TIP CLEARANCE INFLUENCE ON AERODYNAMIC DAMPING MAPS

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ABSTRACT

This paper aims to investigate the influence of tip clearance on aerodynamic damping maps of an oscillating low pressure turbine (LPT) unshrouded blade row. Full-scale time-marching RANS CFD simulations in ANSYS CFX® v.16.2 employing models with 0%, 1%, 2% and 3% of span tip clearance are adopted.

Tip clearance flow leads to a significant influence on the aerodynamic damping for blade torsion mode shape and least stable IBPA = 45°. No trend in local work coefficient are observed on the suction side, although increasing tip gap leads to a stabilizing trend on the pressure side.

Aerodynamic damping maps are constructed for each tip gap. Torsion axes on the forward portion of the airfoil are found to be more stable than its aft counterparts. Bending dominated mode shapes are more stable in the cascade tangential direction. Finally, tip clearance leads to small changes in flutter characteristics for a few pitching axis locations.

KEYWORDS

TIP CLEARANCE, FLUTTER, AERODYNAMIC DAMPING MAP

NOMENCLATURE

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<th>Symbol</th>
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<tr>
<td>b</td>
<td>blade span</td>
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<td>c</td>
<td>blade chord</td>
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<td>cₚ</td>
<td>static pressure coefficient</td>
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<td>h</td>
<td>blade maximum displacement</td>
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<td>k*</td>
<td>reduced frequency</td>
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<td>M</td>
<td>Mach number</td>
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<td>N</td>
<td>number of blades</td>
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<td>n</td>
<td>normal vector</td>
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<td>p</td>
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<td>T</td>
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<td>velocity vector</td>
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<td>w</td>
<td>local work coefficient</td>
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<td>W</td>
<td>work per cycle</td>
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<td>α</td>
<td>torsion amplitude</td>
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<td>ξ</td>
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<td>ω</td>
<td>blade’s oscillation frequency</td>
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<td>Ξ</td>
<td>aerodynamic damping</td>
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Abbreviations

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<tr>
<td>AETR</td>
<td>Aeroelastic Turbine Rig</td>
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<td>CFD</td>
<td>computational fluid dynamics</td>
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<td>IBPA</td>
<td>interblade phase angle</td>
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<td>LPT</td>
<td>low pressure turbine</td>
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<td>PS</td>
<td>pressure side</td>
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<td>RANS</td>
<td>Reynolds Averaged Navier Stokes</td>
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<td>RMS</td>
<td>root mean square</td>
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<td>SS</td>
<td>suction side</td>
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<td>TBR-FT</td>
<td>Transient Blade Row Fourier Transformation Method</td>
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<td>TD</td>
<td>tie-dye</td>
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<td>TWM</td>
<td>traveling wave mode</td>
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<td>URANS</td>
<td>Unsteady Reynolds Averaged Navier Stokes</td>
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Subscripts

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<td>0</td>
<td>total quantity</td>
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<td>quantity at the inlet</td>
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<tr>
<td>2</td>
<td>quantity at the outlet</td>
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<td>aero</td>
<td>aerodynamic</td>
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<tr>
<td>imag</td>
<td>imaginary part</td>
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INTRODUCTION

Aerodynamic loading in turbomachinery blades has continuously increased with the development of modern advanced high performance aero-engines. Concurrently with the drive towards thinner and lighter blades, the engine's susceptibility to flow induced vibrations such as flutter increases. Flutter is defined as an unstable and self-excited aeroelastic phenomenon resulting from the coupled action between unsteady aerodynamic forces and blade motion: once the blade starts to vibrate, it induces unsteady aerodynamic forces, which in turn will act back on the blade surface. Such fluid-structure interaction leads to a net energy exchange per period between the vibrating blade and the working fluid and might result in structural failure within a short period of time. Turbomachinery flutter investigation still represents a challenging research field.

