Sixth European Rotorcraft and Powered Lift Aircraft Forum

Paper No. 52

AN INTEGRATED APPROACH TO EFFECTIVE ANALYTICAL SUPPORT OF HELICOPTER DESIGN AND DEVELOPMENT

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Abstract

There are many ways of providing analytical support to an engineering project, but comparatively few methods ever survive to become established. The use of an authentic approach coupled with sufficient effort should ensure eventual success by one important criterion, good correlation with test data. Failure usually involves time and cost. The method outlined is an attempt to afford adequate design support, providing a means to predict what could happen well before the event; before critical decisions have to be made. It is, moreover, an attempt to achieve such a desirable objective at moderate cost. These two go hand-in-hand with the principle of integration which implies a basic unity in methods of analysis, modelling and programming. The approach is illustrated in application to a conventional helicopter with emphasis on the particular problem of effecting recovery following total power failure, and the feasibility of achieving a safe landing.

1. Introduction

The method to be described owes its origin to the persistent problem of landing a helicopter safely following total loss of power at low height. There are a number of ways to approach this problem analytically, but it was decided for a variety of reasons to go the way of flight simulation. Power failures are unpredictable, and could involve violent maneuver. It was deemed necessary, at the outset, to provide a model with adequate scope; one capable of simulating flight realistically, over, and even beyond the usable flight envelope. There are however economic constraints. When affecting a recovery, the pilot tends to react instinctively but is also faced with making a series of critical decisions within a time span of seconds. To determine what margin of error is tolerable involves numerous repetitions of a procedure, with variations. It was clear that achievement of economy would entail rapid execution. Capability of operation within a real time frame work was considered, and as both a desirable and feasible goal was adopted as a criterion of satisfactory performance.

The prototype model was not general but tailored to the OH6A helicopter for which suitable flight test data was readily available. The model was verified in the first instance by comparison of computed with measured trimmed performance points. Concurrently, sets of stability derivatives were computed, one set per point, providing a means of assessing model validity and suitability for controlled flight. The simulated execution of any formal maneuver requires a command structure, and a means of transmitting the commands. In effect it is necessary to simulate a human pilot. How successful the simulation was can be inferred from Figure 1 which illustrates the first attempt at correlation with an actual flight maneuver. The methodology was allowed to evolve, basing the fundamental decision framework on information gleaned from flight records and interviews with experienced test pilots. The end product emerged in two parts. The first is hybrid containing logical decisions, commands, and such pilot actions as are best described by adaptive control laws. The second is a model of a stabilizing system. Whether it represents an actual system or a pilot functioning as such, it constitutes an essential link in the computational cycle. Both parts process commands, originating in the first. Output is summed in the second part for passage to the vehicle model. For man-in-the-loop applications; the first part is replaceable by an interface module.

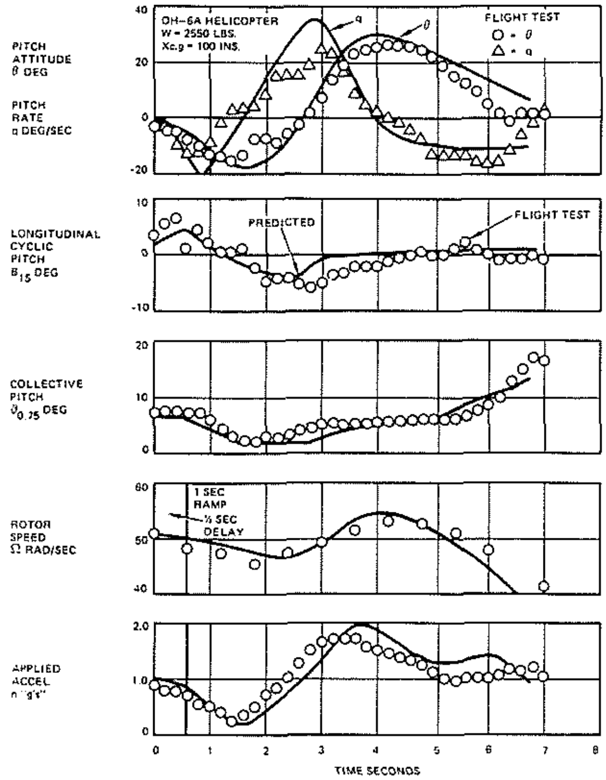

Figure 1. Correlation with Flight Test Data for an OH-6A During Emergency Landing from 50 Knots Level Flight and 250 Ft ISA

The basic method is now developed to the stage where a variety of maneuvers and helicopters have been treated. Though all fall into a single main rotor/tail rotor category, blade retention systems differ widely and it has been necessary to cater for teetering, articulated and spring constrained systems. To this extent, treatment has been general.

