The alternative to finite-element analysis is the modified Galerkin approach in CAMRAD. The advantage of the latter approach is that it resides in the same code that will be used for the aerodynamic analysis. The disadvantage is that the method does not ordinarily generate the matrices M, C, and K which are needed for the analytical derivatives (e.g., eq. (19)). Thus, the modified Galerkin approach may require the use of finite difference derivatives. This was done in reference 7 without any ill effects. Nevertheless, studies are underway to find ways to generate equivalent M, C, and K matrices based on the modified Galerkin method and use these in the calculations of analytical derivatives.

ROTOR BLADE STRUCTURAL DESIGN

Mark W. Nixon

In this section the structural design of rotor blades is discussed. The various topics associated with the structural design include constraints, load cases, and analyses.

Design Constraints

The constraints associated with traditional structural design can be categorized as aerodynamic, autorotation, buckling, frequency, and material strength. As discussed in reference 10, some of these constraints are based on maintaining characteristics required by other disciplines involved in the integrated optimization. Constraints associated with aerodynamics, autorotation and frequency are not addressed in this section, since they are addressed in other sections of the paper.

Of the remaining structural constraints, the most important is the material strength constraint. All stresses in the blade structure must be less than the design allowable stress of the material for all load cases. To account for stress interactions, a failure criterion such as Tsai-Hill (ref. 25) is calculated based on the material limit allowable stresses. The governing equation is

$$\bar{R} = \sqrt{\frac{\sigma_1^2}{X^2} - \frac{\sigma_1\sigma_2}{X^2} + \frac{\sigma_2^2}{Y^2} + \frac{\sigma_{12}^2}{S^2}} \tag{20}$$

The quantity $(1 - \bar{R})$ is a margin of safety which must be greater than zero at all points of the blade. This constraint must be evaluated for several load cases which are discussed in detail in the Load Cases section.

A constraint is also applied for buckling of the blade spars. Buckling is not likely to occur when the blade system is rotating because of the high tensile loads induced by centrifugal forces. However, there are load conditions, discussed later in the Load Cases section, in which the blade is not rotating. In the absence of centrifugal forces, it is possible that buckling occurs at a stress below the allowable static stress. The buckling constraint is violated if the compressive stress due to bending exceeds a critical value which is given by reference 26 as

$$\sigma_c = K_c \frac{Et^2}{W^2} \tag{21}$$

for a D-shaped spar. In equation (21), E is Young's modulus, t is spar thickness, W is spar width, and K_c is a constant dependent on cross sectional geometry.

Load Cases

The static load cases used herein for structural optimization are outlined in reference 26, and are discussed in detail in reference 27. They are described below in terms of flapwise, inplane, torsional, centrifugal, and non-flight loads. The flapwise, inplane, torsional, and centrifugal loads are applied simultaneously. The non-flight loads are a separate case applied to the non-rotating cantilevered blade. The method of calculating the load magnitudes and distribution for each case are covered in this section.

Flapwise loads.- Flapwise load magnitudes are defined as a function of load factors, N_z, and the structural design gross weight of the total helicopter system, SDGW. The load factors are applied to account for the load increases which occur in maneuvers as well as appropriate factors of safety. The maneuver loads generally cannot be directly predicted with sufficient accuracy using current analysis techniques. The critical flapwise load factors used under current structural design requirements range from -0.5 to +3.5 for most military helicopters. The total flapwise load is equal to N_z times the structural design gross weight of the system. Thus, the magnitude of the flapwise load, L_f, carried by one blade in a rotor system of N blades is given by

$$L_f = (N_z)(SDGW)/N \tag{22}$$

Distribution of the load, which is a function of azimuthal position, should be representative of actual airloads the blade produces in steady level forward flight. The airloads include both steady and oscillatory parts, and are scaled proportionally at each spanwise segment until the total load (the sum of the load on each segment) equals the required load, L_f. The steady and oscillatory level flight blade airloads are obtained from an aerodynamic analysis using CAMRAD (ref. 14). The load distributions associated with several azimuthal positions will be considered. This increases the likelihood that all critical load distributions have been identified.