Typically, unsteady CFD calculations are performed as part of the flutter analysis and the user must decide the level of the fidelity of the CFD calculation. This is always a trade-off between accuracy and computational effort. It is typical nowadays to perform 3D URANS calculations for a flutter analysis but it is not standard to include tip clearance flows. The influence of the tip clearance flow on the flutter analysis can be critical for some turbomachinery applications. If the flow near the tip is transonic then the tip clearance flow will strongly influence the flow field, which will also influence the unsteady work near the tip and the flutter stability of the whole blade. Tip leakage flow originates from the pressure side over to the suction side of the blade, creating the so called tip clearance vortex. Characterized by strong viscous forces and high velocity gradients, this region of the blade is related to energy losses and directly affects engine efficiency. The influence of tip clearance flow on flutter analysis has been investigated previously.

Fu et al. (2015) investigated the tip clearance effects on the aeroelastic stability of transonic axial compressor blades using two independent three-dimensional flutter prediction approaches (energy method and aeroelastic eigenvalue analysis). Tip clearances in the range from 0.17 mm to 2.70 mm were investigated and it was concluded that the aeroelastic stability was not monotonic with the size of the tip clearance. The least stable tip clearance predicted by both methods was 1.69 mm.

Gjodic et al. (2012) studied the influence of tip clearance modeling in predictions of aeroelastic response in an oscillating low pressure turbine (LPT) cascade with low subsonic outflow Mach number ($M_2 = 0.40$). The study validated different numerical models (full-scale time-marching RANS CFD simulations) against experimental data (experimental sector cascade with one oscillating blade in axial bending mode). The authors had obtained an equivalent result for the imaginary part of the complex unsteady force up to 70% span on their model without tip clearance when compared to the nominal tip clearance case (with size of 1% span, or 0.97 mm). Differences relative to the nominal tip clearance model at 90% span section amounted to about 25% between the cases. In addition, it was concluded that the resolution of the tip clearance CFD mesh in the spanwise direction of the gap might have a considerable impact on prediction accuracy. Nonetheless, the results have not shown significant variations in aeroelastic stability when small variations in the tip clearance size where imposed (0.7%, 1% and 3% span gap sizes, 0.68 mm, 0.97 mm and 2.91 mm respectively). Typical clearances for axial turbines lie in the range of 1-2% span according to Moore et al. (1988).

In contrast, experiments by Huang et al. (2006) found that the tip gap size had a significant influence on the unsteady flow response for an oscillating linear turbine cascade. The authors observed that stabilizing contributions could be observed near the tip for small tip clearances (1.25% chord or 1.79 mm), while a bigger gap size (5% chord or 7.15 mm) led to prevalent destabilizing contributions at the aft suction surface associated with a well-developed vortex.

More recently, Teixeira et al. (2015) investigated the influence of tip clearance in the aeroelastic stability of a LPT blade row with high subsonic outflow Mach number ($M_2 = 0.80$). Based on the work of Gjodic et al. (2012), the authors considered three tip clearance configurations: no gap and tip clearances of 1% and 3% span. The influence of the mesh resolution in the tip clearance was investigated. For the circumferential bending mode shape considered, the tip clearance flow did not appear to have a significant influence on the overall aerodynamic damping for the AETR rig geometry.
at the prescribed flow condition. Despite that, the study showed a significant stabilizing shift in the work coefficient on the pressure side for the 3% tip clearance near the tip.

The effects of tip clearance flow on turbomachinery blade flutter is highly case dependent and more work is required to validate CFD methods for the prediction of aerodynamic damping (flutter stability) when tip clearance flows are present. The focus of the present investigation is to further study the influence of tip clearance gap size in the aeroelastic response (i.e. flutter) of a LPT blade row with high subsonic outflow Mach number ($M_2 = 0.80$). More specifically, the goal is to build aerodynamic damping maps and observe differences in aeroelastic stability for different pitching axis positions for four tip gap sizes: 0%, 1%, 2% and 3% of span.

