# Aeroelastic Characteri stics of Rotor Blades with Trailing Edge Flaps

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## Abstract

The aeroelastic analysis of rotor blades with trailing edge flaps, focused on reducing vibration while minimizing control effort, are investigated using large deflection-type beam theory in forward flight. The rotor blade aerodynamic forces are calculated using two-dimensional quasi-steady strip theory. For the analysis of forward flight, the nonlinear periodic blade steady response is obtained by integrating the full finite element equation in time through a coupled trim procedure with a vehicle trim. The objective function, which includes vibratory hub loads and active flap control inputs, is minimized by an optimal control process. Numerical simulations are performed for the steady-state forward flight of various advance ratios. Also, numerical results of the steady blade and flap deflections, and the vibratory hub loads are presented for various advance ratios and are compared with the previously published analysis results obtained from modal analysis based on a moderate deflection-type beam theory.

**Key Word** : Aeroelastic analysis, Trailing edge flaps, Large deflection-type beam theory, Forward flight, Vibratory hub loads

## Introduction

Vibration has great impact on helicopter performance and reliability and is an issue in helicopter design. Vibratory motion usually involves fatigue of structural components, reduces availability and increases maintenance cost of helicopters. The need for vibration reduction is also even more critical when passenger and crew comfort is concerned. The most significant source of vibration in a typical helicopter is the main rotor. The main rotor blades are subject to a highly unsteady aerodynamic environment arising from the airspeed differential between the advancing and retreating side of the rotor disk. The blades themselves are elastic, vibrating in response to the airloads, and in turn the airloads are dependent on the blade elastic motions. Generally, studies on rotor blades have been performed for global deformation and cross-sectional analyses. One-dimensional global deformation analyses of rotor blades with consideration of the geometrical nonlinearity have been classified into two types of beam theory, moderate deflection and a large deflection. Most of the structural dynamic models for rotor blades are based on moderate deflection type beam theories. These theories are based on ordering schemes and are valid for moderate deflections[1,2]. A general purpose analysis, however, demands a large deflection model without any artificial restrictions on displacements or rotations due to the deformation and the degree of nonlinearity. The ordering scheme, although a valuable tool in special purpose research, is not desirable in a general purpose approach. To overcome the limitations of previous models,

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structural models that are valid for large deflection and are not based on ordering schemes have been developed. There are no small angle approximations made, and all kinematic nonlinear effects are included in the formulation. To date, there have been relatively few studies on aeroelastic analysis of rotor blades using large deflection-type beam theories[3,4]. The blade airloads and aeroelastic response are essentially periodic in forward flight. The loads observed in the rotating system occur at multiple integer harmonics of the rotational speed. For a rotor with $N_b$ identical blades, it is well known that the blade root shears and moments sum in such a manner that only $pN_b/rev$ loads are transmitted to the fixed system, with $p$ an arbitrary integer. The goal of most helicopter vibration reduction system is the reduction of this $N_b/rev$ motion. There are two different approaches for vibration reduction. One is to focus on the low vibration design process[5]. The other is to employ suppression devices. During the past several decades, active control of vibration has been investigated by many researchers. Active vibration control of the helicopter can be classified into two subcategories : the airframe-based control[6] and the rotor-based control. Among the various active rotor-based control approaches, higher harmonic control(HHC) is one such active method in which the swashplate is excited at the higher harmonics of the rotor rotational speed[7]. An alternative to HHC is the individual blade control(IBC). IBC permits a wide range of excitation frequencies by placing the actuators directly into the rotating frame. Among an implementation of IBC, the active trailing edge flap(ATF) approach has received considerable attention due to its simplicity of implementation and enhanced airworthiness[8]. Milgram et al. presented a comprehensive study of vibration reduction in helicopters using an active trailing edge flap[9]. The aeroelastic analysis in their study included a nonlinear aeroelastic rotor model, unsteady compressible aerodynamics of the flap, and a multicyclic flap controller. The analytical results presented in their study were validated using experimental wind tunnel data. Straub and Charles investigated the dynamics and aerodynamics of rotors with trailing edge flaps using two different aeroelastic codes-CAMRAD/JA and CAMRAD II [10]. Shen and Chopra developed a comprehensive aeroelastic analysis of a fully coupled blade-flap-acutator system. The objective of the study was to investigate the effect of this coupling on vibration reduction with trailing edge flaps[11]. Zhang et al. looked at active-passive vibration reduction using trailing edge flaps and optimal blade structural properties[12].

