Passive control of rotorcraft high-speed impulsive noise

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Abstract. A strong, normal shock wave, terminating a local supersonic area located at the tip of a helicopter blade, not only limits the aerodynamic performance, but also constitutes an origin of the High-Speed Impulsive (HSI) noise. The application of a passive control device (a shallow cavity covered by a perforated plate) just beneath the interaction region weakens the compression level, thus reducing the main source of the HSI noise. The numerical investigation based on the URANS approach and Böhnig/Doerffer (BD) transpiration law (SPARC code) confirms a large potential of the new method. Two exemplary implementations, adapted to model helicopter rotors tested at NASA Ames facility in transonic conditions: Caradonna-Tung (lifting, transonic hover) and Caradonna-Laub-Tung (non-lifting, high-speed forward flight), demonstrate the possible gains in terms of the reduction of acoustic pressure fluctuations in the near-field of the blade tip. The CFD results are validated against the experimental data obtained for the reference configurations (no control), while the analysis of the passive control arrangement is based on a purely numerical research. The normal shock wave is effectively eliminated by the wall ventilation exerting a positive impact on the generated level of the HSI noise.

1. Introduction
Impulsive noise is believed to be one of the most annoying and loud sounds that can be generated by a helicopter rotor. It is described as a series of intense, low-frequency impulses radiated in the direction of forward flight. Two main types of impulsive noise (often called “blade slap”) are attributed to helicopters: Blade Vortex Interaction (BVI) and High-Speed Impulsive (HSI) [1]. The HSI noise develops when the advancing tip Mach number $Ma_{AT}$ reaches values above 0.75, giving rise to transonic conditions in high-speed forward flight. For $Ma_{AT} > 0.9$ shock wave delocalization may arise, emitting intense thumping, harsh sounds heard for many kilometres, compromising the detection distance of an approaching helicopter and increasing the community annoyance. Delaying the appearance of HSI noise and shock wave delocalization is still one of the fundamental considerations in the design and development process of rotor systems. The target is often met by a passive tip shape alteration, relying on combined sweep, taper and thinning [2]. Such a major modification of the blade's most important part effectively damps the HSI noise. Unfortunately, the achieved benefits in the acoustic signature may be outbalanced by a decreased performance. On the contrary, the proposed type of a passive control device does not require any alteration of the outer shape (section) of the blade, assuring a beneficial effect on the radiated noise without degradation of the retreating side aerodynamic performance [3].
At the beginning of the 90’s W. Braun (University of Karlsruhe, Germany) investigated experimentally passive control of the shock wave terminating a local supersonic area [4] – a flow configuration commonly found on airfoils or helicopter rotor blades in transonic conditions. A shallow cavity covered by various perforated plates was embedded into the convex wall of the test section, just below the interaction, to permit air ventilation. It was shown for the first time that when the cavity covered by a perforated plate is sufficiently prolonged (called here “extended” passive control) a strong, normal shock wave may be interchanged with a system of oblique waves, reflecting between the wall and the edge of the supersonic region (see the experimental interferograms in figure 1, obtained at shock upstream Mach number $Ma = 1.32$). Due to the expected high loss of the aerodynamic performance this configuration has been abandoned for many years in favour of the “classical” passive control. In contrast to the fixed wing applications, the “extended” passive control may constitute a basis of a new method designed for the HSI noise reduction. Fortunately, in high-speed forward flight the advancing blade is overproducing the lift, which necessitates a very low (or even negative) pitch angle setting close to the tip (almost non-lifting conditions) in order to maintain the lateral stability of a helicopter. In this case the reduction of lift due to the effect of “extended” passive control seems not to be a relevant factor, but other questions immediately arise. What benefits are expected in terms of the acoustic pressure attenuation and what impact on the aerodynamic performance of the rotor is anticipated?

Figure 1. “Extended” passive control of shock wave at convex wall ($Ma = 1.32$) [4].