2. Vehicle Model

It was never the intention that the vehicle model be fully general. Each particular helicopter model is assembled from modules, one for each major vehicle component. Each individual module is tailored to the peculiarities of that component. Current applications are limited to performance and handling qualities with attention to failure modes. To this end certain features are common. Thus all models provide six body degrees-of-freedom plus a seventh for the propulsion train linking the two rotors, coupled with or decoupled from the engine system. All modules are powered by free turbine with one-degree-of-freedom per gas generator. Motion of N blades in a flapwise mode is treated in two pseudo degrees-of-freedom so that dynamic response can be simulated realistically. Tail rotor flapping is treated quasi-statically. Provision is made for the non-linear aerodynamic characteristics of both rotors and all lifting surfaces, including the fuselage. Aerodynamic interference is defined for main rotor to wing/body, main rotor to tail assembly, wing to horizontal tail and mutual interference between tail rotor and vertical tail. Other combinations have been considered. Perhaps the most attractive feature of the basic model is the ability when implemented by digital computer program on a suitable processing systems, to execute within a real time framework. This ability is conferred in part by the approach to the modelling of the main rotor, and the computation of main rotor hub forces.

2.1 Main Rotor Sub-Model

Each particular main rotor model is identified with a data array generated off-line from a master blade element model. The master could in turn be generated by a dynamic model yet higher in the hierarchy and would then be definable as a truncated series of normal model in vacuo. To date, the master has been defined analytically and restricted to a single mode describing blade flapwise displacement relative to the hub. The generating program computes motion with respect to a rotating frame of reference wherein a single representative blade is disposed at a specified collective pitch and exposed to a uniform incident airstream. Aerodynamic constraints are defined using a bank of non-linear section data. Blade motion is integrated step-by-step, from a quiescent state to cyclical equilibrium, as the frame rotates by discrete steps azimuthwise. Aerodynamic loads are integrated spanwise at each step. Concurrently, six component coefficients representing hub forces are computed progressively, and stored as functions of three describing parameters, collective pitch, advance ratio, axial flow ratio. The concepts are illustrated in Figures 2, 3 and 4. Representative thrust and torque coefficients are plotted in Figure 5. Each set of points defines a trimmed rotor configuration resolved in a swashplate oriented frame of reference. The three-dimensional arrays cover the entire flight envelope and beyond. Their interpretation as the dynamic performance of a rotor involves a series of transformations, associated analysis, and some ingenuity. Further comment is delayed. It is sufficient to say at this point that the use of a synthesized model speeds up the computations cycle significantly, and is contributory to the attainment of real time capability.

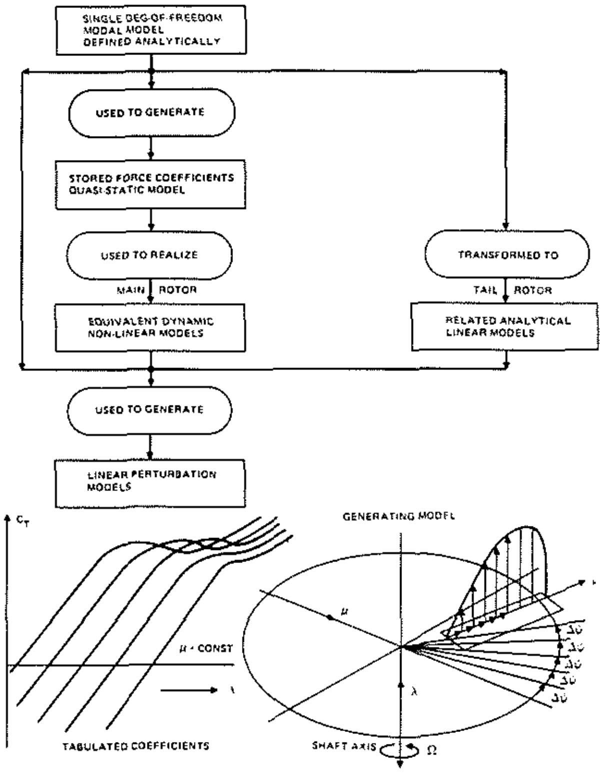

Figure 2. Prototype Module Structure for Rotors

2.2 Tail Rotor Sub-Model

The technique outlined in the previous subparagraph is general and can be applied to any type of rotor. So far it has not been tried on the tail rotor. Because of its proximity to the tail surfaces, there is strong mutual interference at least between the tail rotor and vertical surface. Then the tail rotor has to function over a much wider envelope, well into the region of negative thrust. With economy in mind, it was decided to transform the analytically derived blade element model into a closed form referred to stationary axes. Non-linearities were then admitted empirically and the appropriate parameters tuned by comparison with the equivalent N-blade element model.