In this paper, the finite element approach using large deflection-type beam theory is presented for the aeroelastic analysis of rotor blades with trailing edge flaps in forward flight. The aerodynamic forces are modeled using a two-dimensional quasi-steady strip theory. Nonlinear, periodic blade steady responses are obtained using time finite element method on a full finite element equation in the forward flight condition. Blade responses fully coupled with vehicle trim are solved to obtain the nonlinear blade response, pilot controls, and vehicle attitude. The effect of vibration reduction using trailing edge flap is studied. The objective function, which includes vibratory hub loads and active flap control inputs, is minimized by an optimal control process. The full finite element results using the large deflection-type beam theory are compared with the results obtained by a modal approach using the moderate deflection-type beam theory.

## Theoretical Background

### Structural dynamic model

The geometrical nonlinearities are described using coordinate transformation matrices with the Euler angles in the present large deflection-type beam theory.

$$\mathrm{e}_i^* = \mathrm{t}_e(x_1)\mathrm{e}_i = \mathrm{T}(x_1)\mathrm{i}_i \qquad \mathrm{T}(x_1) = \mathrm{t}_e(x_1)\mathrm{t}_g(x_1) \tag{1}$$

The transformation matrices $\mathrm{t}_g$, $\mathrm{t}_e$, and $\mathrm{T}$ are functions of the curvilinear axial coordinate $x_1$. Assuming that initial curvatures are small and shearing strains are much smaller than unity in the Green-Lagrangian strain components, strain-displacement relations are represented as given in Ref. 12. The trailing edge flaps are assumed to be an integral part of the blade. The flap hinges are assumed to be rigid in all directions except about the hinge axis, thereby allowing only pure rotation of the flap in the plane of the blade cross section. It is also assumed that the flaps do not contribute to the stiffness of the rotor blade and influence the behavior of the blade only through their contribution to the blade aerodynamic and inertial loading. The equations of motion for a rotor blade are obtained using Hamilton's principle :

$$\int_{\psi_i}^{\psi_f} (\delta L + \delta W) d\psi = \delta q^T \mathrm{p}\Big|_{\psi_i}^{\psi_f} \tag{2}$$

$$L = T - V \qquad \mathbf{p} = \frac{\partial L}{\partial \dot{q}}$$

where $\delta V$, $\delta T$, and $\delta W$ are the variation of strain energy, the variation of kinetic energy, and the virtual work of applied forces, respectively. L is the Lagrangian of the system. The generalized coordinates $\dot{q}$ and $q$ are composed of displacements and Euler angles, while $\mathrm{p} = \mathrm{L}_{\dot{q}}$ is the column vector of the generalized moment. $\psi_i$ and $\psi_f$ represent the initial and final states of non-dimensionalized time, respectively.

## Aerodynamic model

In the present work, the aerodynamic lift and pitching moment acting on the blade are obtained by modified Greenberg's extension[13] of Theodorsen's theory for a two-dimensional airfoil undergoing unsteady motion in an incompressible flow (Fig. 1). Considering a large aspect ratio wing in incompressible and inviscid flow, the wing/aileron combination is undergoing two degrees of motion: plunge motion $h(t)$ and pitch motion $\varepsilon(t)$ about the blade elastic axis. The aerodynamically unbalanced trailing edge flap rotates about the flap hinge by the angle $\delta(t)$ relative to the chord line.

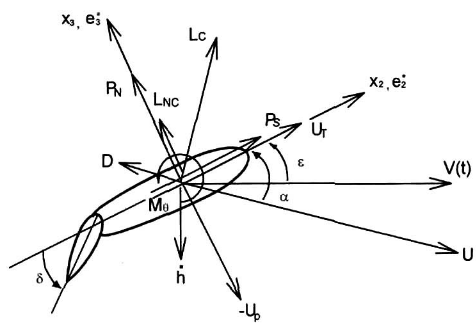


**Fig. 1. Rotor blade airfoil section in general unsteady motion**

Two-dimensional quasi-steady strip theory is used to evaluate aerodynamic forces in forward flight. The components of the resultant velocity $U$ in the deformed blade coordinate system are given by

