MATHEMATICAL MODELING OF VEHICLE DYNAMICS

Part I

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INTRODUCTION

Major topics addressed include:

- -Modeling fundamentals
- -Aerodynamic math models
- -Supporting model components
- -Ground vehicle and ship models
- -Flight test validation data

The simulation of vehicle dynamics on a digital computer requires the development of mathematical models that characterize system behavior. The complexity of the models depends on the application of the simulation. Complexity of the simulator math model determines the size of the effort needed to design the model, acquire validation data, prepare check cases, and implement the model in a simulator platform. For engineering development of aircraft, detailed models are typically developed that focus on the system of interest. Supporting systems or environment models are then applied only to the extent necessary to "feed" the primary system. As a result, detailed models of the engine, the flight control system, the aircraft stability and control characteristics, and aircraft performance may be created but none of these detailed models may be capable of operating together in an integrated fashion. When the engineering goal calls for an integrated model, then the simulation team must agree on common assumptions and interface parameters. Less rigorous models of each subsystem may yield acceptable results for the integrated application. Additional factors that affect the complexity and ultimate validity of a model are computer limitations and data quality. This discussion is intended to illustrate a fundamental concept of modeling: the intended purpose of the model must be declared and understood by all users. Also, the assumptions applied to model construction must be known. Sometimes these concepts are overlooked when simulation projects attempt to reuse existing models.

MODELING FUNDAMENTALS

The nature of a math model used to simulate a particular system is dependent on the characteristics of the system being simulated. Systems may be classified as deterministic or stochastic, continuous or discrete, stationary or non-stationary, linear or non-linear. A deterministic system is defined as a system in which, when an input is precisely repeated, the output will be precisely repeated. This is not true for a stochastic system because the phenomena are probabilistic in nature. A continuous system has an output that is defined for all time, whereas a discrete system is one where the output occurs at discrete intervals of time. It should be noted that a continuous system becomes a discrete system when modeled in a digital computer because of its discrete sampling interval. A stationary system is one for which the describing differential equations have coefficients which are constant with respect to time. Most natural phenomena exhibit non-linear behavior, but linear expressions are developed by various techniques to facilitate analysis.

A typical model structure for flight simulation is shown in Figure 1. Note that the operator (pilot) interacts with the vehicle dynamics through the flight controls and then the forces and moments acting upon the vehicle are computed and summed. The force and moment summation is supplied to the equations of motion where the vehicle state is computed in terms of position, attitude orientation, and velocity. (Allerton 2009, Drier 2007, Stevens 2003, Rolfe 1986)

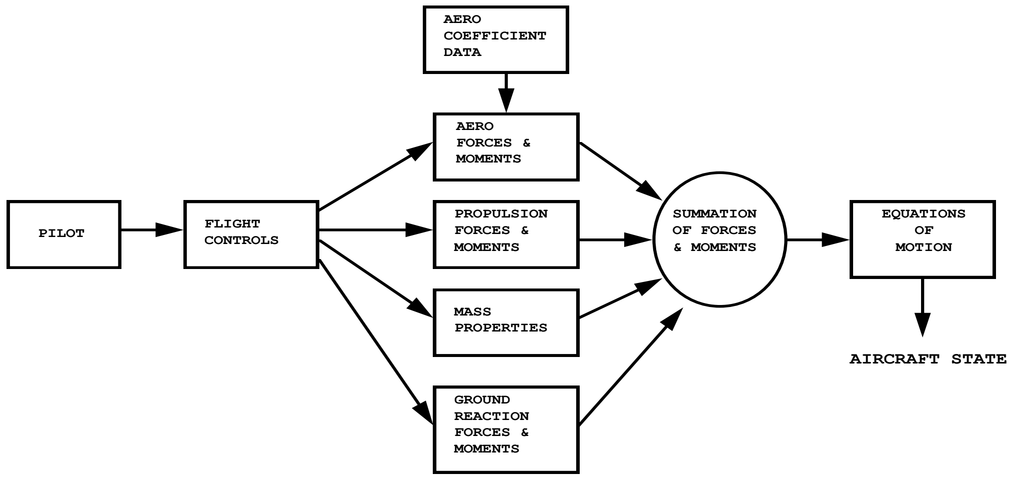


Figure 1
Typical Flight Simulation Model Structure

Equations of Motion

The typical vehicle dynamics simulation math model is a deterministic, continuous, stationary, and linearized set of equations which describe the behavior of a vehicle and its systems. The construction of a simulation model of vehicle dynamic behavior begins with the application of basic physical laws to define the equations of motion. The equations of motion are derived by applying Newton's laws of motion that relate the summation of external forces and moments to the linear and angular accelerations of the system or body. The usual non-linear equations applied in vehicle simulation are as follows:

<u>Force Equations</u>

$X - mg \sin\theta = m(\mathbf{u}\boldsymbol{dot} + qw - rv)$
$Y + mg \cos\theta \sin\phi = m(\mathbf{v}\boldsymbol{dot} + ru - pw)$
$Z + mg \cos\theta \cos\phi = m(\mathbf{w}\boldsymbol{dot} + pv - qu)$

<u>Moment Equations</u>

$L = I_x\, \mathbf{p}\boldsymbol{dot} - I_{xz}\, r\,dot + qr\,(I_z - I_y) - I_{xz}\, pq$
$M = I_y\, \mathbf{q}\boldsymbol{dot} + rp\,(I_x - I_z) + I_{xz}\,(p^2 - r^2)$
$N = I_z\, \mathbf{r}\boldsymbol{dot} - I_{xz}\, p\,dot + pq\,(I_y - I_x) + I_{xz}\, qr$

where: u, v, w are body translational velocities
p, q, r are body angular rates
Ixy, Iyz are normally zero due to body symmetry
mg terms are gravity force components
θ, ϕ are pitch and roll Euler angles

The external forces and moments can be aerodynamic, hydrodynamic, ground reaction, etc., but the general idea is that their net effect is summed in these six fully coupled, non-linear equations. The equations shown above are implemented in the simulation in a rearranged form to compute the resulting accelerations. In other words, the familiar expression, F = ma, is rearranged to: a = F/m, and solved for a. These accelerations are then integrated with respect to time to get velocities and then integrated again to get position and attitude, which are the vehicle state parameters. When computed this way, there are six parameters, three translation and three rotation, and this is referred to as a six degree of freedom model.

Common assumptions for vehicle dynamics modeling include: the vehicle is a rigid body with constant mass. The effects of fuel consumption and release weapons or cargo do not violate these assumptions since we are only talking about one computation cycle.

In addition to this total force and moment approach, two other general approaches are commonly used - within their limitations. One approach is the *perturbation model* which is used to facilitate analysis of complex vehicle dynamic characteristics. The perturbation model is limited to small excursions about a steady state condition and so is not suitable for manned vehicle simulation or large perturbations from steady state initial conditions. Large perturbations violate the linear assumptions used to assign values to coefficients within the model. The other approach, the *kinematic model*, is limited to applications where the relationship between output and input can be directly modeled with simple linear equations. An example is where vehicle attitude and velocity is a direct function of operator input (no forces, moments, or accelerations computed). The nuances of vehicle response that an operator expects are not reproduced with this type of model. Kinematic models are widely used in amusement applications and in training simulations for secondary player vehicle dynamics. More will be said about this later.

Coordinate Systems

Vehicle motion must be computed relative to a fixed frame of reference and vehicle orientation is typically described using several coordinate systems, all based on the familiar right hand rule. The basic coordinate systems are: inertial, earth, and body. The typical aircraft modeling approach will now be described as a general example of how all this works.

For inertial coordinates, the aircraft equations of motion utilize a fixed, earth-centered orthogonal triad reference frame referenced to a rotating earth (Figure 2). The earth's shape is represented as round or more precisely in some models as an oblate spheroid. The X and Y coordinates define the equatorial plane and the Z axis is positive toward the North Pole. The X axis is typically initialized through the zero latitude and longitude point, and the Y axis would therefore pass through the 90 degree longitude point. Earth rotation will cause an earth fixed longitude point to increase in inertial longitude. Integration of aircraft accelerations and velocities are carried out in the inertial frame.

The Earth or NED (North, East, Down) coordinates are centered at the vehicle's center of gravity (Figure 3). The X axis is directed toward true North, the Y axis is directed toward true East, and the Z axis is directed down perpendicular to the ground. Aircraft translational velocities are transformed from inertial to the NED reference frame for the incorporation of wind velocities and computation of ground speed and flight path angles. The aircraft attitude, or Euler angles (θ,theta:pitch, ϕ,phi:roll, ψ,psi:yaw) are defined from the body axes orientation relative to the NED reference frame.