2.3 Airframe Sub-Models

Modelling of the remaining components is conventional except insofar as provision must be made for omni-directional flight and aerodynamic interference from the rotors. Aerodynamic data must be defined over a 360 degree range; by synthesis where no reliable measured data is available. When dealing with lifting surfaces, it is usually possible to account for interference as changes in mean angle-of-attack and local

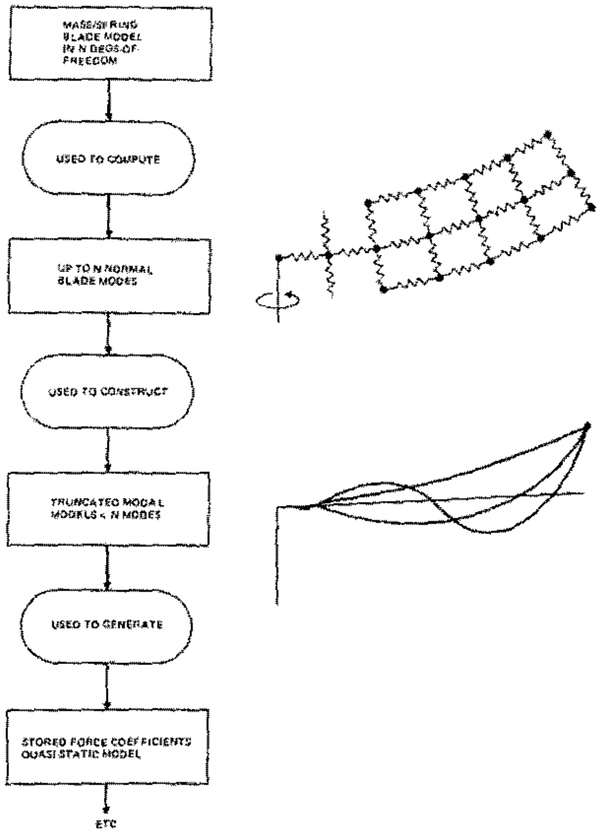

Figure 3. Hierarchical Module Structure for Rotors

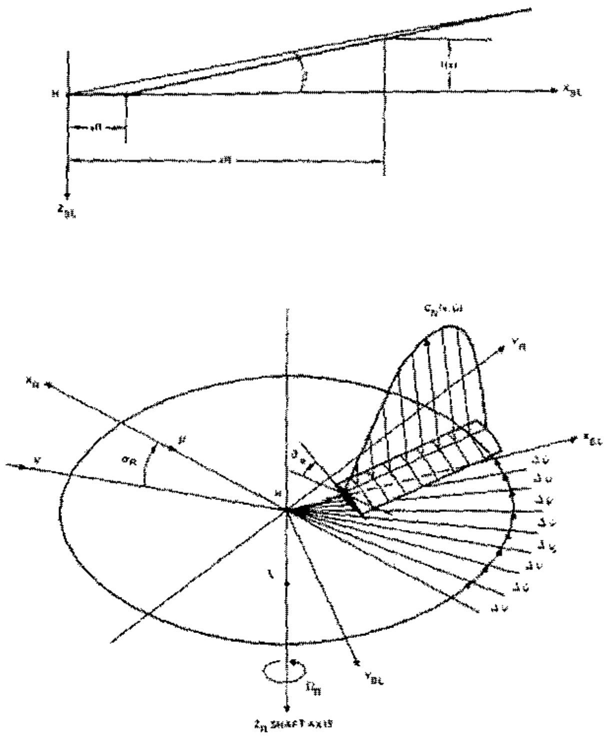

Figure 4. Main Rotor Master Model

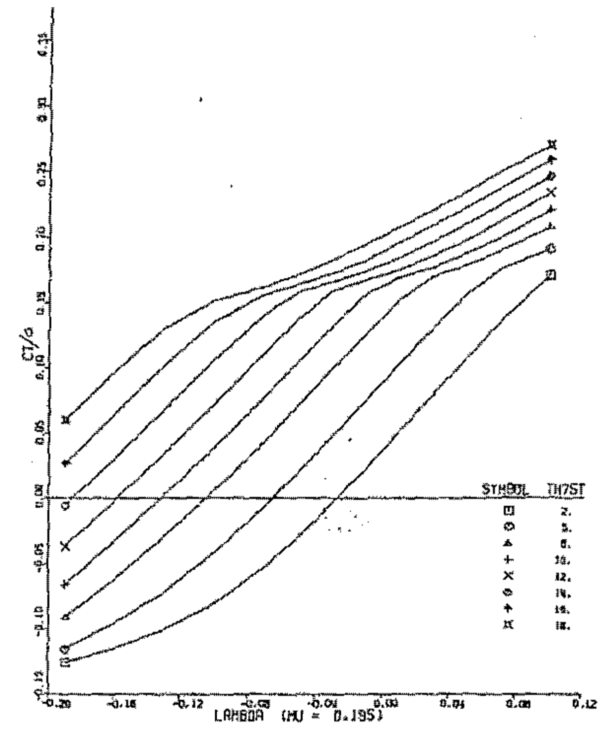

Figure 5a. Main Rotor Thrust Coefficients Hughes 500D

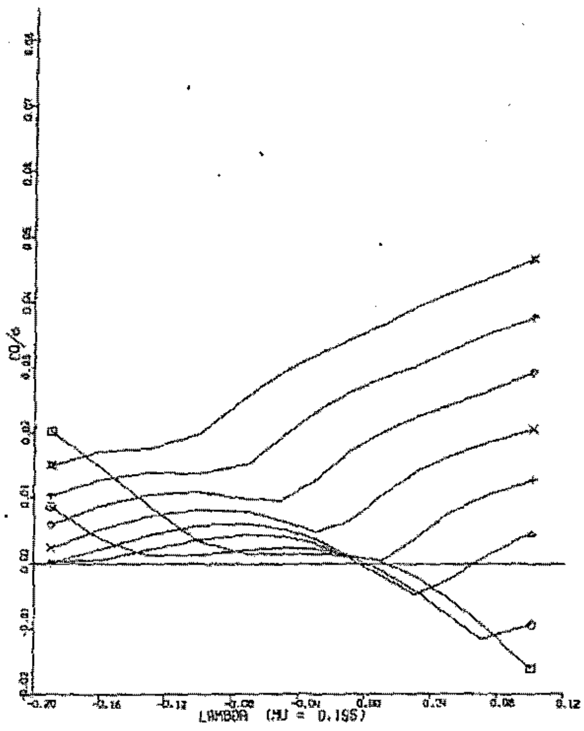

Figure 5b. Main Rotor Torque Coefficients Hughes 500D

dynamic pressure. The conventional approach is to define the changes as a function of main rotor momentum downwash, using a weighting factor. The factor in turn is defined as a function of the main rotor wake skew angle. Assuming that wind tunnel data is available from an unpowered model, the wing/fuselage combination can be treated in the first instance as a lifting surface. Recently when interpreting powered model wind tunnel tests, it was found necessary to introduce a second angular parameter, also definable as a function of main rotor wake skew angle, to account for an appreciable longitudinal bias of induced velocity in the after wake affecting the horizontal tail. The same series of tests also yielded information for deducing fuselage blockage effect. The method of interpretation is illustrated diagramatically in Figure 6. Yet a further refinement, making provision for observed main rotor wake assymetry, involved sub-division of the horizontal tail surface into right and left panels, treating each independently. It is well known that passage of the main rotor wake over a large horizontal tail gives rise to rapid variations of trim within the transition region. Even with all the refinements described above, it is not always possible to match precisely trim profiles measured in flight. A plausible way of accounting for residual discrepancies is to include the effect of the high energy regions of the main rotor wake impinging on the front fuselage, thereby generating viscous tractions. Such forces are incremental and useful for fine tuning.