**METHODOLOGY**

The present investigation is carried out numerically using full-scale time-marching 3D viscous model to obtain the solution of the URANS equations in the time domain. All the flow field features, including nonlinearities, can be included in such unsteady flow solution. With the concept of traveling wave mode, a fixed phase difference in the unsteady flow field between two consecutive blade passages can be established. If this phase is known a priori, periodic boundary conditions can be employed in conjunction with the use of a limited number of passages to model an entire blade row, which significantly reduces the computation time of the simulations. Efforts in the development of such idea are attributed to Erdos et al. (1977) (who proposed the Direct Store method), Giles (1991) (with the Time-inclined method) and He (1990), who established the Shape Correction method.

The cascade is modeled numerically by employing the commercial computational fluid dynamics code ANSYS CFX® v.16.2. Based on the Shape Correction method, the software can represent the unsteady flow solution history at periodic phase-shifted pitch-wise boundaries as well as at inter-row interfaces (in case of multi-row turbomachinery studies) in terms of Fourier series. The so-called Transient Blade Row Fourier Transformation (Patange, 2014) method in ANSYS CFX® v.16.2, adopted in the present investigation, results in a much lower computational time when compared to a full wheel blade row flutter analysis and makes feasible the study of several cases of such engineering problems. The potential of using the TBR-FT method for unsteady aerodynamic investigations is shown in the work of Camara (2015). Elder et al. (2013) also points out that the TBR-FT method provides solutions 3.5 times faster than average periodic reference cases.

In order to assess the blade row flutter stability, the energy method is employed. It assumes that flutter occurs with the blade natural mode shape (or some prescribed oscillation motion) and makes a flutter prediction by calculating the energy exchanged between the blade and the flow field. When mechanical damping is considered, the sum of mechanical and aerodynamic work determines the system stability. If the total work per vibration cycle done by the surrounding flow field (or the sum of mechanical damping dissipation and aerodynamic work) on the blade is negative, then the rotor is unstable, i.e. prone to flutter. In the present study, mechanical damping is not considered. Instead, only the aerodynamic work per cycle is taken to assess flutter stability, as given by the equation below.

$$W_{aero} = -\int_{t}^{t+T} \int_{S} p_{\text{avg}} \cdot \mathbf{n} dS dt$$ [1]

It is often convenient to express flutter stability in terms of the non-dimensional quantity aerodynamic damping, given by:

$$\bar{h} = \frac{h_{\text{max}}}{c} \quad \rightarrow \quad \Xi = \frac{W_{aero}}{\frac{1}{2} \rho c^2 \bar{h}^2}$$ [2]

While the aerodynamic damping is a measure of the overall stability, it does not show any detail about the underlying physics. This can be examined by looking at the local work coefficient on the blade, which is calculated for simple mode shapes as,
Flutter is predominately seen in one of the first few modes (bending and torsion) which can be approximated as rigid body modes. Since LPT flow characteristics are remarkably linear for small amplitudes of oscillation, the stability of any rigid body mode can be approximated by applying the principle of superposition to the three fundamental mode shapes: two orthogonal translations and a rotation. To illustrate this effect, Panovsky and Kielb (2000) created a preliminary design tool called the “tie-dye” (TD) plot, which cleverly combines the interdependency of critical reduced frequency and mode shape all in one plot. In such plot, engineers can assess the aeroelastic stability of a blade row for range of pitching axis location, in a 2D plot. Fig. 1(a) is an example of TD plot for an LPT blade at a fixed exit Mach number. According to Panovsky et al. (1997) and Nowinski et al. (1998), the most important design parameter identified through the experimental correlation effort is the mode shape. Superposition of three fundamental modes, as shown in Fig. 1(b)-(d), allows to obtain the equivalent of the torsion axis at any location. This technique can also address bending modes, since a torsion axis at infinity generates a pure bending mode.

Figure 1 – (a) Critical (flutter) reduced frequency as a function of torsion axis location near the reference blade and three fundamental mode shapes, (b) circumferential bending, (c) axial bending, (d) torsion about 40% axial chord, i.e. \((\xi, \eta) = (0,0)\).