2. Mechanism of passive control of shock wave by wall perforation
A shallow cavity covered by a perforated surface, located just beneath the shock wave-boundary layer interaction region, significantly alters the flow topology due to the occurrence of the transpiration through the plate (figure 2). The depicted transformation of the shock system on the NACA 0012 profile was obtained at exemplary inflow conditions: Mach number $Ma_\infty = 0.8$, Reynolds number $Re_\infty = 9 \times 10^6$ and incidence $\alpha = 1^\circ$. The method of control is passive in nature and does not require any external supply of energy. The cavity connects zones of low and high pressure, located just upstream and downstream of the shock wave. Upstream of the shock wave pressure in the cavity is higher than in the flow-field, leading to blowing into the main stream. An opposite situation arises downstream of the shock wave where lower pressure in the cavity induces suction. The blowing is perceived by the main, supersonic stream as a ramp (or a bump) resulting in a generation of front, oblique compression which in case of the “classical” passive control intersects the normal shock wave, creating a large $\lambda$-foot structure. The suction of the boundary layer downstream of the interaction is not sufficient to counteract the negative influence of blowing. Therefore, the final effect is a balance between two opposing trends: growth of viscous losses (on account of boundary layer disturbance) and reduction of wave losses (through a replacement of a normal shock by a $\lambda$-foot). It was proven during the Euroshock project that for specific configurations the method of passive control in its “classical” form is capable of marginally improving the aerodynamic performance of an airfoil in transonic conditions [5].

Another attitude (undertaken in our research group) towards airfoil performance has to be considered in respect to the rotor blade of a helicopter. In high-speed forward flight the most negative
effect is a generation of shock wave which is responsible for the HSI noise. In contrast to the “classical” passive control it is proposed to limit the shock intensity by a prolonged cavity covered by a perforated plate (i.e. “extended” passive control in figure 2). The induced flow recirculation through the cavity leads to a substitution of the strong, normal compression by a system of oblique waves, reflecting between the surface and the edge of the supersonic region. As a result the static pressure variation in the volume above the suction side of the profile is reduced – counteracting the main source of the HSI noise (figure 3). Without any modification to the blade section (outer shape) the extent of the supersonic area is significantly restricted together with a maximum Mach number in the flow-field.

An obvious consequence of the passive ventilation is flattening of the pressure distribution at the wall, but the most relevant effect becomes apparent right above the surface of the airfoil (figure 3). At a constant distance $h = 0.2c$ an intense and sudden increase of pressure coefficient $c_p$, related to a normal shock wave (black colour), is interchanged with a much more gradual compression (red colour), almost entirely eliminating total pressure wave losses at a vertical cross-section of $x/c = 0.65$, just downstream of the interaction (both locations marked in figure 2). Unfortunately, the viscous effects are magnified due to the disturbing blowing from the upstream part of the cavity, thickening boundary layer and usually intensifying flow separation, therefore augmenting total pressure losses in the region close to the wall. For the “classical” passive control similar trends are present, but barely noticeable (blue colour). A significant drawback of the “extended” passive control is a serious limitation of the normal force coefficient $c_n$ (accompanied by a slight reduction of the drag coefficient $c_d$), making the method unsuitable for fixed-wing applications, where the production of lift is of a primary importance. Favourably, for a helicopter rotor in high-speed forward flight the most intense acoustic signal of the HSI noise is generated when the advancing tip Mach number $Ma_{AT}$ is approaching maximum. At the corresponding blade azimuthal position of $\psi = 90^\circ$ the pitch angle is often limited to approx. $\theta = 0^\circ$, creating nearly non-lifting conditions and a double shock pattern.

![Figure 2. “Classical” and “extended” passive control of shock wave, NACA 0012 airfoil (Ma$_{c}$ = 0.8, Re$_{c}$ = 9·10$^6$ and $\alpha = 1^\circ$).](image)

