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Control of Pitch-Flap Instabilities in Helicopter Rotors using Delayed Feedback

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Abstract: The problem of vibration suppression in helicopter rotors is the main topic of this paper. We revisit a strategy for the stabilization of such systems based on delayed feedback. By means of an improved analysis, made possible by recently developed mathematical tools, we present insights on how the delay can be used as stabilization tool for this problem. The results are supported with numerical simulations. For simplicity, only the case of hovering flight is considered.

Keywords: Helicopter vibration; Rotor instability; Delayed Feedback

1. INTRODUCTION

The main rotor is the principal source of vibrations in a helicopter and the reduction of these vibrations is important to increase the life of the components of the airframe as well as the comfort of passengers and crew. A considerable research effort has been devoted to the study of active control systems for helicopter rotors, in order to guarantee the stability of its motion. Most works on this topic fall in one of two categories: Higher Harmonic Control (HHC) or Individual Blade Control (IBC). Works in the HHC category, like those of Hall and Wereley (1992); Mura and Lovera (2014); Mura et al. (2014) consider the control forces to be applied to the swashplate by means of actuators. The IBC works (Bittanti and Cuzzola, 2002; Arcara et al., 2000; King et al., 2014; Shen et al., 2006) consider that each blade can be actuated independently. The main difference between the two approaches is the number of available degrees of freedom for the control action. Many control algorithms, nevertheless, can be implemented with either actuation technique. Friedmann and Millott (1995) compare several different control methods in both categories. A more recent survey of the developments in these topics was presented by Friedmann (2014).

The present paper revisits a control strategy presented by Krodkiewski and Faragher (2000), which can be implemented in both HHC and IBC frameworks. This control logic uses delayed feedback to improve the stability of the motion of the helicopter rotor. Delayed feedback has been previously used for the stabilization of periodic motion by Pyragas (1992) and for the suppression of vibrations by Olgac and Holm-Hansen (1994). Using novel analysis techniques, we challenge the published results and show a corrected stability region in the parametric space.

Although the original works of Faragher (1996) and Krodkiewski and Faragher (2000) consider both the cases of hovering and horizontal flight, we restrict the scope of this work to the case of vertical motion only. In this way, we can apply tools for the stability analysis of Linear Time Invariant (LTI) systems affected by delays. The case of horizontal flight, and the study of the Linear Time Periodic (LTP) systems with delays, is a matter of further research.

The paper is organized as follows. Section 2 introduces the dynamic model of the rotor in hovering flight, borrowed from Bramwell et al. (2001). Section 3 presents the instabilities that may appear in the operation of an uncontrolled rotor. Section 4 introduces the control law and the stability analysis. Some concluding remarks are finally presented in section 5.

2. DYNAMICS OF THE ROTOR

A helicopter rotor blade is mounted on a set of hinges which allow three angular degrees of freedom. A typical arrangement of the hinges is shown in figure 1. The *flapping* motion is defined as an up and down rotation in a plane which contains both the blade and the shaft. The flapping angle, β, is defined as positive when the blade moves upwards. A flapping blade rotating at high speeds is subject to large Coriolis moments in the plane of rotation, and the *lag* hinge is introduced to alleviate these moments. This hinge allows the motion of the blade in the same plane of rotation. The lagging angle, ξ, is positive when the blade moves in the same sense of the shaft rotation. The *pitching* or *feathering* motion, denoted by θ, is a rotation about an axis parallel to the blade span.

The pitch-flap flutter is an unstable behavior produced by the coupling of the pitching and flapping motions of the rotor. It can lead to very high oscillatory loads in the pitch control mechanism. The following dynamic model, adapted by Krodkiewski and Faragher (2000) from the work of Bramwell et al. (2001), describes the coupling between θ and β for an uncontrolled rotor in hovering flight

$$M\begin{bmatrix}\ddot{\theta}(\psi)\\ \ddot{\beta}(\psi)\end{bmatrix} + C\begin{bmatrix}\dot{\theta}(\psi)\\ \dot{\beta}(\psi)\end{bmatrix} + K\begin{bmatrix}\theta(\psi)\\ \beta(\psi)\end{bmatrix} = 0, \tag{1}$$

with the inertia, damping, and stiffness matrices:

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$$M = \begin{bmatrix} 1 & -12\frac{r_g\sigma}{c_h} \\ 0 & 1 \end{bmatrix}, \ C = \begin{bmatrix} \frac{\gamma}{8} & 0 \\ 0 & \frac{\gamma}{8} \end{bmatrix}, \ K = \begin{bmatrix} \nu_1^2 & -12\frac{r_g\sigma}{c_h} \\ -\frac{\gamma}{8} & \lambda_1^2 \end{bmatrix} \tag{2}$$

As it is customary in helicopter dynamics, the independent variable in (1) is not the time t, but the azimuth angle $\psi = \Omega t$. Here Ω is the angular speed of the rotor shaft, which is assumed constant during the operation of the aircraft. In (1) a dot over a variable denotes its derivative with respect to ψ.

It is important to state that in the derivation of (1) it is assumed that no lag motion occurs.

