CFD ANALYSIS OF THE EFFECT OF PARABOLIC TAPER DISTRIBUTION OF AN UNTWISTED HELICOPTER ROTOR BLADE

Munir ELFARRA* and Mustafa KAYA
Ankara Yıldırım Beyazıt University, Faculty of Aerospace Sciences, Ankara, Turkey,
melfarra@ybu.edu.tr
Ankara Yıldırım Beyazıt University, Faculty of Aerospace Sciences, Ankara, Turkey,
mukaya@ybu.edu.tr
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ABSTRACT

The effect of parabolic taper distribution along the span of a helicopter rotor blade is analyzed in terms of the rotor thrust, torque and Figure of Merit. Various maximum chord length values are investigated. The Reynolds Averaged Navier-Stokes computations are done using the FINE/Turbo flow solver developed by NUMECA International. The Spalart-Allmaras turbulence model is used to calculate the eddy viscosity. The baseline blade is selected as the Caradonna-Tung rotor blade. Different blade shapes were generated by setting the maximum chord length at different spanwise locations for the same planform area as the baseline blade. Three optimum cases are observed: maximum Figure of Merit, maximum thrust for the baseline Figure of Merit and maximum Figure of Merit for the baseline thrust. Those optimum cases are noticed when the maximum chord length is 1.3 times the baseline blade chord length.

Keywords: CFD, Helicopter Blade, Taper Distribution.

PARABOLİK SİVRİLME DAĞILIMININ BURULMASIZ BİR HELİKOPTER ROTOR PALI ÜZERİNDEKİ ETKİSİNİN HAD ANALİZİ

ÖZET


Anahtar Kelimeler: HAD, Helikopter Pali, Sivrilik Dağılımı.

1. INTRODUCTION

Increasing the helicopter rotor blade performance in hover is a challenging mission for the designers. The helicopter rotor flow is complicated due to instabilities and vortical motions. That is why accurate analysis with high fidelity means is required. Computational Fluid Dynamics is a strong tool that is able to capture the complicated flow features of the helicopter flows. However, the CFD analysis of rotor blades is harder compared to the analysis of fixed wing [1-8]. Renzoni et al. [1] have developed a method which solves the three dimensional time-accurate Euler equations on overlapping structured grids to analyze the rotor flow aerodynamics. Their method was successfully applied to different helicopter test cases. Gecgel [2] has developed a framework based on unsteady compressible three dimensional Navier Stokes equations for the flowfield analyses of single and coaxial helicopter rotor blades in hover and forward flight. The rotor flowfields were successfully captured and compared using different turbulent models.

* Corresponding Author
Compressible Reynolds Averaged Navier Stokes (RANS) equations were carried by out Pomin and Wagner [3] for 7A helicopter rotor blade in hover condition. The RANS solver was coupled with finite element model based on Timoshenko beam theory to account for the aeroelastic effects. The results were in good agreement with the measured data. The HOST dynamics code and the WAVES Euler aerodynamic solver were weakly coupled to calculate a flexible rotor trim for steady forward flight by Servera et al. [4]. This weak coupling method was implemented to 7A and 7AD rotors and the results obtained have shown a pronounced improvement for the predictions of torsion and pitching moment.

Park and Kwon [5] have developed a three dimensional Euler solver running in parallel environment to simulate the unsteady helicopter rotor aerodynamics using unstructured grid in hover and forward flights. They have concluded that the solver was a grid efficient and robust for the predictions of complicated unsteady rotor flowfields. Chen et al. [6] have developed a three dimensional Euler solver based on a finite volume upwind scheme to calculate the helicopter rotor flowfields in forward flight condition. Their results were in agreement with the experimental data.