3. **Numerical implementation of the Bohning/Doerffer (BD) transpiration law**

One of the achievements of the Euroshock project is a development of the BD transpiration law [6]. The empirical model describes a relation between the pressure difference over the perforated plate and the induced mass flow rate. The experiments proved that the pressure in the cavity $p_c$ may be considered as constant. The determination of the pressure difference $\Delta p = p(x) - p_c$ requires the wall distribution in the main stream $p(x)$. According to the BD formulation for passive control of the shock
wave the effective Mach number in a single perforation hole $M_{a_h}$ is calculated locally based on the expression:

$$M_{a_h} = 1.2 \left( \frac{|\Delta p|}{p_0} \right)^{0.55},$$  

(1)

where $p_0$ denotes the stagnation pressure value at the inlet side of the orifice, different for blowing ($\Delta p < 0$) and suction ($\Delta p > 0$). Knowing the aerodynamic porosity of the perforated plate $p_{\text{aero}}$ (often depending on the flow direction due to low manufacturing quality of the holes), the corresponding mass flow rate is estimated. Passive character of the interaction is reflected in an instantaneous zero net mass flux through the ventilated surface area. As a result an effective transpiration velocity $V_t$ (normal to the wall) distribution is determined and applied in the numerical simulation as a boundary condition.

The present investigation has been carried out with a cell-centred, block-structured, parallel code SPARC, developed by F. Magagnato at the University of Karlsruhe [7]. It solves numerically the compressible, mass-weighted, Favre-averaged Navier-Stokes equations with several turbulence models. The shock wave-boundary layer interaction phenomenon under the influence of the transpiration is fairly predicted by a low-Reynolds closure of Spalart-Allmaras (SA) [8]. The numerical algorithm is based on a semi-discrete scheme, utilizing a finite-volume formulation for the spatial discretisation (central, 2nd order) and an implicit dual-time-stepping approach for the temporal discretisation (2nd order), supplemented with an explicit Runge-Kutta method. To damp numerical oscillations the SLIP artificial dissipation model is implemented. In order to increase the convergence rate the local time-stepping, implicit residual averaging and full-multigrid techniques are incorporated in the solver.

A new, perforated wall boundary condition, designed towards the modelling of passive control of shock wave, has been implemented into the SPARC code. The validation process, based on a comparison with the experimental data obtained for a range of flow configurations: a transonic nozzle with a flat wall (Onera), curved duct with local supersonic area (University of Karlsruhe) and NACA 0012 profile (NASA Langley), has verified the extended functionality of the tool [3, 9]. In case of a classical no-slip wall boundary condition the tangential and normal velocity components (mean and fluctuating), as well as the eddy viscosity, are set to zero at the surface as a consequence of the adhesion of fluid. In an adiabatic flow (no heat-flux) pressure and density are extrapolated from the interior of the computational domain. A construction of the ventilated wall boundary condition is analogous, taking into account a non-zero normal (transpiration) velocity $V_t$, estimated from the BD law in accordance with the local properties, i.e. pressure $p(x)$ and temperature $T(x)$. The pressure in the cavity volume $p_c$ is assumed uniform, adapting itself naturally to ensure vanishing of the total mass flow rate through the perforated plate (passive environment). The temperature in the control system $T_c$.

Figure 3. Pressure coefficient $c_p$ and total pressure $p_0/\rho_0$ variation above the surface, NACA 0012 airfoil ($Ma_{\infty} = 0.8, Re_{\infty} = 9\cdot10^6$ and $\alpha = 1^\circ$).
is fixed to the stagnation value of the free-stream. If $V_t = 0$, then a standard no-slip wall boundary condition is restored, allowing for modelling of the perforated and solid surfaces in a common framework.

For each section of a rotating helicopter blade the stagnation pressure $p_0$ and temperature $T_0$ of the incoming air depend on the radial location $r/R$. In such conditions the application of a passive control system would induce an undesirable spanwise flow inside the cavity towards the tip. To counteract this effect it is proposed to divide the cavity volume into smaller, independent and sealed sub-domains, experiencing a quasi-2d, chordwise recirculation (figure 4). An additional adaptation process of pressure within each sub-cavity $p_c$ separately assures passive ventilation. Similarly, the temperature inside every sub-cavity $T_c$ has to be computed as an average value of the temperature of the sucked-in air. Frequently, the mesh density is increasing in the direction of the tip, the area subjected to large flow gradients and shocks. Hence, the resulting radial arrangement of sub-cavities is uneven with decreasing spanwise extent as the position is shifted towards the outer sections of the blade. The estimated transpiration intensity, based on the BD law, is relying on accurate prediction of the shock location and wall pressure distribution in the interaction area. As a consequence, a suitable correlation of the numerical results with the measurements for the reference rotors (no control) is of an utmost importance.