$$\begin{Bmatrix} U_R \\ U_T \\ U_P \end{Bmatrix} = \mathrm{T} \begin{Bmatrix} \dot{u}_1 - \Omega R_{02} - \Omega R \mu \cos\psi \\ \dot{u}_2 + \Omega R_{01} + \Omega R \mu \sin\psi \\ \dot{u}_3 + \Omega R \lambda_i \end{Bmatrix} \tag{3}$$

where $\dot{u}_i$ is the component of the elastic velocity vector $\dot{\mathrm{u}}$ of the blade and $R_{0i}$ is the component of the position vector R of an arbitrary point on the cross-section in the deformed blade configuration. $R$ is the blade radius, $\Omega$ the constant angular velocity, $\mu$ the advance ratio, $\lambda_i$ the inflow ratio, and $\psi$ the azimuth angle of the blade. A linear inflow model is used for the rotor inflow distribution in forward flight.

## Blade steady response and coupled trim analysis

The nonlinear, periodic steady response is obtained using a time finite element technique. The virtual energy expression for Hamilton's weak form can be obtained as follows:

$$\int_{\psi_i}^{\psi_f} \delta y^T l d\psi = \delta y^T b \Big|_{\psi_i}^{\psi_f} \tag{4}$$

$$\delta y = \begin{Bmatrix} \delta\dot{q} \\ \delta q \end{Bmatrix}; \; l = \begin{Bmatrix} L_{\dot{q}} \\ L_q + Q \end{Bmatrix}; \; b = \begin{Bmatrix} 0 \\ p \end{Bmatrix}$$

where Q is the generalized forces. $L_{\dot{q}}$ and $L_q$ are the partial derivatives of L with respect to the generalized coordinates $\dot{q}$ and $q$, respectively. Using a first order Taylor series expansion of the left-hand side of Eq. (4) with respect to a given state vector $\bar{y}$, the following governing equation can be derived in an incremental form:

$$\int_{\psi_i}^{\psi_f} \delta y^T \bar{l} d\psi + \int_{\psi_i}^{\psi_f} \delta y^T \bar{K} \Delta y d\psi = \delta y^T b \Big|_{\psi_i}^{\psi_f} \tag{5}$$

where the local tangent matrix $\bar{K}$ is defined as

$$\bar{K} = \begin{bmatrix} L_{\dot{q}\dot{q}} & L_{\dot{q}q} \\ L_{q\dot{q}} + Q_{\dot{q}} & L_{qq} + Q_q \end{bmatrix} \tag{6}$$

where $L_{\dot{q}\dot{q}}$, $L_{\dot{q}q}$, $L_{qq}$, $Q_{\dot{q}}$, and $Q_q$ indicate the second and first derivatives with respect to the subscripts, respectively. The time period for one revolution is discretized into a number of time elements with cubic variation. After assembling elements in a global system, a periodic boundary condition is imposed by folding the row and column of the assembled matrix and vector. The propulsive vehicle trim analysis is fully coupled with the previous blade steady response analysis to solve the blade response, pilot control inputs, and vehicle orientation simultaneously. The vehicle trim solution is calculated from the overall nonlinear vehicle equilibrium equations: three force equations (vertical, longitudinal, and lateral) and three moment equations (pitch, roll, and yaw). However, the yawing moment equilibrium is neglected in this study because the tail rotor is not considered. Fixed frame hub loads are also calculated by summing the contributions from individual blades.

## Control Algorithm

In the present study, a single active trailing edge flap located at 70% blade span location is used to introduce control input directly in the rotating reference frame (Fig. 2). In steady forward flight, the helicopter rotor system can be assumed to be periodic in time. This periodic nature of the system allows us to transform the control problem from the time domain to the frequency domain[14]. The control algorithm is based on the minimization of an objective function that is a quadratic function of hub vibratory loads and control input magnitudes. In this study, the control input is the flap deflection angle itself. The objective function for optimal control is given by[12]