Srinivasan and Baeder [7] have addressed the capabilities of a free wake Euler and Navier Stokes CFD methodology (TURNS) in calculating and capturing the helicopter rotor blade flowfields and acoustic details under hover and forward flight conditions. Their results were successfully compared with the experimental data. Allen [8] has developed an unsteady multiblock multigrid Euler solver and a structured multiblock grid generator to simulate the rotor in forward flight. His solver and grid generator was validated against experimental results.

Conlisk [9] explains in details the aerodynamic challenges corresponding to the analysis of helicopter rotors. Enhancing the performance of helicopter rotor blades is usually done by changing the distribution of the blade chord, blade twist and airfoil section in the spanwise direction.

Different optimization techniques were conducted in literature to find the optimum rotor blade shape for higher performance using CFD and experimental studies [10-16]. Le Pape and Beaumier [10] present a procedure to optimize the rotor geometry using CFD tools. As a case study, they have shown that it is possible to improve the hover efficiency. A gradient-based discrete adjoint method coupled with a RANS solver was implemented by Dumont et al. [11] to optimize blade shape using many design variables. Their method succeeded in providing interesting blade shapes using little computer resources.

Allen and Rendall [12] have stated their work as the first free form design optimization of a rotor blade using compressible CFD. The results they obtained lead to large geometric deformations which exhibited a significant reduction in torque values. The researchers Choi et al. [13] have optimized the UH-60A rotor blade using an adjoint-based optimization technique and a Navier-Stokes solver in a time-spectral form. They achieved a 5% decrease in torque with almost no loss of thrust. According to the wind tunnel tests conducted by McVeigh and McHugh [14], a significant advance in conventional rotor capability has been demonstrated. They have concluded that further improvements can be gained from planform-structural tailoring.

Walsh et al. [15] have described an optimization procedure for helicopter rotor blade designs which minimizes hover horsepower while assuring satisfactory forward flight performance. The resulting optimization system provides a systematic evaluation of the rotor blade design variables and their interaction. Vua and Leen have presented the results of their optimization study to minimize the power required during hover and forward flight. For this purpose they combined CFD tools with a geometry generator which uses twist, taper ratio, point of taper initiation, blade root chord, and coefficients of the airfoil distribution function as the design variables. The power was reduced by about 4% while airfoil characteristics improved to a desired range.

Many studies use Caradonna-Tung rotor blade as baseline [17-20]. The baseline blade in this study has also been selected same as that blade. In the current study, the baseline Caradonna-Tung [21] helicopter rotor blade shape is modified without changing the blade planform area, and therefore the solidity of the rotor. The parabolic taper distribution is obtained by maximizing the chord length at a certain spanwise location. The maximum chord length used is ranging from 1.1 to 1.5 times the baseline blade chord. The analyses are carried out using a 3-D RANS CFD solver. The results are presented in terms of thrust, torque, Figure of Merit and sectional distribution of pressure coefficient.

2. METHODOLOGY

2.1 Flow Solver

All the computations in this study are done using the commercial CFD package FINE/Turbo [22] developed by Numeca International. The FINE/Turbo solver is a three-dimensional, density-based, structured, multiblock finite volume code. The mesh around the blades is generated using O4H grid topology. The grid generation software employed is IGG/AutoGrid5 [23].

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of Numeca International. A schematic of the 5-block mesh is shown in Figure 1 and described below.

1. an O block around the blade
2. a H block upstream the leading edge of the blade
3. a H block downstream the trailing edge
4. a H block up to the blade section
5. a H block down to the blade section

Figure 1. O4H grid block structure used.

2.2 Caradonna-Tung Rotor Blade

The experimental study of Caradonna-Tung helicopter rotor blade was conducted in the Army Aeromechanics Laboratory’s hover test facility [21]. The rotor has two untwisted and untapered blades with a precone angle of 0.5 degree and aspect ratio of 6. The blades of this rotor the symmetric NACA 0012 airfoil section from root to tip. The experimental study was done for various collective pitch angles ranging from 0 to 12 degrees. The rotor radius is 1.143 m and the fixed chord length along the span is 0.1905m. The taper stacking point (taper axis) of the baseline blade is at 25% chord from leading edge.