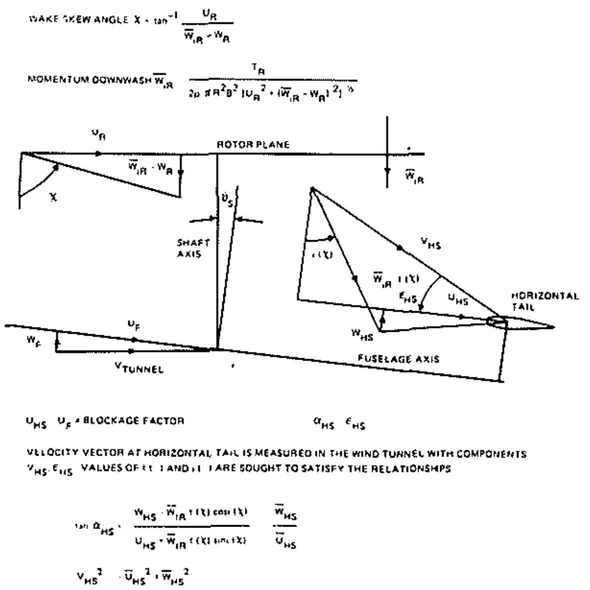

Figure 6. Main Rotor-to-Horizontal Tail Interference

All such interference models are simple in concept, and quite suited to dealing with performance and handling qualities. Treatment of mutual interference between tail rotor and vertical tail has been even simpler. At low speeds the tail rotor sees the vertical surface as a ground plate and generates more thrust than an equivalent isolated rotor. The excess thrust is more than compensated by the induced flat plate drag. At higher speeds, the two components can be represented as a Prandtl biplane.

2.4 Engine Sub-Models

The current engine model is simple and linear, sufficient to act as a link in the control loop. Provision is made for total or partial failure of the subsystem, otherwise torque is computed as a function of the drive train speed error. Since however integration of an engine sub-system is a valid subject for future studies, its true status as a major vehicle component is recognized and provision made within the program structure.

2.5 Flight Control System

Successful simulation of a complicated maneuver requires that commands be imposed on a stable system. Since a helicopter is inherently unstable, the subject vehicle model must include provision for artificial stabilization whether or not it be actually mechanized. Where flight control is manual, then such provision is explainable as pilot action. Some pilot actions are described quite adequately as conventional linear control laws, and the structures of human pilot model for manually controlled vehicles and automatic control systems are superficially similar. Only the characteristics differ. Whatever the label, a module containing control functions is an essential part of the system model. His mechanical control functions apart, the pilot is also required to exercise judgement, and make decisions. Then some actions are best described by adaptive control laws. All these functions are dealt with in a separate maneuver module to be described later.