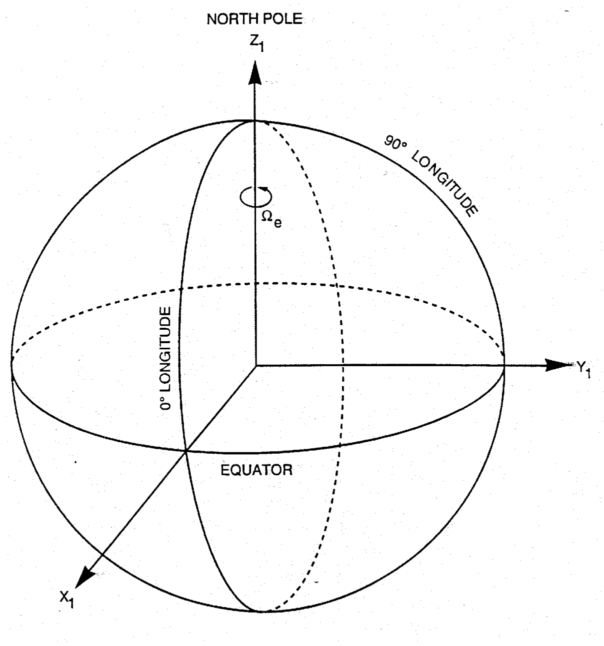


Figure 2
Inertial Coordinate System

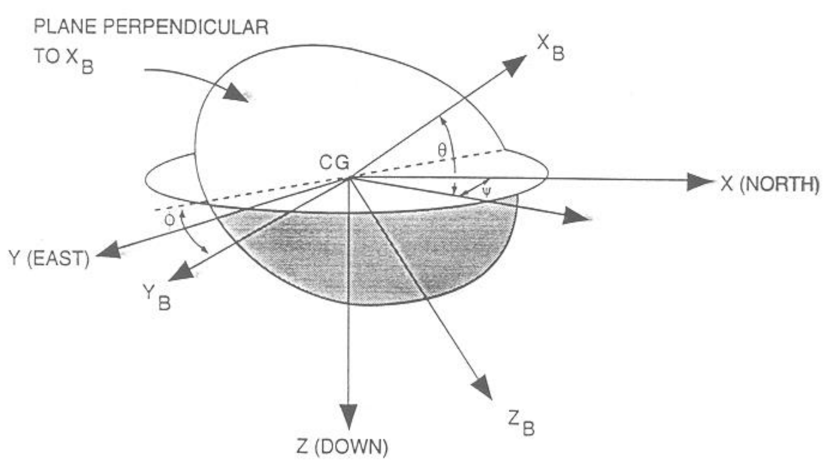


Figure 3
NED Coordinate System and Euler Angles

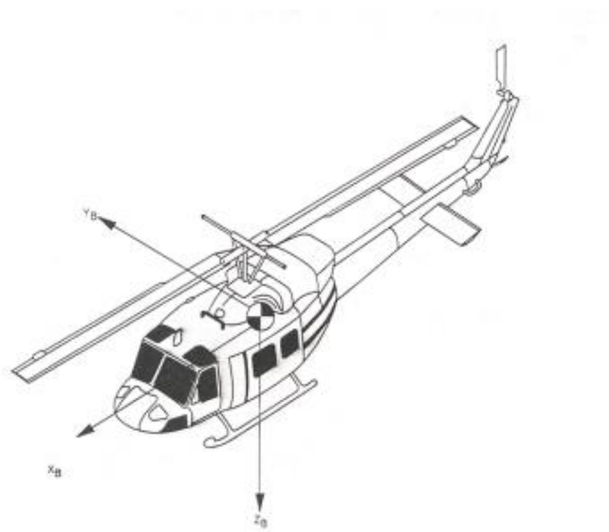


Figure 4
Body Axis Coordinate System

The vehicle body coordinate system has its origin at the center of gravity with the X axis directed toward the nose parallel to the fuselage reference line (Figure 4). The Y axis is orthogonal to the X axis and directed out the right wing (or right fuselage side), while the Z axis is directed downward and orthogonal to the X-Y body axes plane. The body axis system is used to sum all the external forces and moments on the aircraft. Many additional coordinate systems may be superimposed on the body axes for articulated parts (helicopter rotors, tank turrets, etc.) and the modelers must be careful to define and document all of these geometric relationships.