Inplane loads.- The inplane loads are based on two cases of shaft torque transmission from the powerplant. One case emanates from a power increase with subsequent rotor acceleration. Here, a shaft torque is transmitted through the hub creating an inplane moment at the blade root. The limit root inplane moment, M_E, is given by reference 26 as

$$N_E = \frac{1.5M_T}{N - 1} \tag{23}$$

where M_T is the torque developed at the military power rating of the powerplant. The second case requires that twice the maximum braking torque be equally transmitted to all blades. The root moment for both cases is balanced by an inertial force distribution developed along the blade span such that

$$M_E = \sum_{i=1}^{n} m_i r_i^2 \dot{\Omega} \tag{24}$$

where i refers to the ith blade segment of the beam model, m is segment mass and r is radial distance. After solving for $\dot{\Omega}$, the inplane inertial loads can be written as

$$q_i(r) = \frac{m_i r_i \dot{\Omega}}{l_i} \tag{25}$$

where l_i is the length of the i-th segment.

Torsional loads.- There are two basic contributions to the static torsional loads of a rotor blade. The first is due to the aerodynamic pitching moments on the airfoil sections which are obtained from the aerodynamic analysis. The second torsional load contribution is due to the inertial moments created by the centrifugal forces. Because rotor blades generally have a built-in twist, there will always be a part of the blade in which the inertial moments can be significant. The torsional loads produced here are proportional to centrifugal force, root angle of attack, and rotor twist such that

$$T_{PM_i} = I_{\theta_i} \Omega^2 \theta_i \tag{26}$$

where I_θ is the moment of inertia about the blade axis, and θ is the blade pitch angle.

Centrifugal loads.- Axial and flapwise components of centrifugal force in the ith blade segment $(CF_i)_a$ and $(CF_i)_f$, are shown in figure 5. In the inplane load case shown in figure 6, an inplane distributed inertial load, q(r), creates a lag condition. Lead-lag rigid body displacements resulting from the inertial load do not create large opposing centrifugal force components because the centrifugal force vector acts nearly along the c.g. axis of the blade. The magnitude and distribution of the centrifugal load is governed by the equation

$$CF_i = m_i r_i \Omega^2 \tag{27}$$

where i refers to an individual blade segment of the beam model.

Non-flight loads.- The last load case covers aspects of non-flight loads. Reference 26 requires that an articulated rotor blade be designed for a static load equal to its weight multiplied by a limit load factor of 4.67. Reference 27 indicates that this load case can be used to cover other adverse conditions such as ground handling, stop-banging, turning the rotor at low speed in a strong wind, and the condition in which a helicopter with an untethered rotor is in the vicinity of an operating helicopter. For the non-flight load case, the blade is assumed to be cantilevered at the blade stops, and under no rotational effects. The non-flight load case is used to check for buckling of the blade spars.

Blade Structural Analyses

A choice must be made regarding the type of analytical model to use in the structural analysis. There are two analysis procedures which can give the detailed ply-by-ply stresses required to assess material strength margins of safety. One procedure is completely finite element based and uses a two-dimensional finite element model. The other procedure is a combination of a beam analysis (finite element or not) applied to a planform model and a laminate analysis applied to one or more cross section models. The two-dimensional finite element procedure requires significantly more computation time than the combination procedure. Time efficiency is very important when using a discipline-integrated optimization procedure because hundreds (possibly thousands) of analysis iterations are necessary. In preliminary design the accuracy of a combination of beam and laminate analyses is sufficient. Further, the superior efficiency with respect to computational time makes the combination procedure more desirable than the two-dimensional finite element procedure.

The combined beam and laminate analyses procedure requires two types of blade models: a beam model and a cross-section model. The beam model consists of a series of beam segments connected at spanwise grid points. Each segment contains equivalent beam properties such as the stiffnesses and masses. These properties are constant along a single beam segment, but may vary between segments, thus forming a step function of beam property distributions along the blade span. Displacements (translational and rotational) and beam forces (shears and moments) resulting from the applied loads are computed at the grid points.

A cross section model is a representation of the internal blade structure which is composed of several components. These components generally consist of one or more spars, a leading edge weight, an aft honeycomb or balsa core, and a skin. The cross section models serve two purposes. First, they are used to calculate the equivalent beam properties of the beam segments. Thus, there will be a different cross section model corresponding to each unique beam segment. Secondly, the cross section models are used to calculate stresses resulting from the forces associated with each beam segment. The stresses are then used in a laminate analysis to determine the margins of safety at various points in the cross section.

ACOUSTIC DESIGN CONSIDERATIONS

Ruth M. Martin

Review of Rotor Acoustic Sources

The acoustic signal from a helicopter rotor arises from several very complicated sources due to the aerodynamic loading of the blades, the interaction of the rotor with its wake, and the physical process of the blades moving through air. The