A single rotor without reaction torque: a violation of Newton's Laws or feasible?

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Summary

It is shown in the paper that a single rotor without reaction torque is theoretically possible, thus eliminating the need for anti-torque devices. The principle is to actively excite the flapping motion so that the blades provide both lift as well as their own propulsion, similar to the wings of birds. It is furthermore investigated in the paper whether this principle could be used in a practical helicopter, considering the power required, the magnitude of flapping angles, vibrations, and control characteristics. The preliminary conclusions are that such an "ornicopter" could be practically feasible.

1 Notations

Restricted to those not defined in the text or in figures:

dL lift on blade element

I inertia moment w.r.t. flapping hinge

I_f inertia moment fuselage

K spring constant

m blade mass

r radius of blade element

R rotor radius

S static moment w.r.t. flapping hinge

t time

v_i induced velocity

V flight velocity

α_s shaft plane angle of attack

γ Lock number $\rho C_{l_\alpha} c R^4/I$

μ advance ratio $V/(\Omega R)$

ψ azimuth angle

Ω angular speed rotor

2 Introduction

The tail rotor of helicopters, needed to counteract the reaction torque of the engine and to control the helicopter in yaw, has always been considered a necessary evil. It is expensive, costs power (5 to 10% of the total power), it has only marginal control authority under unfavorable wind conditions, and is on top of that noisy, vulnerable and dangerous. Numerous attempts have been done to find alternative solutions. However, even the best of these alternatives solve only a part of the problem. The ideal solution would be to have a main rotor without reaction torque at all, which at first sight seems asking for a system which would violate Newton's laws.

Taking a closer look at bird flight may provide the answer nevertheless. In the usual helicopter the rotor blades have the same degrees of freedom as bird wings: the blades are free to flap up and down, to have a lead-lag motion in their plane of rotation, and they may be rotated around their span axis (pitching). In conventional helicopters only the pitch is actively controlled by means of the swash plate, so that the thrust can be vectored to obtain longitudinal or roll control of the aircraft.

The wings of birds possess the same degrees of freedom, but all of them are actively used so that by suitably coordinating the different wing motions birds are able to derive lift, control as well as propulsive force from their wings (see fig. 1).

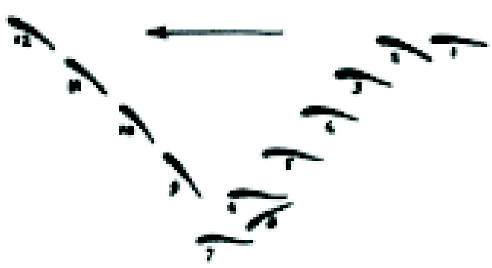

Figure 1: Motion of bird wing, according to Magnan (Ann. Sci. Nat. Zoo (10)5, 1922)

In principle a similar coordination of motions should be possible in the case of helicopter blades, by applying a direct flapping moment to the blade roots with the correct phase angle. In this way the rotor blades would propel themselves like bird wings, without needing shaft torque and therefore without the need for an antitorque device. The necessary power to drive the rotor is in such a case entirely supplied by the flap forcing mechanism, again similar to a bird where the necessary power for flight is supplied by the flapping muscles.

During the patenting of this idea by Delft University of Technology it was discovered that the principle had already been independently suggested by Vladimir Savov on a Bulgarian website (ref.[1]). Probably due to the lack of a quantitative analysis, there was at the time of this earlier publication however serious doubt whether Savov's so called "Rotopter" could be applied to a practical helicopter.

A quantitative analysis - which also indicates that several refinements and modifications of the basic principle are necessary - shows that the system may be perfectly feasible in practice. The resulting helicopter configuration is here called an "ornicopter" (helicopter + ornithopter).

The following questions will be addressed and quantitatively treated in the present paper:

1. Can the rotor indeed be driven by flap forcing to a sufficiently high angular speed as needed by a practical helicopter ?
2. How large is the required power compared with conventional shaft drive ?
3. How large are the necessary flapping angles, and are vibration problems to be expected due to the additional flapping ?
4. Are the cyclic control and trim characteristics influenced by the flap forcing ?
5. Is a mechanism feasible which can, although eliminating the reaction torque, at the same time provide powerful yaw control?

3 Principles of bird flight

Consider a symmetric aerofoil (fig. 2 to 4) moving through the air along an undulating path, i.e. the superposition of forward speed and a flapping motion. It is seen from fig.2 that, averaged over a flapping cycle, an efficient aerofoil will experience a positive propulsive force due to the periodic forward tilt of the lift vector. To sustain the flapping motion power will be needed, since the lift is always opposing the vertical motion. On the average there will be no upward force.