The geometry used for the numerical analysis in the present study is such that the blades are attached to the hub through a circular section. The airfoil transition starts at 10% of the blade span in the radial direction from the rotor rotating center.

2.3 Cases Analyzed

The parabolic taper distribution is obtained by maximizing the chord length at a certain spanwise location provided that the blade planform area is the same as the baseline blade area. The maximum chord length used in the present study is ranging from 1.10 to 1.50 times the baseline blade chord (c). The taper stacking point location is kept fixed at 25% chord as in the original blade.

The cases analyzed are summarized in Table 1. Since the blade planform area is kept fixed, some cases are not geometrically generated. The generated and analyzed cases are shown by sign X in the table.

All the analyzed cases are investigated at the rotational speed of 1750 RPM under the collective pitch angle of 8 degrees. For the cases in which the maximum chord length is 1.3 times the baseline chord length, the blade planforms are sketched in Figure 2.

3. VALIDATION STUDY

The space and time resolution is determined according to the results of validation study. The validation study is performed in the framework of Caradonna-Rung experimental rotor blade [21]. The mesh was generated for a single blade while the periodic condition was applied to account for the other blade. The number of points on the entire mesh including the blade and external flow is about 7 million points. The thickness of the first cell to the wall was kept at 3x10^{-6} m so that the y⁺ value is close to 1, which is suitable for the implemented Spalart-Allmaras turbulence model and the Reynolds number which is about 4000000. The meshes for all the cases are generated in the same way. The mesh of the baseline blade is shown in Figure 3.

The validation case is the experimental data for collective pitch angle of 8 degrees and rotational speed of 1750 RPM. The computed sectional lift (thrust) coefficients in the spanwise direction are compared to the experimental values in Figure 4. The experimental results are plotted in two different ways; one way is directly obtained from the lift coefficient data of Caradonna-Tung experiment report, the second way is obtained by integrating the pressure coefficients of the experiment along the chord of the corresponding section. The second way is also used in [18].

Pressure coefficients calculated at different spanwise sections are also validated against the experimental data for rotational speed of 1750 RPM as shown in Figure 5. The computational results for both pressure coefficient and sectional lift coefficient are in a good agreement with the experimental data.
Table 1. The analyzed cases.

<table>
<thead>
<tr>
<th>Maximum chord length by baseline chord length</th>
<th>Maximum chord length location in the spanwise direction</th>
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</thead>
<tbody>
<tr>
<td></td>
<td>10%</td>
</tr>
<tr>
<td>1.1</td>
<td>X</td>
</tr>
<tr>
<td>1.2</td>
<td>X</td>
</tr>
<tr>
<td>1.3</td>
<td>X</td>
</tr>
<tr>
<td>1.4</td>
<td>X</td>
</tr>
<tr>
<td>1.5</td>
<td>-</td>
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</tbody>
</table>

Figure 2. Top view of the blade planforms: (a) baseline blade, (b) maximum chord is at 10% span, (c) maximum chord is at 40% span, (d) maximum chord is at 70% span, (e) maximum chord is at 100% span.

Figure 3. 3-D mesh structure of the baseline blade.

Figure 4. Comparison of sectional lift coefficient.
Figure 5. Comparison of pressure coefficient distribution at different spanwise sections for 1750 RPM.
4. RESULTS

The flow solutions are computed according to turbulent flow assumption. The turbulence model used is the Spalart-Allmaras model. The steady-state flow solutions are obtained using local time stepping in each cell. All the computations are carried out in a parallel computing environment using 16 processors.

The computation of a typical flow solution for a 6th order convergence takes about 70-80 minutes of wall clock time.

The computed thrust and torque values for all the analyzed cases are plotted and compared to the baseline values in Figures 6 and 7.