3. Program Structure

Each model of a major component is independent with its own frame of reference. Each is realized as a sub-system or part of a sub-system within a replaceable program module. Thus wing/fuselage and tail rotor/vertical tail are examples of combinations, whereas main rotor and horizontal tail are accorded individual treatment. For inertial purposes the vehicle is treated as a rigid body with the rotor masses concentrated at the hub centers. Components are linked aerodynamically by mutual interference as defined in paragraph 2. Modules communicate each with its own data bank and with the main program, accepting velocity, attitude and control vectors as input and returning a force vector. The main programs are organized according to function, and are modular in construction. There are two types:

3.1 Trim Program

The trim program served originally to validate the vehicle model, its main features being the vehicle equations of motion and a perturbation cycle. In operation, starting from an arbitrary datum, a selected vector is perturbed systematically element-by-element, and the resultant increment of the vehicle acceleration vector used to compute a matrix of partial derivatives. A trim vector has six components, usually comprising main rotor collective pitch, longitudinal and lateral cyclic pitch, tail rotor collective pitch and the two Euler angles, pitch and bank attitude. Others may be substituted according to the desired trim status. The trim matrix, when inverted, can be used to iterate towards a steady flight configuration, for when post-multiplied by the acceleration vector, it yields an incremental vector of trim parameters. The updated trim vector is then used to compute a new residual acceleration vector, which should be driven towards a zero value. The perturbation procedure is illustrated in Figure 7. Following attainment of trim, the perturbation cycle can be re-activated to operate successively on the velocity and control vectors, thereby generating a linear perturbation model related to the subject

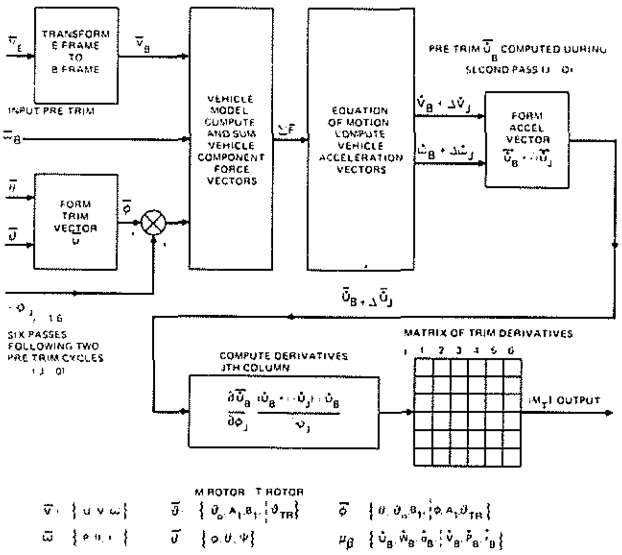

Figure 7. Trim Program - Pre-Trim Perturbation Mode

flight configuration. To facilitate the generation of such models, the six Euler equations have been converted to state variable form. The remaining two equations are kinetic and can be computed analytically. Use of the perturbation mode is not confined to the six body degrees-of-freedom. As implied in Figure 2, the main rotor module can be replaced by one based on the generating model. The perturbation process is thereby complicated by the need to transform from a rotating to a stationary frame of reference. The end product is a model with extra degrees-of-freedom in blade dynamic motion, expressed as multi-blade modes, usually dominated by the collective and cyclic regressive modes. Such models have many important applications.

Following initial validation, the main program was organized to generate the various categories of matrix on option and to transmit them to permanent files for access by other programs. The trimmed configuration itself is defined and transmitted as a data string and constitutes initialization for the fly program.

3.2 Fly Program

The fly program, as its name implies, is organized to simulate specific flight maneuvers. To this end additional modules have been supplied, for resolution and time integration, models of the engines and power train, flight control system and/or pilot as well as extended provision for input and output of data. What identifies each maneuver is a module containing the requisite command structure. During the validation period, the module contained nothing more elaborate than options to pulse each control channel selectively. Later modules have reflected the complication and duration of the maneuver. The most extensive module to date is used to simulate recovery following partial or total power failure or achievement of a safe landing and will be outlined for illustration later. The fly cycle is illustrated diagramatically in Figure 8.

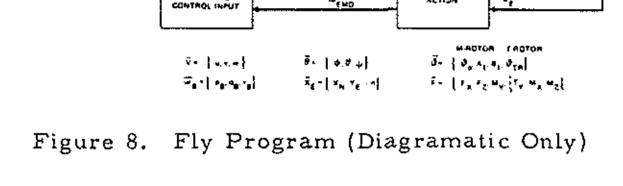

Figure 8. Fly Program (Diagramatic Only)

It was implied in paragraph 2.1 that real time capability of the fly program is due in part to the simplicity of the main rotor model. The simplicity is merely apparent and was achieved only through considerable effort. Conversion of the body equations of motion to state variable form has been mentioned earlier in connection with the trim program perturbation mode. Such conversion imposes restrictions on the definitions of the expressions for the body forces generated by the vehicle components. In particular inertial components of the forces cannot be functions of the body acceleration vector. A major analytical effort, devoted to the elimination of such items from the main rotor contributions, was well justified, for, in state variable form, each of the six equations is independent and of first order. When the Z-transform is applied to derive the time integration difference equations, they emerge in the simplest possible form. Use of the Z-transform was not arbitrary, although in this instance it might seem trivial. The need to model the flight control system, or the pilot, or both, was taken into account noting that the Z-transform is particularly well suited to such applications. Information on analog systems is usually supplied as block diagrams depicted in the S-plane and is readily transformed to the Z-plane. Digital systems pose the least problems as being already depicted in the Z-plane. Better still the control laws might be already cast as difference equations.