Setting the aerofoil at a positive incidence and keeping the incidence constant whilst performing the flapping motion may obtain propulsion as well as a net upward force. In the case of small incidence- and path angles the total lift will be the superposition of a constant value and the lift variations of fig.2, as sketched in fig.3.

Birds do apply incidence variations when flapping their wing, as may be seen from the experimental data shown in fig.1. A limiting case is shown in fig.4, where the incidence variations are so large that the angle of attack is kept constant. In this case the lift will be constant, but averaged over one cycle there will be no net thrust.

Birds apply incidence variations in magnitude somewhere between the situations of fig.3 and 4 depending on the flight phase: during accelerating flight their wing motion is closer to the situation of fig.3 than during steady flight. Actually, the flight of birds is a lot more sophisticated and complex than sketched above. The flap down occurs in general at a larger horizontal speed (using the lead-lag degree of freedom) than the flap up motion. Furthermore, different parts of the wing describe different motions, and often the aerofoil shape is changing during the motion.

However, the above sketched principles suffice to explain the rotor system without reaction torque. Looking at the flow angles of the blade element of a helicopter rotor blade, it is seen from fig.5 that a strong downward flapping motion leads to a forward tilt of the lift vector, thus giving rise to a propulsion force. Of course, upward flapping has the opposite effect. We could however try to approximate the situation of fig.3 , in order to obtain a net propulsive force during one blade revolution. To achieve this, the flapping motion will have to be excited by some kind of mechanism which supplies power to the rotor blade, like the flapping muscles of birds.

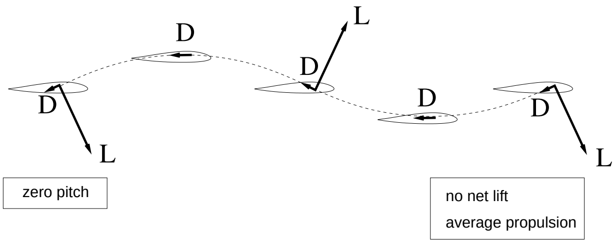

Figure 2: Flapping wing with a pitch angle equal to zero.

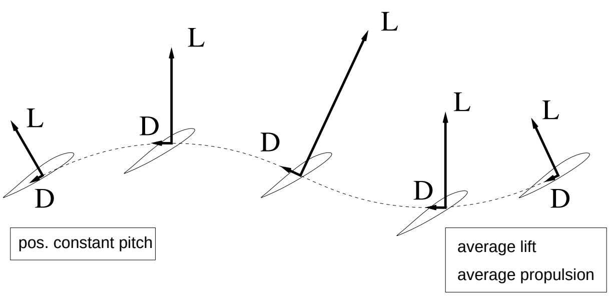

Figure 3: Flapping wing with a constant pitch angle.

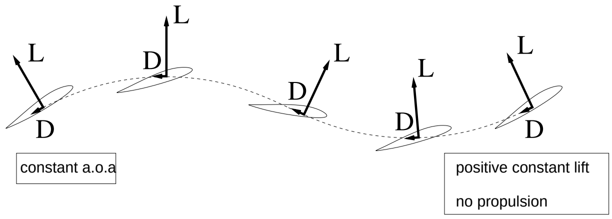

Figure 4: Flapping wing with a constant angle of attack.

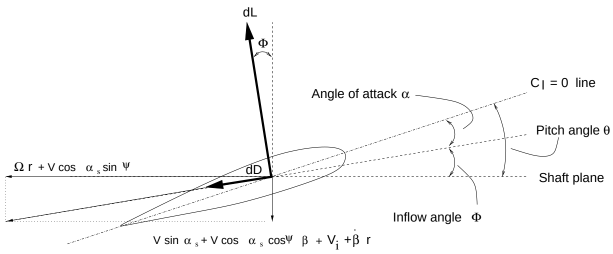

Figure 5: Velocity diagram of blade element.

4 Principle of a flap forcing mechanism

The principle of a flap forcing mechanism for a two-bladed helicopter rotor is shown in fig.6. For clarity the conventional swash plate mechanism is not drawn, although it is also present. The flap forcing mechanism consists of a push-pull rod through the center of the hollow rotor shaft, the rod co-rotating with the shaft and the rotor. The once-per-rev push-pull motion is converted to a flapping moment on both the blades. Note that there is an essential difference between the flap forcing and the application of cyclic pitch by a swash plate: both are periodical with a frequency of once-per-rev, but cyclic pitch is asymmetrically applied (the magnitude is equal but the direction is different for the two blades), whereas the flap forcing is symmetrical.