![Figure 6. Thrust variation for all the cases.](image1)

![Figure 7. Torque variation for all the cases.](image2)

It is seen from Figure 6 that as the maximum chord location moves further from 60% span towards the blade tip, the thrust increases for all the cases. After 80% span location, the thrust is almost constant. Also it is observed that as the maximum chord length value increases (from 1.10c to 1.50c), the thrust increases as well. However, for the cases where the maximum chord location is less than 60% span, the thrust decreases as the maximum chord length value increases. From the same figure, one may notice that the thrust values calculated for all the cases are almost
same and equal to the baseline thrust value when the maximum chord location is at 60% span.

A similar trend for the torque variation is also obtained as seen in Figure 7: the torque increases after 60% span and becomes constant after 80% span. After 60% span, it also increases as the maximum chord lengths increases. Before 60% span, the torque decreases as the maximum chord length increases. In contrast to the thrust observations, although the torque values for all the cases are almost the same when the maximum chord location is at 60% span, they are less than the baseline torque value. Therefore, one may conclude that it is possible to have a configuration which provides more increase in thrust for less increase in torque.

The results of the calculated thrust and torque for all the cases are summarized in Tables 2 and 3 as a percentage increase compared to the baseline blade. From the tables, it is clear that the maximum thrust occurs when the maximum chord location is at 100% span and the chord length is 1.5c. However, at this point, the torque is also high. The minimum torque was obtained when the maximum chord location is at 10% span and the chord length is 1.4c. However, at this location the thrust value is very low. For a high performance helicopter rotor blade, the focus should be on maximizing the thrust while minimizing the torque.

The helicopter rotor blade efficiency during hover condition is usually expressed in terms of Figure of Merit (FM). More thrust generation is better for the helicopter performance. However, more torque generation means that more power is needed to attain that thrust, which makes the helicopter less efficient. The Figure of Merit is an indication of both the thrust and torque (Eqn. 1) [24].

\[
FM = \frac{c_3^{3/2}}{\sqrt{c_q}}
\]  

Table 2. Percentage thrust difference from the baseline thrust.

<table>
<thead>
<tr>
<th>Maximum chord length by baseline chord length</th>
<th>Maximum chord length location in the spanwise direction</th>
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<tbody>
<tr>
<td></td>
<td>10%</td>
</tr>
<tr>
<td>1.1</td>
<td>-4.9</td>
</tr>
<tr>
<td>1.2</td>
<td>-9.8</td>
</tr>
<tr>
<td>1.3</td>
<td>-16.0</td>
</tr>
<tr>
<td>1.4</td>
<td>-24.0</td>
</tr>
<tr>
<td>1.5</td>
<td></td>
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Table 3. Percentage torque difference from the baseline thrust.

<table>
<thead>
<tr>
<th>Maximum chord length by baseline chord length</th>
<th>Maximum chord length location in the spanwise direction</th>
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<tbody>
<tr>
<td></td>
<td>10%</td>
</tr>
<tr>
<td>1.1</td>
<td>-11.7</td>
</tr>
<tr>
<td>1.2</td>
<td>-19.8</td>
</tr>
<tr>
<td>1.3</td>
<td>-28.8</td>
</tr>
<tr>
<td>1.4</td>
<td>-39.1</td>
</tr>
<tr>
<td>1.5</td>
<td></td>
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Table 4. Figure of Merit.