3.2 Matrix Analysis Program

The ability to generate large quantities of linear perturbation models at will mandates the availability of a dedicated program to process them. The program provided is typical in that its repertoire includes all the capabilities required for classical servo-mechanism analysis. It was however written with more in mind. For production purposes it has selective access to large quantities of data pre-stored systematically in permanent files. Operationally it is well integrated with the simulation programs. Any system of simultaneous equations it accepts whether transmitted from a permanent file, or read as random input, is first converted and then reprinted in state variable form. This form facilitates the rapid computation of the roots defining the numerator and denominator of each transfer function. Transient response solutions are computed in closed analytical form so that dominant components can be identified. A typical application was to compute the main rotor flapping responses within a stationary frame of reference as an essential stage in synthesizing the equivalent first order equations of flapping motion. The resulting time constant was an important by-product in that it is critical in determining the optimum integration time increment or sampling period used by the fly program. One of the program's more powerful features is an option to transform transfer functions from S-plane to Z-plane and recast the closed form transient response solution as difference equations. This option has been used to model control sub-systems higher than the second order.

4. Power-Off Landing Maneuver

The description covers a wide range of actions having in common a feature that the main and tail rotors are energized solely by the incident airstream. The simplest of these is a transition into steady autorotational descent at constant ground speed. The most critical occurs when power is lost at low speed with insufficient height margin to complete a transition, and incidentally is a good example for illustrating the procedures adopted when programming a specific maneuver. It is first desirable that a maneuver be divided into readily recognizable stages. Thus, four stages have been identified in this maneuver sequence:

1. Initial Reaction: Involves delay in recognizing the situation and is characterized by vehicle acceleration forward and downward in response to pilot reaction. Rotor speed decays rapidly.
2. Initial Flare: Vehicle downward acceleration and rotor deceleration is checked as the pilot applies a nose up command. Collective pitch has been reduced to a minimum. Normal acceleration builds up rapidly.
3. Final Flare: The helicopter rounds out to approach a suitable landing configuration, attaining maximum nose-up attitude for rapid deceleration. Maximum rotor speed is approached and controlled by progressive application of collective pitch.
4. Pre-Touchdown: Rate-of-descent has been reduced below a safe margin. Residual rotor energy is expended by rapid application of collective pitch, reducing forward speed. Attitude is controlled carefully to synchronize attainment of a safe landing speed, rate-of-descent and nose-down rate-of-pitch.

These four stages are not necessarily distinct in terms of pilot action, and certainly not in terms of vehicle response. It is assumed that power loss occurs either during climb out or before attainment of speed for minimum power. Typical pilot reactions are available from flight records. Application of forward stick appears to be instinctive. Collective pitch is dumped deliberately after a specified delay. The resultant acceleration is controlled by an abrupt stick back command signaling entry to initial flare. Figure 9 indicates that recovery starts before completion of collective pitch dump. Initial flare proper has been simulated using a blend of adaptive control laws based on pitch rate, normal acceleration and rate-of-descent. Collective pitch is usually inactive throughout the initial flare. Timing of entry into final flare is critical. The simulated pilot uses as criteria a combination of attained attitude and normal acceleration. Alternatively, a steady approach to maximum rotor speed is a signal to switch collective pitch from speed to height control at low gain, and to reverse the stick command to a forward bias. When the control actions are phased correctly, ground speed, and pitch attitude approach zero together at a safe rate of descent. The final decision is most critical; when to increase rate of collective pitch application, simulated by increasing gain. Ground contact should be made, ideally, as rotor speed decays below a usable level. Regular success was achieved when stage 4 was divided into three sub-stages each identified by an arbitrary check point. Check