Coordinate Transformations

Transformations between the above described coordinate systems require the computation of transformation matrices. Traditionally there are three methods of expressing the orientation of one coordinate system with respect to another. These three methods are Euler Angle, Direction Cosine, and Quaternion. The Direction Cosine method is not commonly used now and so will not be described here. However, the final output of all these transformation processes is a direction cosine matrix that projects a vector from one coordinate into another. These notes will provide a brief description of the two common methods along with advantages and disadvantages of both. A comprehensive review of the attitude representations used for aircraft kinematics is presented in Phillips 2001 and 2004 and to some extent in Allerton 2009 and Dreier 2007.

Euler Angle Method. The Euler Angle method is used frequently in vehicle simulation because it is simple to mechanize in digital computer programs and the computation process utilizes parameters with obvious physical significance. The negative attribute of the Euler Angle method is the mathematical singularity that exists when pitch angle is +/- 90 degrees. This is the familiar gimbal lock phenomenon and it can be overcome by either prohibiting pitch angle excursions near 90 degrees or by employing small mathematical deadbands to prevent a division by zero. The Euler Angle method is derived from the direct kinematic relationship between the body axis angular rates (p, q, r) and the Euler attitude angles. The implemented equations compute the rates and integrate with respect to time in the following order:

Yaw: $\psi dot = (r \cos \phi + q \sin \phi) / \cos \theta$
Pitch: $\theta dot = q \cos \phi - r \sin \phi$
Roll: $\phi dot = p + q \tan \theta \sin \phi + r \tan \theta \cos \phi$

Next, the elements are computed for the direction cosine matrix necessary to transform from the moving body axes to the desired fixed axes system. This matrix typically looks like:

Direction Cosine Matrix

-for transformation from fixed to moving axes
-use transpose for transformation from moving to fixed axes

$$[M] = \begin{matrix} a_1 & a_2 & a_3 \\ b_1 & b_2 & b_3 \\ c_1 & c_2 & c_3 \end{matrix}$$

Transformation matrix definitions

$a_1 = \cos\theta \cos\psi$
$a_2 = \cos\theta \sin\psi$
$a_3 = -\sin\theta$
$b_1 = \sin\phi \sin\theta \cos\psi - \cos\phi \sin\psi$
$b_2 = \sin\phi \sin\theta \sin\psi + \cos\phi \cos\psi$
$b_3 = \sin\phi \cos\theta$
$c_1 = \cos\phi \sin\theta \cos\psi + \sin\phi \sin\psi$
$c_2 = \cos\phi \sin\theta \sin\psi - \sin\phi \cos\psi$
$c_3 = \cos\phi \cos\theta$

Quaternions. A more elegant method for overcoming the singularity problem of the Euler Angle method is a four parameter system that was first developed by Euler in the 1700's. This system was subsequently modified by Hamilton in the 1800's and he named it the Quaternion System. Detailed descriptions and derivations for modern aerospace applications can be found in the literature(ANSI/AIAA 1992, Stevens 2003, Phillips 2004, Robinson 1958). The concept defines the aircraft body axis orientation with respect to another axis system by a direction cosine matrix whose elements are functions of four quaternions. Quaternions are defined as a rotation through some angle about some specific fixed axis. Quaternion implementation in simulation may appear to be computationally more intense than the Euler Method, but in practice it is not and it eliminates the need for additional logic to deal with the singularity problem.

In a typical simulation application, the four parameters, usually labeled e_0, e_1, e_2, e_3, are defined by the initial values of the three Euler angles. Subtle definition differences appear in the various references and applications, therefore, the simulation modeler must be careful to maintain a consistent implementation.

$e_0 = \cos\psi/2 \cos\theta/2 \cos\phi/2 + \sin\psi/2 \sin\theta/2 \sin\phi/2$
$e_1 = \cos\psi/2 \cos\theta/2 \sin\phi/2 - \sin\psi/2 \sin\theta/2 \cos\phi/2$
$e_2 = \cos\psi/2 \sin\theta/2 \cos\phi/2 + \sin\psi/2 \cos\theta/2 \sin\phi/2$
$e_3 = -\cos\psi/2 \sin\theta/2 \sin\phi/2 + \sin\psi/2 \cos\theta/2 \cos\phi/2$

Next, the quaternion rates are computed by a defined relationship with the vehicle body angular rates (p, q, r) and then integrated to obtain updated values for the quaternions.