$$J = Z_n^T W_z Z_n + \delta_n^T W_\delta \delta_n \tag{7}$$

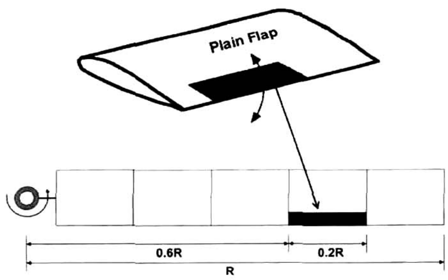


**Fig. 2. Configuration of rotor blades with trailing edge flaps**

where $Z_n$ is hub vibratory load vector containing $N_b/rev$ sine and cosine harmonics. $\delta_n$ is active flap control input vector containing cosine and sine higher harmonics, typically *3, 4, 5/rev* or plus *2/rev* harmonics for a four-bladed rotor. The trailing edge flap deflection as a function of azimuth can be written as,

$$\delta(\psi) = \sum_{i=2}^{5} [\delta_{ic}\cos(i\psi) + \delta_{is}\sin(i\psi)] \tag{8}$$

The optimal control input is obtained by the following optimality criteria:

$$\frac{\partial J}{\partial \delta_n} = 0 \tag{9}$$

## Numerical Results

Numerical results are obtained for a four-bladed, soft inplane, uniform hingeless rotor. The baseline rotor blade and trailing edge flap properties are shown in Table 1. A plain flap configuration is applied in this study (Fig. 2). It is assumed that the presence of flaps does not alter the mass distribution of rotor blades, and that the stiffness changes of baseline blades are negligible. The blade is discretized into five four-noded cubic elements in space domain and the time period of one rotor revolution is discretized into eight-noded cubic elements in time domain. The blade responses and vibratory hub loads are analyzed at two different advance ratios: the

**Table 1. Baseline blade and flap properties**

| Blade properties | |
|---|---|
| Number of blades, $N_b$ | 4 |
| Radius, $R$ $(ft)$ | 16.2 |
| Hover tip speed, $\Omega R$ $(ft/\mathrm{sec})$ | 650 |
| $c/R$ | 0.08 |
| Solidity, $\sigma$ | 0.1 |
| $C_T/\sigma$ | 0.07 |
| Lock number, $\gamma$ | 6.34 |
| $m_0$ $(slug/ft)$ | 0.135 |
| $EI_y/m_0\Omega^2R^4$ | 0.008345 |
| $EI_z/m_0\Omega^2R^4$ | 0.023198 |
| $GJ/m_0\Omega^2R^4$ | 0.00225 |
| Trailing edge flap properties | |
| Flap chord ratio, $c_f/c$ | 0.20 |
| Flap radius location | 0.60-0.80 |
| Flap chordwise CG, $r_I/c_f$ | 0.149 |

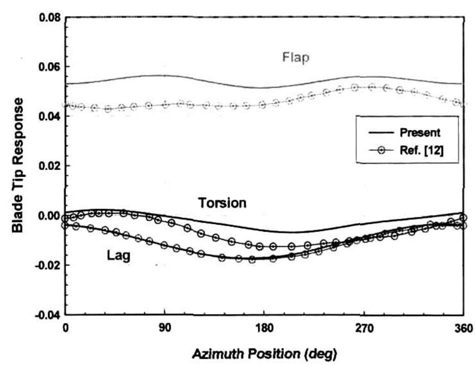


**Fig. 3. Blade tip response without flap deflection ($\mu = 0.15$)**

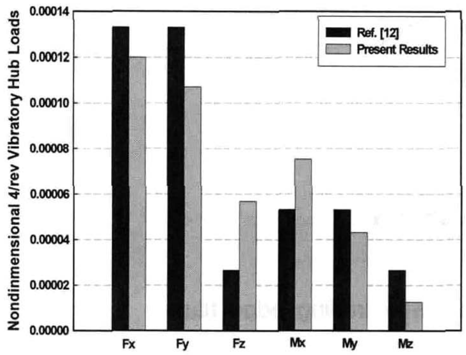


Fig. 4. The $4/rev$ vibratory hub loads without flap deflection($\mu=0.15$)

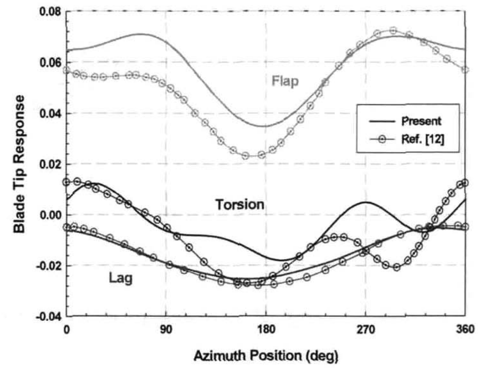


Fig. 5. Blade tip response without flap deflection ($\mu=0.35$)