The once-per-rev push-pull motion is derived from a mechanism analogous to hydraulic pumps: a radial extension of the push-pull rod is forced to rotate in an inclined, stationary plane. The driving power is derived from the main engine, via the main rotor shaft. If the flap forcing is just sufficient, no direct shaft power needs to be transmitted to the rotor to maintain its speed, and all the power is then available for the driving mechanism of the push-pull motion. In this situation there is an average torque exerted on the inclined

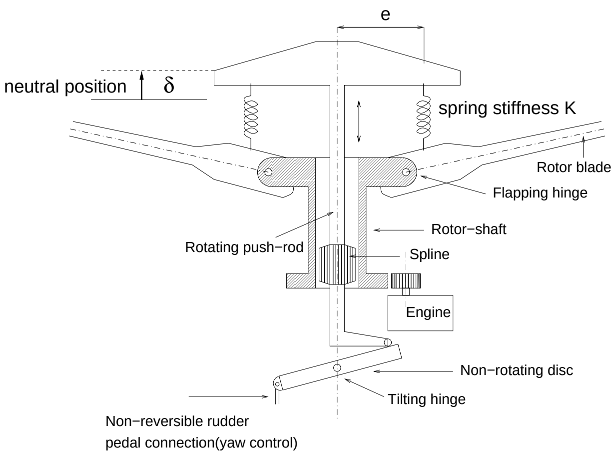

Figure 6: Principle of a possible mechanism for flap forcing.

plane equal in magnitude but opposite in direction to the torque exerted by the main engine on its mountings (ref.[2]). Hence the absence of reaction torque on the helicopter as a whole.

The pilot regulates the inclination of the plane. The amplitude of the push-pull motion determines the division of the total engine power between shaft- and flap power. It thus determines whether there is some reaction torque left and in which direction it acts. Therefore a powerful yaw control for the helicopter as a whole may be realized by varying the plane's inclination.

5 Power required

In steady forward flight the inflow angle ϕ of a blade element is, according to fig.3:

$$\phi \approx \tan\phi = \frac{V\sin\alpha_s + v_i + \dot{\beta}r + V\cos\alpha_s\cos\psi\cdot\beta}{\Omega r + V\cos\alpha_s\sin\psi} \quad (1)$$

The inflow angle determines the magnitude of the lift component opposing the blade rotation, so that the shaft power required to maintain steady rotation is:

$$P_{sh} = P_p + \frac{1}{2\pi}\int_0^{2\pi} d\psi \int_0^R dL\cdot\sin\phi\cdot\Omega r \quad (2)$$

where the power P_p has been added, the so called "profile power" associated with profile drag.

By some algebraic manipulation and substitution of eq.1the latter expression may be interpreted as follows:

$$\begin{aligned} P_{sh} &= P_p + \frac{1}{2\pi}\int_0^{2\pi} d\psi \int_0^R dL\sin\phi\,(\Omega r + V\cos\alpha_s\sin\psi) \\ &\quad - V\cos\alpha_s\frac{1}{2\pi}\int_0^{2\pi} d\psi\int_0^R dL\sin\phi\sin\psi \\ &= P_p + T(V\sin\alpha_s + v_i) + \frac{1}{2\pi}\int_0^{2\pi}\dot{\beta}d\psi\int_0^R dL\cdot r \\ &\quad - V\cos\alpha_s\cdot\frac{1}{2\pi}\int_0^{2\pi} d\psi\int_0^R dL\,(-\beta\cos\psi + \phi\sin\psi) \\ &= P_p + T(V\sin\alpha_s + v_i) + H_iV + P_\beta \end{aligned} \quad (3)$$

where the symbol P_β stands for:

$$P_\beta = +\frac{1}{2\pi}\int_0^{2\pi}\dot{\beta}d\psi\int_0^R dL\cdot r \quad (4)$$

The equilibrium of forces in steady flight is (see fig.7):

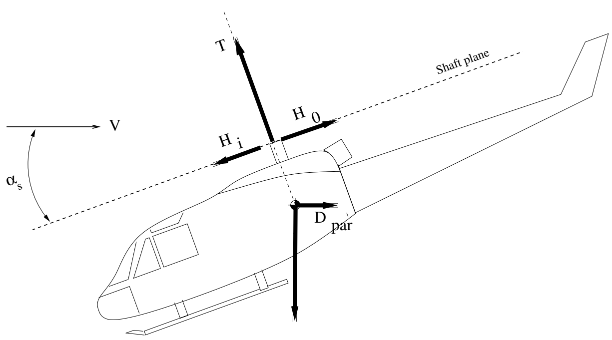

Figure 7: Equilibrium of forces in steady horizontal flight.

$$T\sin\alpha_s + (H_i - H_0)\cos\alpha_s - D_{par} = 0 \quad (5)$$

where H_i is the in-plane force associated with the lift on the blade elements, and H_0 is the in-plane force associated with the drag. D_{par} is the parasite drag. Substituting eq.5 into eq.3 yields:

$$P_{sh} = P_p + H_0V + P_i + P_{par} - P_\beta \quad (6)$$

where P_i is the induced power, and P_{par} is the parasite power.