![Figure 4. Passive cavity arrangement at the tip of a helicopter rotor blade.](image)

4. Caradonna-Tung (1981) model helicopter rotor in lifting, transonic hover

To present the effectiveness of the proposed HSI noise reducing method the Caradonna-Tung (C-T) rotor was chosen, operating in severe high-speed, transonic hover conditions (tip Mach number $Ma_T = 0.877$, tip Reynolds number $Re_T = 3.931 \times 10^6$ and collective $\theta_0 = 8^\circ$) [10]. The model consists of 2 rectangular, untwisted and untapered NACA 0012 rigid blades, mounted on a tall column (figure 5). The rotor aspect ratio $AR = 6$, chord length $c = 0.1905$ m and diameter $2R = 2.286$ m. The pressure coefficient $c_p$ distributions at 5 cross-sections were measured for the reference case (no control).
The computational domain was subdivided into 80 blocks (figure 6). The flow-field of a hovering rotor is quasi-steady in respect to the blade and periodic in nature – only a single blade needs to be considered. The base computational grid ($5.7 \times 10^6$ of volumes) was refined locally within two most important mesh blocks (marked by red colour in figure 6), containing the region of transonic flow and shock wave-boundary layer interaction. It increased the mesh size up to $7.0 \times 10^6$ of volumes. The cavity covered with a perforated plate was placed at the outer 20% of the span, on the suction side, between $x/c = 0.1$ and $x/c = 0.65$ (within block 2). The aerodynamic porosity value was taken as $p_{aero} = 5\%$. The computational domain was surrounded by boundary conditions of 4 types: viscous and inviscid walls, rotational periodicity and pressure inlet/outlet. The unsteady simulation (necessary for the acoustic analysis) was progressed with a time step $\Delta t = 0.25^\circ$ of azimuth (1440 time steps per period of rotation).

A fully turbulent unsteady simulation of the hovering rotor flow-field requires 40 revolutions to develop the complex wake, tip vortices and their interactions to reach a quasi-steady state. The experimental thrust coefficient $c_t = 0.00473$ is overpredicted by 20% (inviscid model, $c_t = 0.00565$) and 10% (turbulent SA, $c_t = 0.00522$). The rotor wake (vertical inflow) affects mainly the inner sections of the blade where only low velocities are induced by the rotation. At the outer sections rotational velocity dominates. At $r/R = 0.5$ and 0.68 flow is subsonic (see comparison of $c_p$ distributions in figure 7). The shock system is building up at the outer 20% of the span ($r/R = 0.8, 0.89$ and 0.96). The agreement with the test data is satisfactory at all cross-sections, validating the computational model.
The acoustic analysis is based on a large set of data files (containing 3d flow-field) recorded during 1 rotor revolution, at every time step (1440 samples per period T) and averaged in time in the near-field of the rotor (figure 8). The overall sound pressure level OASPL at point p, located near the blade tip, is reduced by 2 dB. The corresponding acoustic pressure fluctuation $p'_{\text{ref}}$ is indicating that not only the root mean square $p'_{\text{rms}}$ is damped by 21%, but also the amplitude of the negative peak $p'_{\text{min}}$ is decreased by 27%. Moreover, the duration time of the signal is extended, limiting the impulsive character of the fluctuation. At some frequency ($f$) bands of the A-weighted sound pressure level spectrum $SPL$ the drop of more than 10 dBA is predicted ($\Delta OASPL = 4.0$ dBA). The cost of passive control is a decrease of thrust coefficient $c_T$ by 5% and an increase of torque coefficient $c_Q$ by 9%.