By exploiting linearity, the unsteady solution for torsion of amplitude \(\alpha\) about any arbitrary location \((\xi, \eta)\) can be determined by the superposition of the three fundamental modes: circumferential bending, axial bending and torsion about the origin. Using vector relations, Panovksy (1997) showed that the amplitude of these fundamental modes is given by:

\[
a(\xi, \eta; \alpha) = \begin{bmatrix}
h_\xi & h_\eta & \alpha^T
\end{bmatrix} = \alpha \begin{bmatrix} \eta & -\xi \end{bmatrix} 1^T
\]

Following the approach proposed by Whitehead (1987) and Panovsky and Kielb (2000), the work for a torsion axis at the point \((\xi, \eta)\) can be obtained directly by superposing the solutions for the three fundamental modes through the expression:

\[
W_{\text{aero}}^\sigma = \alpha \begin{bmatrix} \eta & -\xi \end{bmatrix} 1 = \begin{bmatrix} W_{\xi\xi}^\sigma & W_{\xi\eta}^\sigma & W_{\xi\alpha}^\sigma \\
W_{\eta\xi}^\sigma & W_{\eta\eta}^\sigma & W_{\eta\alpha}^\sigma \\
W_{\alpha\xi}^\sigma & W_{\alpha\eta}^\sigma & W_{\alpha\alpha}^\sigma
\end{bmatrix} \begin{bmatrix} \eta & -\xi \end{bmatrix} 1^T = a^T b^* a
\]

Where \(W_{ij}^\sigma\) is aerodynamic work done by unsteady pressure from mode shape \(i\), acting on mode shape \(j\), for interblade phase angle \(\sigma\). The aerodynamic damping for an arbitrary torsion pitching axis is given by:
\[
\bar{\alpha} = h_{\text{max}} / \left[ c \left( \sqrt{\xi^2 + \eta^2} + \sqrt{\xi - 1}^2 + \eta^2 \right) \right] \quad \rightarrow \quad \Xi = W_{\text{aero}} / \kappa \kappa q \bar{c}^{2} \bar{\alpha} \]

[6]

Where the term inside brackets is used in order to keep the aerodynamic work finite as \((\xi, \eta) \to \infty\) (recovering bending modes), as proposed by Panovsky and Kielb (2000).

**CASE BACKGROUND**

The geometry considered in this study was taken from the Aeroelastic Turbine Rig at the Royal Institute of Technology, Stockholm, Sweden. This experimental setup is characterized by a high subsonic low pressure axial turbine cascade. The blade features a real chord of 54.34mm at midspan and an aspect ratio of 1.78 (span of 97mm). The annular cascade is composed by 80 blades, with pitch to chord ratio of 0.62 at midspan. The nominal tip clearance is 1% of span (2.15% of axial chord), with hub and shroud radii of 383mm and 480mm, respectively (see Glodic et al., 2012 and Teixeira et al., 2015 for more specifications).

The investigated blade oscillation motions considered were simplified rigid body bending modes in the circumferential and axial direction and one torsion mode about 40% axial chord, as depicted in Fig. 1. The displacement was constant from hub to tip. The oscillation's maximum nondimensional displacement corresponds to \(h_{\text{max}} = 0.25 \text{ mm or 0.5% of the midspan's axial chord. A reduced frequency of } k^* = 0.3 \text{ based on half-chord was used, corresponding to a natural frequency } \omega = 213.87 \text{ Hz.}

**STEADY STATE ANALYSIS**

The AETR blade row was modeled in ANSYS CFX® v.16.2 using a mesh of finite volumes. The Reynolds stresses terms of the RANS equations were computed using a standard k-\(\varepsilon\) turbulence model with wall functions. High resolution scheme is used to deal with the advection term and second order backward Euler scheme is used for the transient term. Convergence was said to be achieved when the RMS residuals of momentum and mass variables reached the target value of \(1 \times 10^{-4}\).