Eq.6 is the usual expression for the total shaft power required in forward flight, except for the term P_β. The latter will be analyzed more carefully below.

6 General flapping equation

Let us for simplicity assume a simple, centrally hinged rotor blade (fig.8), where a mechanical flapping moment is applied to the root. The equation of motion is:

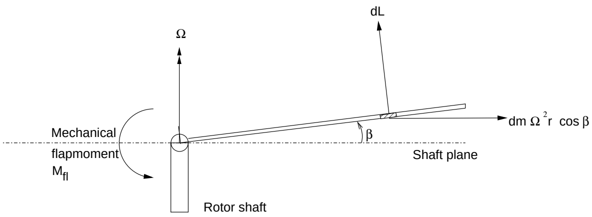

Figure 8: Moments on flapping blade.

$$\ddot{\beta} + \Omega^2 \beta = \frac{M_a}{I} + \frac{M_{fl}}{I} \tag{7}$$

where M_{fl} stands for the mechanical flapping moment, and M_a is the aerodynamic flapping moment. Using the flat disc approximation for the tip path plane:

$$\beta = a_0 - a_1 \cos\psi - b_1 \sin\psi \tag{8}$$

it follows that

$$M_a = -M_{fl} + a_0 \Omega^2 I \tag{9}$$

In eq.4 for P_β it is recognized that:

$$\int_0^R dL \cdot r = M_a \tag{10}$$

so that it finally follows that

$$P_\beta = -\frac{1}{2\pi} \int_0^{2\pi} \dot{\beta} M_{fl} d\psi \tag{11}$$

The physical interpretation of this expression is clear. The product $\dot{\beta} M_{fl}$ is the instantaneous power exerted by the flap forcing mechanism on the blade. Denoting the cycle averaged power by P_{fl} it may thus be concluded that

$$P_\beta = -P_{fl} \tag{12}$$

7 Decoupling the cyclic control from the yaw control

Returning now to the power equation

$$P_{sh} = P_p + H_0 V + P_i + P_{par} - P_{fl} \tag{13}$$

it is seen that one can cancel the required shaft power by making P_{fl} sufficiently large. The necessary driving power is then entirely supplied to the rotor by the flap forcing mechanism. In this case there will be no reaction torque. The power needed by the flap forcing mechanism to maintain rotor speed is equal to the shaft power it replaces.

It should be realized that the flapping angle occurring in eq.11 is the total flapping angle, which not only arises from the mechanical flap forcing, but also from cyclic control and the effect of the flight speed. At first sight one would conclude that the required moment M_{fl} would depend on the cyclic control position. In other words, one would expect a strong, and probably unacceptable cross coupling between the cyclic and yaw control.

This is not the case however, as will be shown now. Considering fig.6 the mechanically applied flapping moment M_{fl} may be expressed like:

$$M_{fl} = K (\delta - \beta \cdot e) e \tag{14}$$

where δ is the upward displacement of the push-pull mechanism, and e is the offset distance of the spring, with spring constant K.

Substituting eq.14 into eq.11 it follows under steady flight conditions (i.e. the flapping angle β

returns after one revolution to its initial value):

$$P_{fl} = Ke \cdot \frac{1}{2\pi} \int_0^{2\pi} \dot{\beta} \delta d\psi \tag{15}$$

Using linear aerodynamics the flapping equation reads after writing out the aerodynamic flapping moment M_a:

$$\beta'' + f_1(\psi,\mu)\beta' + f_2(\psi,\mu,K)\beta = f_3(\psi,\mu)\theta(\psi) + f_4(\psi,\mu)(\lambda_c + \lambda_i) + \frac{Ke}{\Omega^2 I}\delta(\psi) \tag{16}$$

where the symbols β' and β'' denote differentiations w.r.t. the azimuth angle ψ.

Eq.16 is a linear, second order differential equation with variable coefficients. There are three forcing terms on the right hand side of the equation. They are respectively associated with the cyclic control, the inflow state of the rotor (which in turn depends on forward velocity), and on the mechanical flap forcing. Thanks to the linearity of the equation the solution will be the sum of three contributions:

$$\beta = \beta_\delta + \beta_\theta + \beta_\lambda \tag{17}$$

where β_δ is the flapping associated with the flap forcing, β_θ depends on the cyclic control, and β_λ on the flight speed.