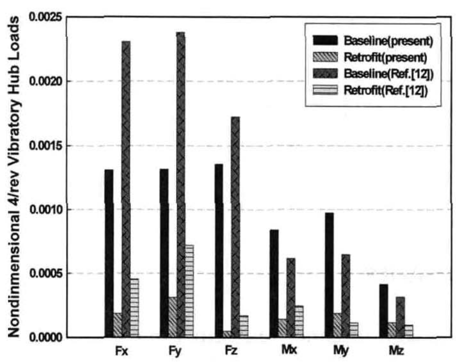


Fig. 6. The $4/rev$ vibratory hub loads for retrofit design ($\mu=0.35$)

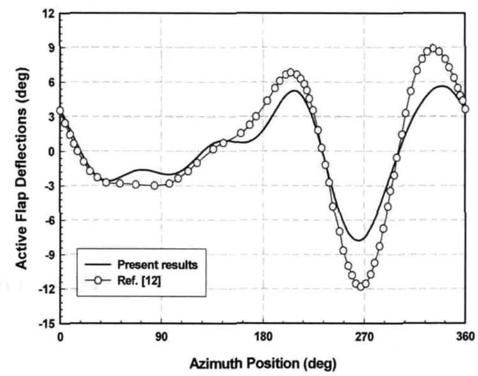


Fig. 7. Flap deflection for retrofit design ($\mu=0.35$)

high advance ratio of 0.35 and the low advance ratio of 0.15. The present results obtained by the full finite element analysis in forward flight using the large deflection-type beam theory are compared with the previous published results obtained by a modal analysis using the moderate deflection-type beam theory. The present paper focuses on investigating the effect of vibration reduction using trailing edge flap and nonlinear kinematic effects due to large deflections through a comparison of two models.

Figure 3 shows non-dimensional flap, torsion, and lag tip deflections of the blade for one revolution at an advance ratio $\mu=0.15$. The present analysis is compared with the previous results given in Ref. 12 and a relatively good correlation between these two results about lag and torsion deflections is shown in the Fig. 3. The results of Ref. 12 are obtained using the modal analysis with a modal basis of eight coupled rotating natural modes (3 flap, 3 lag, and 2 torsion modes). The $4/rev$ vibratory hub loads (longitudinal force $:F_x$, lateral force $:F_y$, vertical force $:F_z$, rolling moment $:M_x$, pitching moment $:M_y$, and yawing moment $:M_z$ for the hub nonrotating fixed frame) are shown at $\mu=0.15$ in Fig. 4. As shown in Fig. 4, the trends between two results are quite similar but some differences appear in magnitude. Figure 5 shows the blade tip response at an advance ratio $\mu=0.35$. There are somewhat differences between the results of two models and these differences increase as the forward speed increases because the nonlinear kinematic effects exist on a large scale. The blade flap and torsion responses have significant high harmonic components. The flap is actuated in 2, 3, 4, and $5/rev$ sine and cosine harmonics to reduce the vibratory hub loads. Figure 6 shows the simultaneously reduced $4/rev$ vibratory hub loads along with the baseline vibratory hub loads at an advance ratio $\mu=0.35$. The present analysis is compared with the previous results given in Ref. 12. The vibratory hub loads are reduced by 70~96 % from the baseline vibration level and a similar reduction trend between two results is observed. The corresponding active flap deflections are shown in Fig. 7. It is observed that $2/rev$ active flap inputs are the largest among the four input harmonics.

## Conclusions

In this paper, the aeroelastic analysis of rotor blades with trailing edge flaps in forward flight has been investigated. The finite element analysis is conducted using the large deflection beam model. Nonlinear, periodic blade steady response is computed using the time finite element method on full finite element equation with full coupling of the propulsive vehicle trim. The periodic steady tip deflections and vibratory hub loads for rotor blade with and without active flap control are compared with those obtained by modal approach using the moderate deflection-type beam theory. The results show that the nonlinear kinematic effects greatly affect the steady response and vibratory hub loads as the forward speed increases and that active trailing edge flaps are effective and efficient for rotor vibration reduction.

## Acknowledgement

This research has been supported by the KARI under KHP Dual-Use Component Development Program funded by the MOCIE.

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