The inlet boundary condition is set by a total pressure profile from hub to shroud, constant total temperature (304 K), flow direction (26 deg) and turbulence intensity (5%). The average total pressure at the inlet was approximately 153.5 kPa. At the outlet a constant static pressure is prescribed (106.4 kPa). The single passage (Fig. 2) hexahedral fluid mesh was generated with ANSYS CFX-TurboGrid® v.16.2 using the Automated Topology and Mesh (ATM) method. Four case-studies were considered: no tip clearance, 1% (0.97mm), 2% (1.94mm) and 3% of span (2.91mm). The tip gap was modelled with different spanwise resolutions above the tip: 12 cells for the 1% gap size, 24 cells for the 2% gap size and 36 cells for 3% gap. This selection is based on the study of Teixeira et al. (2015). The grid is of multi-block type and contains both O- and H-blocks. Based on the mesh convergence study of Teixeira et al. (2015), all steady and unsteady calculations used a medium resolution mesh, with approximately 256k cells.

![Mesh Diagram](image)

Figure 2 – Full wheel (left), single passage computational domain (center) and 3% span gap mesh cut view(right) with medium resolution mesh.
Experimental results obtained by Glodic et al. (2012) were used as validation, as shown in Fig. 3.

![Figure 3](image1.png)

**Figure 3 – Steady blade loading at (a) 50% and (b) 90% span data validation \(M_2 = 0.4\)**

Blade loading curves comparing the solutions calculated with different tip clearances at 50%, 90% and 95% span are shown in Fig. 4. Corroborating to the results of Teixeira et al. (2015), at 50% span, differences between the loading curves are small because the influence of the tip flow is limited to the vicinity of the tip. At 90% span, differences in the blade loading can be seen particularly on the suction side. At 95% span, there are significant differences in the blade loadings particularly for the 3% tip clearance case where an increased drop in pressure on the suction side near mid-chord takes place. This pressure drop is due to the increased tip leakage flow and the interaction of tip leakage vortex with passage flow, leading to a region of high Mach number on the suction side of the blade (Fig. 5). There is also a significant change in static pressure on the pressure side at 95% span for the 3% tip gap. Static pressure is lower in the front half and slightly higher past 25% of the blade’s chord.

Contour plots of Mach number at mid-chord axial slices near the tip region for the four tip clearance cases are shown in Fig. 6. The changes in the flow-field due to the tip clearance are non-linear as the size of the tip clearance is increased. The tip flow for the 2% span gap size causes a reduction in Mach number on the suction side just below the tip while there is an increase in the same region when the tip gap is 3% of span.

![Figure 4](image2.png)

**Figure 4 – Steady blade loading at (a) 50%, (b) 90% and (c) 95% span**

![Figure 5](image3.png)

**Figure 5 – Mach number contour plot at 95% span with (a) 0% and (b) 3% span gap size**
Figure 6 – Mach number contour plot at 50% normalized streamwise location for (a) 0%, (b) 1%, (c) 2% and (d) 3% span tip clearance

UNSTEADY STATE ANALYSIS
The case-studies selected for the aeroelastic investigation comprised: (1) no tip clearance, (2) 1% span gap size with 12 cells, (3) 2% span gap size with 24 cells and (4) 3% span gap size with 36 cells spanwise resolution at the clearance. The unsteady flow analyses were conducted using the TBR-FT approach in ANSYS CFX® v.16.2. For each case study unsteady flow solutions were calculated for the following IBPA: -180°, -135°, -90°, -45°, 0°, 45°, 90°, 135° and 180°. The unsteady flow simulations in the time domain comprised five blade oscillation periods with 25 time steps per period for each tested IBPA. The flow solution was set to be reconstructed in time using two Fourier coefficients on the periodic boundaries of the domains, since no significant differences in aerodynamic damping resulted from storing a higher number of coefficients. The blade oscillation period is set as 213.87 Hz with maximum displacement equals 0.25 mm (0.5% of the axial chord).

RESULTS AND DISCUSSION
Fig. 7(a) shows the aerodynamic damping aeroelastic stability parameter for various IBPA’s for different tip gap sizes and for the three fundamental blade mode shapes. The tip clearance flow does not appear to have a significant influence on the aerodynamic damping for the AETR rig geometry at the prescribed flow condition for the bending modes. The least stable mode can be identified at \( \sigma = -45° \) for the circumferential bending case, for all tip clearance cases investigated. These results are in agreement with Teixeira et al. (2015). The axial bending case exhibited a stable profile for the whole range of IBPA tested. Despite that, the torsion mode, with pitching axis at \((\xi, \eta) = (0,0)\), revealed a significant difference in aerodynamic damping for \( \sigma = 45° \). In addition, a 20% increase in \( \Xi \) was perceived between the 3% gap size and the no clearance case, although the increase in gap size resulted in a non-linear curve, as shown in Fig. 7(b).