5. Caradonna-Laub-Tung (1984) model helicopter rotor in non-lifting, high-speed forward flight
To confirm the potential of the new HSI noise attenuating method the Caradonna-Laub-Tung (C-L-T) teetering rotor was chosen, operating in severe high-speed, non-lifting forward flight conditions (tip Mach number $Ma_T = 0.8$, tip Reynolds number $Re_T = 2.89 \times 10^6$ and advance ratio $\mu = 0.2$) [11]. The model consists of 2 rectangular, untwisted and untapered NACA 0012 rigid blades, mounted on a tall column. The rotor aspect ratio $AR = 7$, chord length $c = 0.1524$ m and diameter $2R = 2.134$ m. The instantaneous pressure coefficient $c_p$ distributions at the cross-section of $r/R = 0.89$ and azimuthal positions of $\psi = 30^\circ, 60^\circ, 90^\circ, 120^\circ, 150^\circ$ and $180^\circ$ were recorded for the reference case (no control).

The computational domain was subdivided into 160 blocks. The base computational grid ($12.2 \times 10^6$ of volumes) was refined locally within eight most important mesh blocks, containing the region...
of transonic flow and shock wave-boundary layer interaction (located above and below the surface of the tip of both rotor blades, that is in high Mach number area). It increased the mesh size up to 18.2 \times 10^6 of volumes. Two identical cavities covered with perforated plates were placed at the outer 20% of the span, on upper and lower sides, between \( x/c = 0.3 \) and \( x/c = 0.9 \). The aerodynamic porosity value was higher, i.e. \( p_{aero} = 10\% \). The transpiration was activated only in the range of \( \psi \) between 60\(^\circ\) and 150\(^\circ\). The computational domain was surrounded by boundary conditions of 4 types: viscous and inviscid walls, rotational periodicity and pressure inlet/outlet. The forward flight state was realised through grid rotation and translation. The unsteady simulation was progressed with a time step \( \Delta t = 0.25^\circ \) of azimuth (1440 time steps per period). The unsteady shock wave development is captured by the numerical method (inviscid Euler and turbulent SA), based on the comparison of \( c_p \) distributions of \( \psi \) between 60\(^\circ\) and 150\(^\circ\). The agreement with the test data is satisfactory at all azimuths, validating the computational model.

The acoustic averaging of the data is similar to the process described for the C-T rotor in hover. The overall sound pressure level \( OASPL \) at point \( p_2 \), located at \( r/R = 0.89 \), \( y/c = 0.3 \) and \( \psi = 129^\circ \), is reduced by almost 2 dB (figure 10). The corresponding acoustic pressure fluctuation \( p'/p_{ref} \) is indicating that not only the root mean square \( p'_{rms} \) is damped by 20\%, but also the amplitude of the negative peak \( p'_{min} \) is decreased by 18\% (figure 11). Moreover, the duration time of the signal is prolonged, limiting the impulsive character of the fluctuation. At some frequency bands of the A-weighted sound pressure level \( SPL \) spectrum the drop of more than 10 dBA is predicted (\( \Delta OASPL = 4.1 \) dBA). The cost of the application of passive control method is a substantial increase of torque coefficient \( c_q \) by 22\%. Since the rotor operates in non-lifting conditions (symmetric flow) and two identical cavities (covered by perforated plates) are implemented on both sides of the blade tip, the thrust coefficient \( c_t \) is still equal 0.

6. Conclusions
The article describes the application of a new method of HSI noise reduction, designed for rotorcrafts, based on the application of perforated plates and passive wall ventilation. It is evident that the normal shock wave has been eliminated for presented C-T and C-L-T rotors in transonic hover and forward flight conditions, causing a positive impact on the generated level of the acoustic pressure fluctuations in the near-filed of the blade. Unfortunately, the cost of the operation cannot be neglected, requiring further investigations and optimisation of the device focused on the aerodynamic performance of the
rotor. Another option is to develop and incorporate a cavity control system, allowing for an instantaneous activation and deactivation of the transpiration depending on the actual flight conditions.

Figure 10. Effect of “extended” passive control on overall sound pressure level OASPL.

Figure 11. Effect of “extended” passive control on acoustic pressure fluctuation $p'/p_{ref}$ and A-weighted sound pressure level spectrum SPL.

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