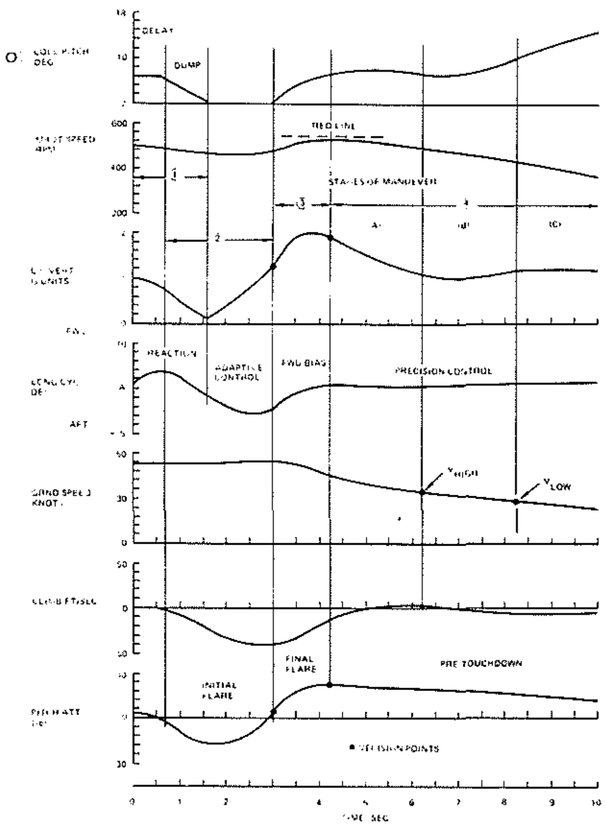

Figure 9. HH500D Simulated Power-Off Landing

point one is signalled by either decay of normal acceleration below a safe value or on attainment of peak nose-up attitude. What is safe turned out to be dependent on drive train inertia. Taking advantage of available rotor energy, collective pitch gain is increased progressively until ground speed falls below a high value at check point two. The high value is chosen as being suitable for turning up collective pitch gain to maximum. Check point three is passed as ground speed falls below a low value, usually marginally higher than the maximum safe landing speed. At each check point, pitch rate command is changed to discrete pre-set values, and allowed to decay slowly to zero. In this way precision control over nose-up attitude and nose-down pitch rate is maintained. Meanwhile, throughout stage 4, in addition to ground speed and pitch rate, pitch attitude and rate-of-descent are monitored. When it is evident that an acceptable landing configuration is being approached smoothly, the ground plane is introduced, a few feet below wheel or skid height, so that contact can be made realistically in ground effect.

Having achieved an acceptable landing, the key control and decision parameters can be varied systematically about the optimum values to assess how much latitude the pilot has. In the process a mean point on the height/velocity curve is generated. Alternatively having defined an optimum point, design parameters can be varied. The procedure tends to be more complicated, for, a change in say the main rotor polar moment of inertia can effect the piloting technique appreciably. Changes in technique are most marked when active auxiliary energizing devices are introduced, a subject beyond the scope of this paper.

5. Scope of Method

No attempt has yet been made to extend the scope beyond application to performance and handling qualities problems. The description is intended to include all feasible formal maneuvers whether executed to simulate actual operational flying or prescribed to reproduce a specified design condition. Figure 10 illustrates a typical operational maneuver, a lateral acceleration from hover. The objective is to attain maximum acceleration, reach a specified target velocity and maintain heading. Incidentally, this maneuver is a severe test of the tail rotor model, exercising it towards the limit of its capability. The rolling pull-out shown in Figure 11 is an example of a prescribed maneuver, deliberately exaggerated. The requirements call for full right stick to initiate a coordinated turn at some specified normal acceleration. In this instance, the maneuver was programmed to approach 3g with 70 deg, bank angle and is intended to saturate the main rotor as well as exercise the whole vehicle model.