<table>
<thead>
<tr>
<th>Maximum chord length by baseline chord length</th>
<th>Maximum chord length location in the spanwise direction</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>10%</td>
</tr>
<tr>
<td>1.0 (baseline)</td>
<td>0.491</td>
</tr>
<tr>
<td>1.1</td>
<td>0.516</td>
</tr>
<tr>
<td>1.2</td>
<td>0.524</td>
</tr>
<tr>
<td>1.3</td>
<td>0.531</td>
</tr>
<tr>
<td>1.4</td>
<td>0.534</td>
</tr>
<tr>
<td>1.5</td>
<td></td>
</tr>
</tbody>
</table>
Figure 8. Comparison of pressure coefficient distribution for maximum chord length of 1.3c located at 30% span at different spanwise sections.
Figure 9. Comparison of pressure coefficient distribution for maximum chord length of 1.3c located at 90% span at different spanwise sections.
Figure 10. Comparison of pressure coefficient distribution for maximum chord length of 1.3c located at 60% span at different spanwise sections.
From Tables 2, 3 and 4, the following observations can be drawn:

1- The maximum FM is obtained when the maximum chord length is 1.3c and located at 30% span. For this case, the thrust is decreased by 21% and the torque decreased much more to 36%. The increase in FM is about 9%.

2- The maximum thrust for the same FM as the baseline blade is obtained when the maximum chord length is 1.3c and located at 90% span. For this case, the thrust increases by 7%.

3- The maximum FM for the same thrust value as the baseline is obtained when the maximum chord length is 1.3c and located at 60% span. For this case, the FM value is 0.51 which is about 4% higher than the baseline FM.

The chordwise pressure coefficient, $C_p$, distribution at various span locations of the above three cases are compared with the baseline distributions in Figures 8-10. It is seen from the figures that as the sectional chord length becomes much smaller than the baseline chord length, the difference in $C_p$ distribution from the baseline blade gets much more pronounced. This difference is observed to be in the direction of increasing sectional thrust coefficient. On the other hand, when the sectional chord length increases, the difference in $C_p$ distribution from the baseline blade is less pronounced. Moreover, this difference is in the direction of decreasing sectional thrust coefficient. However, the decrease/increase in sectional thrust coefficient is not directly proportional to the total thrust of the blade.

5. CONCLUSION

The results obtained show that changing the location and the length of the maximum chord in the spanwise direction affects the thrust and the torque of the rotor blade. The change in thrust and torque is observed to be in the same direction but in different rates.

As the maximum chord location moves further from 60% span towards the blade tip, the thrust and torque values increase for all the investigated cases. At locations ahead of 80% span, the thrust and torque values are almost not changing. Moreover, as the maximum chord length value increases from 1.10c to 1.50c, both of the thrust and torque increase for all the maximum chord locations after 60% span.

For the cases when the maximum chord location is before 60% span, the thrust and torque decrease as the maximum chord length value increases from 1.10c to 1.50c.

Three cases of interest came into the picture after the analyses. The first case (maximum chord location at 30% span) of interest gives a maximum FM value which is about 9% higher than the baseline value. The second case (maximum chord location at 90% span) provides a maximum thrust value for the same FM as the baseline blade. In this case the thrust increases by 7% compared to the baseline blade. In the other case of interest (maximum chord location at 60% span), FM is increased by 4% without decreasing the thrust according to the baseline values. All of these three cases occur when the maximum chord is 30% more than the baseline blade chord length.

6. REFERENCES


**VITAE**

**Dr. Munir ELFARRA**

Dr. Elfarra has obtained his BSc, MSc and PhD from the department of Aeronautical (Aerospace) Engineering, Middle East Technical University in Ankara, Turkey.

His main focus is on the computational fluid dynamics of external rotating flows and wind energy assessment. He is currently a faculty member at Ankara Yildirim Beyazit University.

**Dr. Mustafa KAYA.**

Dr. Kaya has got his BSc degree from the Aeronautical Engineering Department at the Middle East Technical University (Ankara, Turkey). His MSc and PhD degrees are also from the same university under the Aerospace Engineering Department.

His interest areas mainly focus on the applications of computational fluid dynamics. Most of his researches are on the thrust and power production from flapping wings.

He is currently a faculty member at Ankara Yildirim Beyazit University in the Aeronautical Engineering Department. He has also about 10 years of experience in defense industry both as engineer and consultant.