section lift and drag coefficient prediction from XFOIL.

1. Variation in airfoil camber

Overall sound pressure level (OASPL) are shown in Figure 8 calculated with NAFNoise predictions at 1/3 octave intervals between 10 and 20,000 Hz, at a distance of 10 m directly below the trailing edge of the wing. The total noise output from NAFNoise which includes the TBL-TE, separation, and LBL-VS noise models were used to calculate OASPL. No weighting was used in the calculation of OASPL. There is little difference in the OASPL between each cambered profile with the exception of the highest cambered airfoil considered. The NACA 8410 had the highest OASPL at low angles of attack, with all four profiles producing nearly the same noise at higher angles of attack. OASPL increased as angle of attack became positive for both speeds considered.
Figure 8. OASPL for various airfoil profiles.

Figure 9. Predicted SPL across frequency spectrum with variation in camber, BL untripped. Black lines are MIL-STD-1474 Level I & II non-detectability curves (d = 10m).

Figure 9 shows predicted SPL across the frequency spectrum for various airfoil camber and angles of attack. The predictions are made at a distance 10m directly below the trailing edge of the wing (flyover point). The two black lines correspond to MIL-STD-1474 Level I & II non-detectability limits at a distance of 100 meters, for sound measurements made at 10 meters. The MIL-STD-1474 curves are plotted to give the reader some reference to the noise level predicted, versus a noise level that can be sensed by a listener. The human ear is more sensitive to sound between 1 kHz and 10 kHz indicated by the dip in the MIL-STD-1474 curves. The noise predictions in Figure 9 show the significant increase in SPL across the entire spectrum as angle of attack increases. At the higher angles of attack, increasing camber causes a narrowband peak due to the LBL-VS noise model, to shift toward higher frequencies. This shifts the narrowband peaks toward the ear’s more sensitive frequency range.

The variation of OASPL with lift coefficient, and $C_l/C_d$, is shown in Figure 10 a & b, respectively. A significant increase in section lift with little increase in OASPL can be achieved by increasing profile camber. The plot of OASPL versus $C_l$ in Figure 10 a shows a minor increase in OASPL as lift and camber increases. The amount of
noise increase, as camber and lift increases, decreases at higher angles of attack. The predictions of Figure 10 are important to aircraft design, because for a constant wing area, lift can be increased significantly through an increase in camber with relatively little noise penalty. Figure 10 b shows that not only $C_l$, but $C_l/C_d$ can increase significantly with a moderate increase in camber. There does appear to be a limit in increasing camber to gain aerodynamic efficiency, as the exception to this is the highest camber profile analyzed (NACA 8410).

![Figure 10. Variation of OASPL with a.) section lift coefficient, b.) $C_l/C_d$ for each airfoil, Re = 2.37x10^6, M=0.1.](image)

Artificial boundary layer tripping was simulated in NAFNoise by turning off the LBL-VS model and specifying trip locations on both surfaces of the airfoil. Figure 11 shows that effect of the boundary layer state on predicted OASPL. At low angles of attack the natural transition boundary layer results in a lower OASPL for each camber level. At the highest angles of attack considered, tripping the boundary layer resulted in a lower OASPL for all camber values, with the exception of the highest camber of 8%.

![Figure 11. The effect of BL transition on OASPL.](image)

Turbulent boundary layers produce more skin friction compared to laminar boundary layers. Early boundary layer transition through artificial tripping of the boundary layer will impact vehicle performance at cruise. There are of course situations in which tripping the boundary layer is aerodynamically beneficial, and the pros and cons of artificially tripping a boundary layer would need to be considered in further detail for a specific application.
2. Variation in airfoil thickness

The effect of airfoil thickness on airfoil noise was investigated by varying the thickness of symmetric airfoils from 10% to 18%. OASPL was calculated at two different Reynolds numbers shown in Figure 13. Airfoil thickness has the largest effect on overall noise at low angles of attack. OASPL increases with thickness, with most significant increase at low angle of attack. The OASPL of the 18% thick airfoil was 5 dB higher than the 10% thick airfoil at angles of attack of -2° and 0°. The relationship between lift coefficient and OASPL is shown in Figure 14. A thinner airfoil will produce lift with less noise than a thicker airfoil.
The SPL spectrum plots of Figure 15 show that thickness does have an acoustic benefit, since increasing the thickness of an airfoil will shift the narrowband noise peaks to lower frequencies, out of the ear’s sensitive range. This is very beneficial at higher angle of attack. Combining a higher thickness with higher camber value, would allow the benefit of increased lift from higher profile camber, while shifting the narrowband noise peaks of Figure 9 out of the ear sensitive range.