If we consider a second blade as sketched in fig.6, the flapping equation will be identical except for the flap forcing term, which will have a negative sign. Fig.9 shows the tip path planes of the two blades. Cyclic control and flight speed will result in both blades rotating in the same tip path plane. The mechanical flap forcing is such that the tip path plane associated with it is antisymmetric for the two blades. Therefore, if we sum the flapping power of the two blades, the following expression is obtained:

$$P_{fl_{tot}} = N \cdot Ke \cdot \frac{1}{2\pi} \int_0^{2\pi} \dot{\beta}_\delta \delta d\psi \tag{18}$$

where N is the number of blades, in this case N=2.

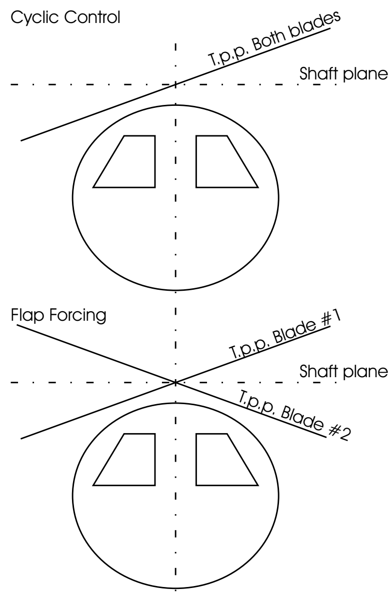

Figure 9: Split of tip path plane due to flap forcing.

The important conclusion from this expression is, that the total flapping power does not depend on the position of the cyclic control. Looking at fig.9 it is seen that the flap forcing does not cause any additional forces or moments on the fuselage, because of the antisymmetric motion of the blades as far as the flap forcing is concerned. This means that the required cyclic control is not influenced by the flap forcing system or by the yaw control. In other words, there is a complete mutual decoupling of the cyclic and yaw control. Trim curves and the response to cyclic input are both entirely conventional (refs. [3] and [4]).

8 Magnitude of the required flapping

Assuming linear aerodynamics, the aerodynamic flapping moment M_a in eq.16 may be written out analytically. If we now consider only the flapping motion associated with the flap forcing, the flapping equation reads:

$$\beta'' + \frac{\gamma}{8}\left(1 + \frac{4}{3}\mu \sin\psi\right)\beta' + \left[1 + \frac{Ke^2}{\Omega^2 I} + \frac{\gamma}{6}\mu\cos\psi\left(1 + \frac{3}{2}\mu\sin\psi\right)\right]\beta = \frac{Ke^2}{\Omega^2 I}\frac{\delta_c}{e}\cos\psi \tag{19}$$

where $\delta(\psi)$ has been assumed to be $\delta = \delta_c \cos\psi$. Using once again the flat disc approximation eq.8 an approximate solution of the flapping equation 19 can be obtained. The flapping coefficients a_1 and b_1 are given by:

$$a_1 = -\frac{\kappa^2 \cdot \left(\frac{\delta_c}{e}\right)}{\kappa^2 + \left(\frac{\gamma}{8}\right)^2} \tag{20}$$

$$b_1 = -\frac{\kappa \cdot \left(\frac{\delta_c}{e}\right) \frac{\gamma}{8} \left(1 - \frac{1}{2}\mu^2\right)}{\kappa^2 + \left(\frac{\gamma}{8}\right)^2} \tag{21}$$

in which:

$$\kappa = \frac{Ke^2}{\Omega^2 I} \tag{22}$$

Next the integral for the total flapping power eq.18 can be solved:

$$\begin{aligned} P_{fl_{tot}} &= N \cdot Ke \cdot \frac{1}{2\pi} \int_0^{2\pi} \dot{\beta}_\delta \delta d\psi \\ &= \frac{\Omega^3 I}{\pi} \int_0^{2\pi} [a_1 \sin\psi - b_1 \cos\psi] \left(\kappa \frac{\delta_c}{e}\right) \cos\psi d\psi \\ &= -\Omega^3 I \left(\kappa \frac{\delta_c}{e}\right) b_1 \\ &= +\Omega^3 I \frac{\kappa^2 \frac{\gamma}{8} \left(1 - \frac{1}{2}\mu^2\right)}{\kappa^2 + \left(\frac{\gamma}{8}\right)^2} \cdot \left(\frac{\delta_c}{e}\right)^2 \end{aligned} \tag{23}$$

Eq.13 provides the required value of the flapping power in order to achieve the situation of no reaction torque. In combination with 23, 20 and 21 the amplitude of the required flapping may be calculated. Fig.10 shows an example for a typical light helicopter. It has been assumed here that the springs in the mechanism of fig.6 are soft, so that the control characteristics of a semi-rigid rotorsystem are obtained. It is seen that the flapping angles needed are relatively modest.