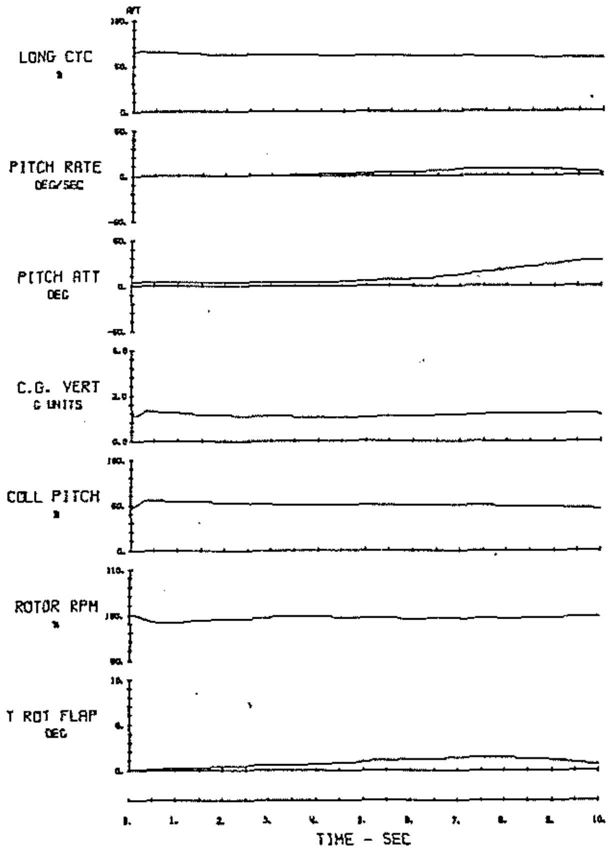

Figure 10a. Simulated Lateral Acceleration Maneuver

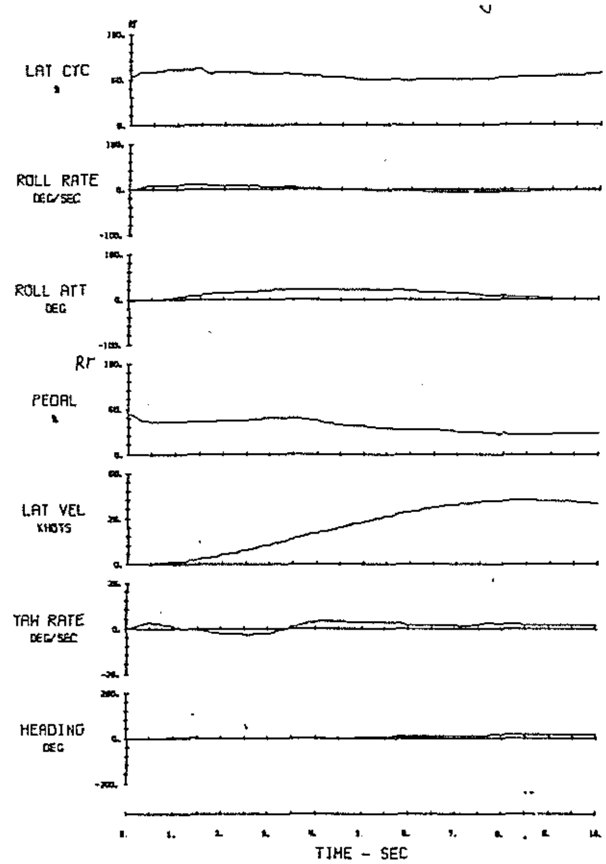

Figure 10b. Simulated Lateral Acceleration Maneuver

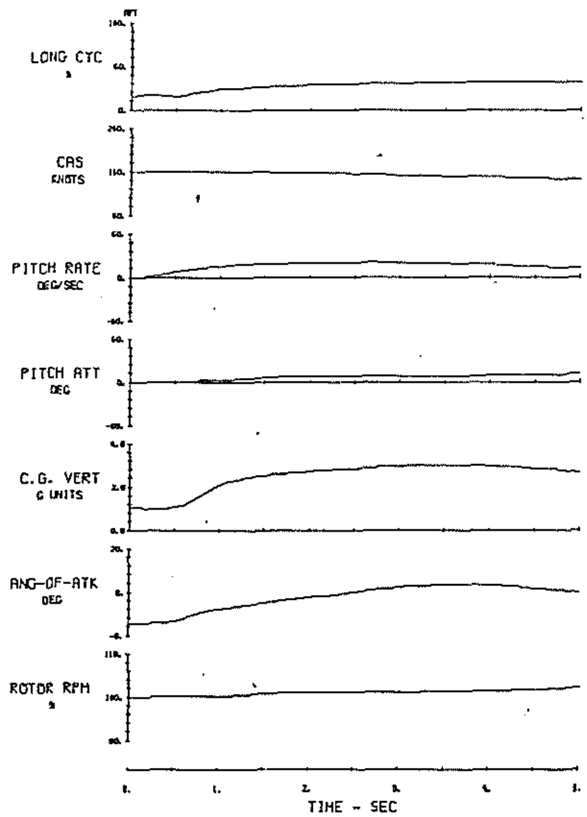

Figure 11a. Simulated Rolling Pull-out Maneuver

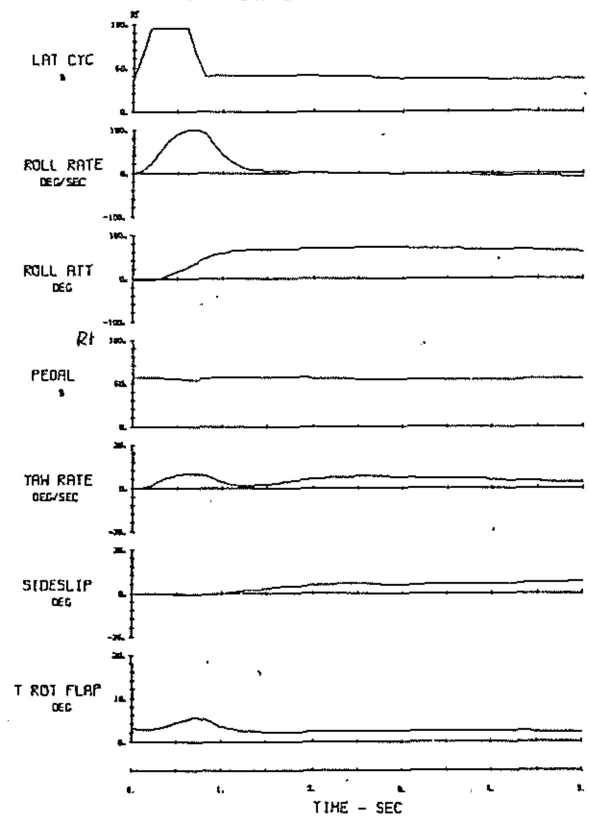

Figure 11b. Simulated Rolling Pull-out Maneuver

The two cases are offered as routine examples displaying the potential of the method.

6. Conclusion

In a paper of this kind, there are neither conclusions nor conslusion. The method outlined is in a continuous state of development as component models are extended and refined, or program material is added to the repertoire in response to consumer request. The more obvious lines of future development have been hinted at. The main rotor model is amenable to considerable expansion, for example; the admission of lagwise motion in order to accommodate more advanced engine system models or the admission of dynamic feathering under elastic restraint to enable realistic computation of swashplate loads. With regard to the air frame model, options to admit body modes in elastic deformation have been considered for special applications at the sacrifice of real time capability.