Among the parameters presented in (2), the stability is usually studied with respect to the two parameters which have the greatest influence on it. These parameters are σ, which is the distance of the center of gravity (c.g.) of the blade from the center of pressure, expressed as a fraction of the chord; and the non rotating torsional natural frequency ν_1. The other parameters, described in table 1, are taken as fixed. Notice that the natural frequencies ν_1 and λ_1 are given as fractions of Ω. Table 1 also shows the values used for the fixed parameters in the numerical analysis section. These values are taken from the work of Faragher (1996).

Using two sticks in the cockpit the pilot can exert a control moment which directly affects the pitch motion of the rotor. The basic effect of a pitch change is a change of the lift force produced by the blade. If all blades change the pitch simultaneously, the lift of the whole rotor is changed and vertical motion ensues. This is the so called collective pitch control and is changed using one stick in the cockpit. If the pitch of the blades changes as a function of its angular position, a directional force can be created and horizontal flight (forwards, rearwards, sideways) is attained. This is the cyclic pitch control and is controlled by a second stick in the cockpit. The uncontrolled model (1) is then expanded using an input signal $u(\psi)$, which corresponds to the motion of the lower end of an actuation rod. Following again Faragher (1996), the controlled model with the parametric values under consideration becomes

$$\begin{aligned} M \begin{bmatrix} \ddot{\theta}(\psi) \\ \ddot{\beta}(\psi) \end{bmatrix} + C \begin{bmatrix} \dot{\theta}(\psi) \\ \dot{\beta}(\psi) \end{bmatrix} + K \begin{bmatrix} \theta(\psi) \\ \beta(\psi) \end{bmatrix} \\ = Bu(\psi) = \begin{bmatrix} 3777\nu_1^2 \\ 0 \end{bmatrix} u(\psi) \end{aligned}. \tag{3}$$

Equations (1) and (3) are models for hovering or vertical flight only. In this situation, the matrices in (2) are constant, and we deal with linear, time-invariant (LTI) systems. When forward (or sideways) motion of the helicopter is considered, the aerodynamic forces and moments become time variant. More specifically, in these cases the matrices in (2) are periodic, with a time period $T = 2\pi\Omega$ or simply $T = 2\pi$ when the azimuth angle ψ is used as independent variable. This case is certainly more complicated and is left aside for future studies.

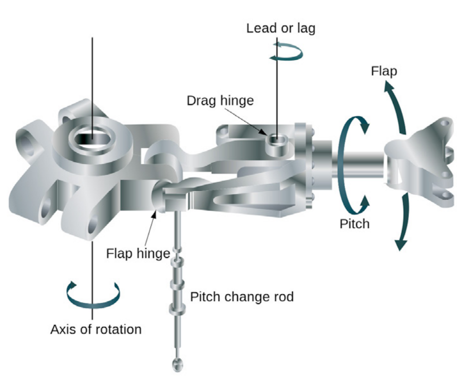

Fig. 1. Typical hinge arrangement and degrees of freedom of a helicopter rotor blade. (Source: Wikimedia Commons)

Table 1. Parameters of the Dynamic Model

Param.	Description	Value
r_g	Span wise location of blade's cg	4.1 m
c_h	Chord	0.527 m
γ	Lock Number	6.95
λ_1	First flap natural frequency	1.1
σ	Pos. of the cg from the center of pressure	$[0, 0.08]$
ν_1	Torsional natural frequency	$[0, \sqrt{8}]$

3. STABILITY ANALYSIS OF THE UNCONTROLLED SYSTEM

With the state vector $x(\psi) = \begin{bmatrix} \theta \, \beta \, \dot{\theta} \, \dot{\beta} \end{bmatrix}^T$, equation (1) can be represented in state space as the fourth order system

$$\dot{x}(\psi) = A_u x(\psi) = \begin{bmatrix} 0_{22} & I_{22} \\ -M^{-1}K & -M^{-1}C \end{bmatrix} x(\psi), \tag{4}$$

with 0_{22} being a 2-by-2 matrix with all elements equal to zero and I_{22} the identity matrix of the same dimension.

As the parameters σ and ν_1^2 change while keeping the other constant, two types of instability can appear. When A_u has a purely real unstable root, the unstable mode is known as *pitch divergence* and it is evidenced by a steady increase in the pitch and flap angles. When the unstable roots are complex conjugate, the oscillatory instability known as *pitch-flap flutter* appears.

The boundary at which the transition from stability to pitch divergence occurs is given by the presence of a characteristic root at the origin of the complex plane. A sufficient condition for A_u to have a root at the origin is that the determinant of the stiffness matrix is equal to zero, i.e., $|K| = 0$. From here we obtain

$$\nu_1^2 = \frac{3\gamma r_g \sigma}{2 c_h \lambda_1^2} \tag{5}$$

To establish the boundary between stability and flutter, it is necessary to find a conjugated pair of purely imaginary characteristic roots at a frequency ω_f. This is achieved by solving the characteristic equation $|j\omega_f I_{44} - A_u| = 0$, leading to the following two equations in terms of the crossing frequency

$$\nu_1^2 = 2\omega_f^2 - \lambda_1^2, \tag{6a}$$

$$\sigma = \frac{c_h \left(\gamma^2\omega_f^2 + 64\omega_f^4 - 128\omega_f^2\lambda_1^2 + 64\lambda_1^4\right)}{96 r_g \gamma \left(\omega_f^2 - 1\right)}. \tag{6b}$$

The two stability boundaries are presented in Fig. 2. The red, dashed line depicts the boundary at which an