In order to examine the aeroelastic behavior in greater detail, plots of aerodynamic damping as a function of span height for IBPA = 0° and IBPA = 45° are shown in Fig. 8. At IBPA = 0°, the plots did not show significant differences between different gap sizes. This can also be seen in Fig. 7. At IBPA = 45° the plots are different near the tip for the torsion and circumferential bending mode shapes. In addition, for this IBPA, the increase in gap size stabilizes the blade row in torsion, but destabilizes it for circumferential bending. Overall, the magnitude influence of the tip clearance flow on the aerodynamic damping appears to be dependent on the interblade phase angle.
Figure 7 – (a) Nondimensional aerodynamic damping stability parameter vs $\sigma$ and (b) normalized $\Xi$ for $\sigma = 45^\circ$, torsional mode shape vs tip clearance size

Figure 8 – Nondimensional aerodynamic damping stability parameter vs span position for (a) $\sigma = 0^\circ$ and (b) $\sigma = 45^\circ$

Fig. 9 depicts the distribution of local work coefficient on the blade’s surface for the least stable nodal diameter of the torsional mode (negative = unstable). The cases with tip clearance exhibit a progressively smaller peak of the local work coefficient near the blade’s tip at about 10% axial chord, on the suction side. This trend is more explicit on the local work coefficient distribution from the 90% span position upwards, as depicted in Fig. 10.

Figure 9 – Local work coefficient for, (a) no tip clearance, (b) 1% tip clearance, (c) 2% tip clearance, (d) 3% tip clearance for $\sigma = 45^\circ$, torsional mode shape.
Figure 10 – Local work coefficient distribution along the blade's chord, $\sigma = 45^\circ$, at (a) 70\% ,
(b) 90\% and (c) 95\% span, torsional mode shape.

With the unsteady state results, aerodynamic damping maps could be constructed for each of the interblade phase angles analyzed. Results were taken for a single reduced frequency $k^* = 0.3$ based on half-chord, therefore, the data is one step before obtaining a complete critical reduced frequency TD plot, which would require repeating the analysis for at least five more reduced frequencies. The compilation of plots for the analyzed range of IBPA generates the global plots shown in Fig. 11 for the case with no tip clearance. Fig. 11(a) reveals contour levels of aerodynamic damping that are either stable (towards blue) or unstable (towards red). The regions toward the upper right and lower left are the least stable, and are related to $\sigma = 45^\circ$, 90$^\circ$ and 135$^\circ$. The most stable regions are toward the upper left and lower right corners of this plot. On the reference blade itself, a torsion axis on the forward portion of the blade is more stable than on the aft portion.

Figure 11 – (a) near field and (b) far field aerodynamic damping map for $\sigma \in [-180^\circ, 180^\circ]$ , no tip clearance

As pointed by Panovsky and Kielb (2000), the behavior in a more global sense is also investigated by considering a much larger range of possible torsion axis locations, e.g. $(\xi/c, \eta/c) \in [-10, 10]$, as shown in Fig. 11(b). This larger range provides the behavior for bending dominated modes, though the analysis is still performed by considering torsion about distant points. Fig. 11(b) confirms the behavior shown in the Fig. 7: axial bending mode shapes (obtained when $\xi/c \rightarrow 0$ and $\eta/c \rightarrow \pm\infty$) are predominantly stable (since the least stable nodal diameter has $\Xi > 0$); circumferential bending mode shapes (obtained when $\xi/c \rightarrow \pm\infty$ and $\eta/c \rightarrow 0$) are predominantly unstable (since the least stable nodal diameter has $\Xi < 0$).