Figure 14. OASPL versus section lift coefficient for various airfoil thicknesses.

Figure 15. Variation of SPL for different airfoil thickness.
B. Noise-metric Comparison with NAFNoise

A comparison of scaled noise levels predicted by NAFNoise using the Brooks et al. (SELF) TBL-TE noise model, the TNO TBL-TE noise model and the RANS based noise-metric method of Hosder et al. are shown in Figure 16. Since the RANS model did not include transition prediction and the experimental results seemed to have larger discrepancies for the tonal peaks of the untripped cases we decided to present all results here for the tripped cases. The LBL-VS noise model was not used in the calculation of noise for any of the cases. A significant increase in overall noise is predicted by both Brooks et al. (SELF) models using XFOIL as the camber is increased from 0 to 2%. Beyond 2% camber, a less dramatic increase is observed by the SELF model predictions as camber is increased. As angle of attack increases, results that include the separation noise model in addition to the TBL-TE noise model show a rapid increase in noise as angle of attack increases beyond 4 degrees as the angle dependent separation noise model dominates. With the exception of the NACA 0010, the TNO TBL-TE model predicts less noise than the SELF TBL-TE noise model. The TNO model predicts a gradual increase in noise as angle of attack increases. The predicted trends of overall noise by the RANS based noise-metric agree best with the TNO model. The RANS noise-metric predicts lower noise than the TNO model at camber values of 6% or less, and predicts higher values than the TNO model for the NACA 8410.

Non-scaled noise prediction data is plotted in Figure 17. A comparison of each of the models with the noise metric prediction is shown versus lift coefficient. Examining the RANS based noise-metric predictions (dashed line in each figure) it is obvious that the prediction method contradicts earlier trends predicted with the full NAFNoise model that included the LBL-VS noise (refer to Figure 17f). Rather than predicting a potential benefit of increasing camber to obtain higher lift, without significant additional noise penalty, the RANS noise-metric prediction indicates that noise increases with section lift coefficient independent of camber in the low angle of attack range considered here. It is clear by comparison of Figure 17 b, d, and f, that the beneficial relationship between noise, camber, and lift coefficient predicted by NAFNoise in Figure 17f is due solely to the LBL-VS model. Whether the LBL-VS model is applicable to airfoils other than the NACA 0012 on which it is based should be evaluated further. Figure 17b also indicates that both the trend and magnitude of the RANS noise-metric agree well with the TNO TBL-TE model predictions.

Figure 16. Comparison of noise prediction methods scaled using Equation 15 with OASPL at α = -2° used as a reference value.
Figure 17. Comparison of Various NAFNoise generated noise predictions (solid lines) compared with the RANS based noise-metric of Hosder et al. [5] (dashed lines).

VI. Conclusions

A survey of existing airfoil aeroacoustic measurements showed that a large amount of measurements on the NACA 0012 profile have been made at low speed over a range of Reynolds numbers, Mach numbers and angles of attack, in several different facilities. Much less data was available on airfoils with camber and thickness in order to evaluate the effect of camber and thickness and on airfoil self-noise. The airfoil self-noise prediction code called NAFNoise and a relatively new RANS based noise metric were used to predict the effect of camber and thickness on noise production compared with lift production. A series of NACA airfoils with fixed chord and span, operating at two different Reynolds numbers were used for the evaluation. Airfoil self-noise increased with airfoil thickness, with the largest increase in OASPL occurring at lower angles of attack. However, at higher angles of attack increasing airfoil thickness shifted narrowband noise peaks away from the ear’s more sensitive frequency bands. The predicted relationship between noise production, section lift coefficient, and airfoil camber was dependent on the models used. If the LBL-VS, TBL-TE, and angle dependent noise model of NAFNoise was used:

- Increasing airfoil camber led to higher OASPL at lower angles of attack, but lower OASPL at higher angles of attack. However, at higher angle of attack, acoustic noise peaks shift to more audible frequencies as camber increased, making it more likely for a listener to sense the noise produced by the airfoil.
• Increasing airfoil camber results in increased lift at a lower angle of attack, and in general, lower angle of attack leads to lower noise production. This means that increasing camber can provide increases lift with the same wing planform area, with minimal increase in noise. Our study indicated that there was an upper limit to how much camber could be increased (up to 8%) before noise also increased.

Artificially tripping the boundary layer had the effect of reducing the narrowband peaks of cambered airfoils at moderate angles off attack, thus making their noise less likely to be noticed by a listener. Excluding the LBL-VS model from NAFNoise results changed the relationship between noise production and section lift coefficient. Both the RANS based noise-metric and NAFNoise without the LBL-VS model predicted that noise increases with lift coefficient independent of camber.
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References