9 Vertical vibrations

In order to estimate the magnitude of the vertical vibrations of the fuselage the dynamic model shown in fig.11 was used (ref.[5]), based on an alternative but equivalent form of the flap forcing mechanism (fig.12). For simplicity we consider only hovering, and a blade excitation of the form $\delta = \delta_c \cos\psi$.

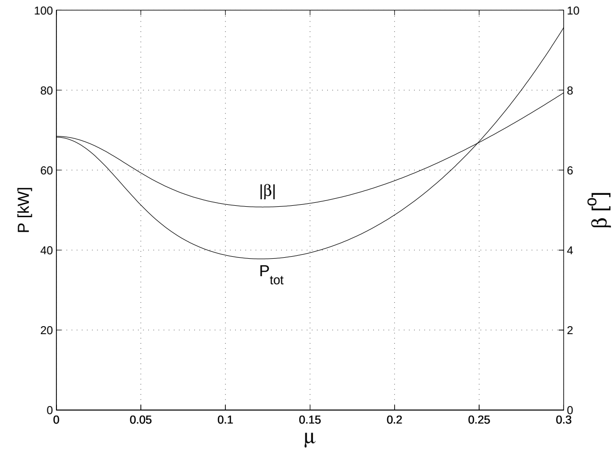

Figure 10: Flap forcing power required and flapping angle for typical light helicopter.

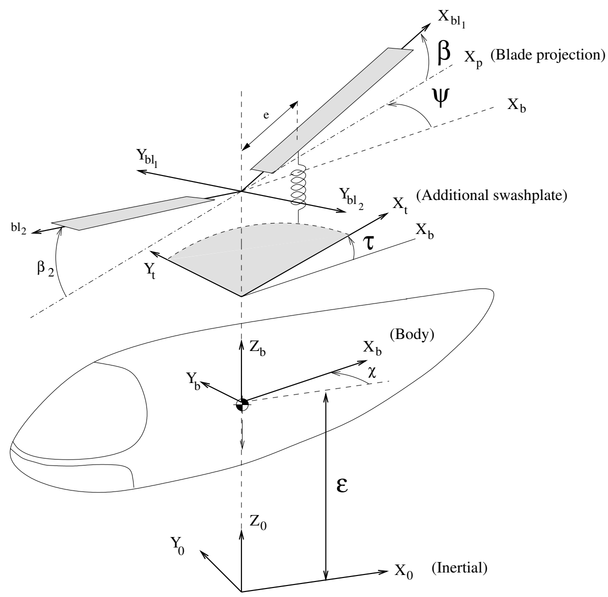

Figure 11: Dynamic model for analyzing vertical vibrations.

The resulting equation of motion for the vertical movement ε is:

$$(M+m)\,\ddot{\varepsilon} + \frac{\gamma}{4}\frac{\Omega I}{R^2}\dot{\varepsilon} = -S\ddot{\beta} - \frac{\gamma}{6}\frac{\Omega I}{R}\dot{\beta} \tag{24}$$

where S is the static moment of the blade, and M is the fuselage mass.

From this equation it appears that the physical effects which cause a vertical vibration are the lift variations due to flapping, as well as the vertical acceleration of the blade center of gravity. It is interesting to note that the rotor drive system, which exerts oscillating vertical forces on the blades (and by reaction on the fuselage) does not enter into the equation of motion 24. There is in fact a closed loop for these vertical forces. They find a direct reaction through the flapping hinges and the main rotor shaft.

Eliminating the vertical vibrations at the source is possible in the case of a four bladed rotor. Forcing an opposed flapping motion upon the two sets of blades (each set consisting of two blades coupled in the way shown in fig.6) the excitations in the equation of motion 24 cancel each other (see fig.13).

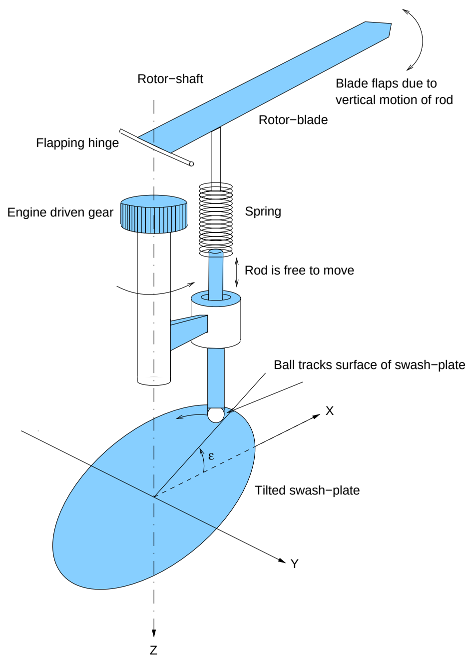

Figure 12: Alternative flap forcing mechanism.