Finally, it is interesting to investigate the influence of tip clearance on the aerodynamic damping map of the LPT blade row analyzed here. More specifically, the interest lies on determining the change in sign of any region in such plots, indicating that the presence of tip leakage flow induced either a more stable or unstable aeroelastic behavior of the cascade. Fig. 12 was constructed by multiplying the aerodynamic damping values for each location of torsion pitching axis of one the
clearance cases by the corresponding values of the no gap configuration, previously shown in Fig. 11. It is observed that there is a rounded shape region on the lower right side of the map that becomes unstable (red color) if the gap size is 1% of the span. However, if the gap size is 3% of the span, an adjacent region becomes stable (blue color).

Figure 12 – Near field change in aerodynamic damping map due to tip clearance, comparison between no clearance and (a) 1%, (b) 2%, (c) 3% span tip gap sizes.

CONCLUSIONS
The influence of tip clearance on the aeroelastic behavior of an oscillating low pressure turbine (LPT) blade row have been investigated through full-scale time-marching unsteady RANS CFD simulations employing models with and without tip clearance using the commercial software ANSYS CFX® v.16.2. Expanding the work of Glodic et al. (2012) and Teixeira et al. (2015), the study considered four tip clearance cases: no gap and tip clearances of 1%, 2% and 3% of the span. The no gap configuration was taken here as theoretical reference since it cannot be attained in practice. The effect of increasing the size of the tip clearance on the steady flow was found to be non-linear, particularly on the suction side. There was a significant drop in static pressure at 95% span on the suction side near mid-chord for the 3% tip clearance but not for the 1% tip clearance. The stronger tip clearance flow for the 3% gap creates a region of high Mach number on the suction side that is not present for the 1% gap.

All the four tip clearances scenarios pointed the same interblade phase angle as being the least stable TWM of the LPT cascade: \( \sigma = -45^\circ \) for circumferential bending, \( \sigma = 0^\circ \) for axial bending and \( \sigma = 45^\circ \) for torsion about 40% axial chord. Tip clearance flow does not appear to have a significant influence on the overall aerodynamic damping for the AETR geometry at the prescribed flow condition for the bending mode shapes, corroborating the results of Teixeira et al. (2015). However, the presence of tip gap led to a stabilizing effect on the least stable TWM for the torsion oscillatory mode, with a maximum increase in flutter stability of +20% for the 3% of span gap, compared to its zero clearance counterpart. The unsteady work done on the blade was investigated in more detail by examining plots of aerodynamic damping versus span and local work coefficient on the blade at various blade heights. Differences in the local work were observed, particularly near the tip.

The methodology proposed by Panovsky and Kielb (2000) in utilizing the stability parameter in conjunction with superposition of fundamental bending and torsion modes was implemented for the LPT row. For torsion-dominated modes, a torsion axis located on the forward portion of the airfoil was found to be more stable than one on the aft portion. For modes which were dominated by bending, the most stable region was for torsion axes along the cascade circumferential direction. The least stable region lies along the cascade axial direction. The influence of tip clearance gap size on aerodynamic damping maps indicated that the presence of tip leakage flow influences the aeroelastic behavior of the cascade. A rounded shape region on the lower right side of the aerodynamic damping
map became unstable for the 1% span tip gap. However, if the 3% span gap size was present, an adjacent region became stable.

For the unshrouded rotor blade configuration considered, the results of tip clearance and aerodynamic damping studies revealed that pitching axis location was extremely deterministic to the aeroelastic behavior of the LPT cascade. A future work is the analysis of flutter for different reduced frequencies, which would allow the construction of a complete critical reduced frequency TD plot.

In addition, it would be interesting to conduct a similar investigation of tip leakage flow and its effect on aeroelastic behavior on shrouded blades as future work. In such configurations, the geometry of the region where the leakage over shrouded blades is formed is usually more complex than that of the tip leakage over unshrouded blades and assumes the shape of a labyrinth. Thus, unlike in the cases investigated in the present study, the leakage flow over the shroud does not roll-up into a vortex structure but forms an axisymmetric mixing zone. Given the aforementioned strong link between the leakage vortex and the turbine blade row flutter response for unshrouded configurations, it is expected that leakage flow would have a reduced influence on flutter of shrouded rotor blades.

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