10 Alternative rotor configurations

In order to solve the problem of vertical vibrations, there is an alternative rotor configuration possible. This consists of two sets of teetering rotors mounted perpendicular to each other on the same shaft (fig.13). Once again the excitation of the vertical vibrations is then completely cancelled. In this case the flapping motion should be

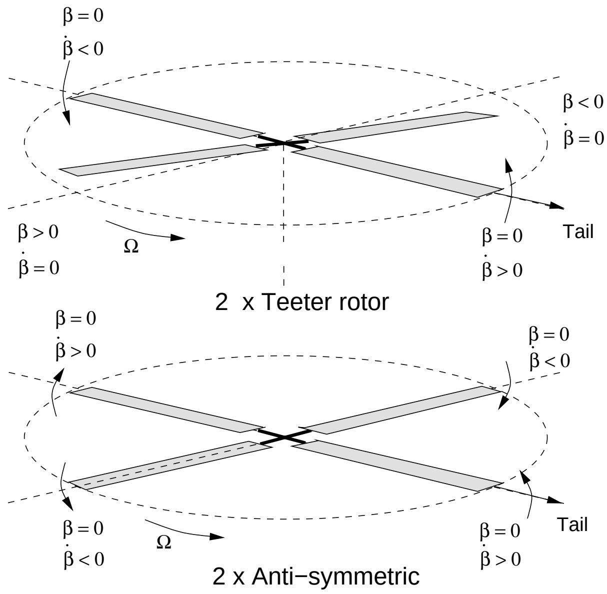

Figure 13: Four-bladed configurations to eliminate vertical vibrations: a) Double teeter, b)Twice two-bladed configuration according to fig.6, in opposite phase.

forced in the way shown in fig.14 , so that the two tip path planes are anti-symmetrical, as in fig.9.

11 Vibrations around the yaw axis

The dynamic model is shown in fig.14. The fuselage now has a degree of freedom χ around the top axis. Again we consider the hover. As a first step a one-bladed configuration is considered where the blade forcing is assumed to be $\delta = \delta_c \cos\psi$.

Writing out the equation of motion for the yaw it is found:

$$\frac{I_f}{I}\chi'' + \kappa\left(\frac{\delta_c}{e}\cos\psi - \beta\right)\frac{\delta_c}{e}\sin\psi = -\frac{Q_{eng}}{\Omega^2 I} \quad (25)$$

In this equation Q_{eng} is the engine torque, assumed to be constant. The flapping angle is expressed like $\beta = -a_1\cos\psi - b_1\sin\psi$.

The physical interpretation of the second term is given in ref.[2]. It is the instantaneous torque around the top axis which originates from the fact that the spring in fig.6 presses on the inclined

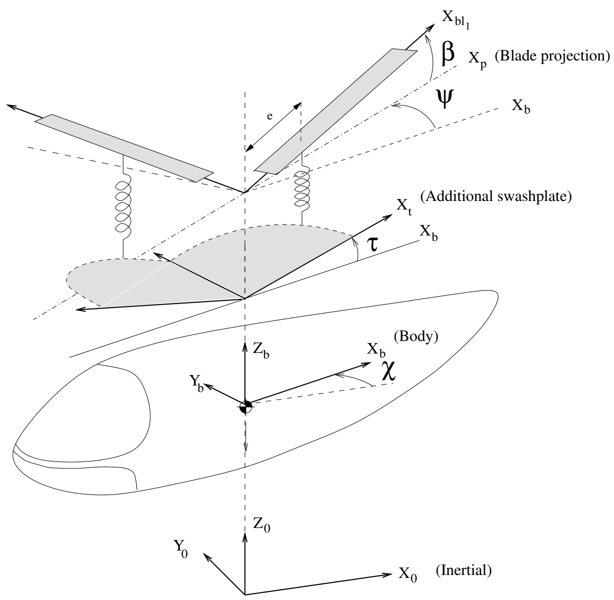

Figure 14: Dynamic model for analyzing yaw vibrations, double teeter.

plane and so gives rise to a horizontal force whose work line does not pass through the top axis.

It must be remarked that it was assumed during the derivation of eq.25 that there is a soft torsional spring between the engine and the drive system sketched in fig.12. The blade is allowed to undergo rotational accelerations, and is not restricted to turn at a constant angular speed. In the case of a torsionally stiff connection between the rotor head and the engine the results would be different.

In the case of the double teeter configuration we have to add a second rotor blade perpendicular to the first rotor blade. For the second blade the same expression holds, with $\psi_2 = \psi + \pi/2$, $\delta_{c_2} = -\delta_c$, $a_{1_2} = -a_1$ and $b_{1_2} = -b_1$. Summing the contributions of the two blades, we find the equation of motion

$$\frac{I_f}{I}\chi'' + \kappa\left(\frac{\delta_c}{e}\right) b_1 = -\frac{Q_{eng}}{\Omega^2 I} \tag{26}$$

From the flapping equation the value of b_1 was found (see eq.21), from which it appears for the case of hovering:

$$\frac{I_f}{I}\chi'' - \frac{P_{fl_{tot}}}{\Omega^3 I} = -\frac{Q_{eng}}{\Omega^2 I} \tag{27}$$

In other words $\chi" = 0$ in the case of yaw equilibrium. The vibrations around the top axis not only are cancelled in a cycle-averaged way, but even instantaneously in the case of the double-teeter configuration.

The four-bladed anti-symmetric configuration shown in fig.13 displays a different behavior. In this case it is found that a strong two-per-rev torque oscillation results, which would require a vibration isolation system.

12 Roll and pitch vibrations

For the analysis of roll- or pitch vibrations again a dynamic model like fig.14 has been used. In this case the vertical degree of freedom ε has been replaced by a roll degree of freedom φ (see fig.15).

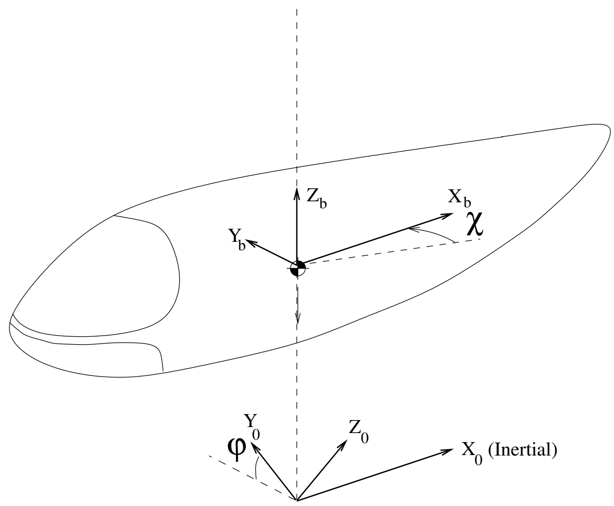

Figure 15: Dynamic model for roll and pitch.

The dynamic equation for the single blade configuration is:

$$\begin{aligned} & I\ddot{\phi}\sin\psi + I_f\ddot{\phi} + \Omega I\dot{\phi}\sin 2\psi + \frac{\gamma}{8}\Omega I\dot{\phi}\sin^2\psi \\ & = I\ddot{\beta}\sin\psi + \Omega^2 I\beta\sin\psi + \frac{\gamma}{8}\Omega I\dot{\beta}\sin\psi \\ & \cong M_{fl}\sin 2\psi \end{aligned} \tag{28}$$

If the springs in the push-pull mechanism of the rotor drive are soft so that the eigenfrequency of the blade flapping equation is almost equal to Ω, a clear physical interpretation is possible. Under such conditions the r.h.s. of equation 28 may be approximated by the flap forcing moment $Ke\delta$, multiplied by $\sin\psi$. In other words, the reaction of the flap forcing moment acts on the fuselage. Vectorially, the flap forcing moment rotates with the blade.

In the case of the antisymmetric blade configuration of fig.13 no resulting roll- or pitch moments will be carried over to the fuselage. In the case of the double teeter configuration there will be a resulting 2-P excitation of the fuselage. In the latter case of the double teeter a dynamic vibration absorber may be needed in order to counteract the vibrations.

13 Conclusions

Using a quantitative analysis it has been shown that active flap excitation can drive a helicopter rotor, so that there is no need to supply shaft power. In this way a single rotor without reaction torque may be realized, which eliminates the need for anti-torque devices. The actual mechanism to effect the flap forcing can be arranged in such a way that nevertheless yaw control is available. The power to drive the rotor by flap forcing is equal to the shaft power in a conventional layout. The additional flapping angles are comparable in magnitude to the usual flapping angles. The flap forcing mechanism can be arranged such that there is no interference with the normal cyclic control. Cyclic control does not affect the drive function, nor is the force and moment balance influenced by the flap forcing. Therefore no undesirable cross coupling between roll- and pitch control on the one hand and yaw control on the other occurs. A socalled "ornicopter" will display a conventional trim behavior, comparable with that of a soft semi-rigid rotor system. In the case of a four-bladed configuration it is possible to eliminate vertical vibrations of the fuselage. A double-teeter layout seems an attractive arrangement because of its relative simplicity and since no torque fluctuations around the top axis occur. Dynamic vibration absorbers will however be necessary to reduce roll- and pitch vibrations of the fuselage.

The characteristics found so far seem to indicate that ornicopters could be feasible in practice. Further investigation is worthwhile, e.g. into the matter of blade stresses, mechanical complexity and overall costs.

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