$e_0\,dot = -0.5\,(e_1p + e_2q + e_3r) + ke_0$
$e_1dot = 0.5\,(e_0p + e_2r - e_3q) + ke_1$
$e_2dot = 0.5\,(e_0q + e_3p - e_1r) + ke_2$
$e_3dot = 0.5\,(e_0r + e_1q - e_2p) + ke_3$

where ke_i is for integration drift correction (unnecessary if word size at least 32 bits)

Finally, the nine elements of the direction cosine matrix are computed using defined combinations of the quaternion values. This direction cosine matrix is identical to the one described previously for transforming from the fixed (earth/inertial) axes to a vehicle (moving) coordinate system. As always, the transpose of this matrix is used to transform from the vehicle to a fixed axis system.

$a_1 = (e_0^2 + e_1^2 - e_2^2 - e_3^2)$
$a_2 = 2(e_1e_2 + e_0e_3)$
$a_3 = 2(e_1e_3 - e_0e_2)$

$b_1 = 2(e_1e_2 - e_0e_3)$
$b_2 = (e_0^2 + e_2^2 - e_1^2 - e_3^2)$
$b_3 = 2(e_2e_3 + e_0e_1)$

$c_1 = 2(e_1e_3 + e_0e_2)$
$c_2 = 2(e_2e_3 - e_0e_1)$
$c_3 = (e_0^2 + e_3^2 - e_1^2 - e_2^2)$

The updated Euler angle values are computed from elements of this matrix.

$\theta = \sin^{-1} [-a_3]$
$\psi = \tan^{-1} [a_2/a_1]$
$\phi = \tan^{-1} [b_3/c_3]$

Other Transformations. Usually, a few other coordinate transformations have to be performed to complete the simulation problem. A transformation from earth velocities back to body velocities is necessary to compute local flow parameters such as airspeed, angle of attack, and sideslip. This involves inverting the transformation matrix described above and applying it. Another important transformation is necessitated by the graphics database implemented in the simulator visual system as a "flat earth" representation of the local operating area. Also, movement of secondary players in a simulation may be computed only in flat earth coordinates. Therefore, the round (or near round) earth coordinates of the primary vehicle must be transformed by correcting aircraft heading based on local latitude and longitude. Some simulators may skip the round, rotating earth entirely and simply designate the flat earth system as the inertial reference frame. This is suitable for limited applications where navigational accuracy in not very important.

The interest and activity devoted to linking separate simulators together for networked operation created a need for a universally applied 'world' coordinate system. Most of the players in a network exercise are operating in their own local flat earth, or topocentric, reference frame. Exercises conducted according to the Distributed Interactive Simulation (DIS) guidelines utilize a common geocentric coordinate system to describe player locations in the whole gaming area. A third coordinate system, the geodetic, was introduced to enable transformations between the topocentric and geocentric coordinate systems. Geodetic coordinates are defined using three quantities: latitude, longitude, and geodetic height. The latter defines the position of a point on the earth's surface with respect to a reference ellipsoid, and for DIS, the shape of the earth is specified using the World Geodetic System 1984 (WGS84). The Defense Mapping Agency documented WGS84 in a thorough technical report (DMA TR 8350.2, 1987). Also, technical papers can be found in the literature that discusses methods for efficient conversions between these axis systems (Lin 1992).

Numerical Integration

In the EOM process described above, integration with respect to time was required at several points. When implementing integration on a digital computer, digital sampling causes the system to behave as a discrete system. The discrete nature of the digital computer model presents some unique problems for the representation of a continuous dynamic phenomenon. Since real-time, man-in-the-loop simulation of vehicle dynamics involves the summation of all forces, solving for acceleration and integrating twice to obtain velocity and position of the vehicle, then the issue of how these integrations are performed is important. In addition, integrations are performed in other parts of the simulation including control systems, navigation systems, weapon systems, motion cueing, and visual image generation.

Integration on a digital computer requires that a numerical approximation be employed to represent a continuous process in a the discrete time environment. Integration is defined mathematically as the area under a curve of the function being integrated; hence, the numerical integration algorithm is designed to approximate this area. There are many such algorithms, but these notes will describe those most commonly applied in real-time simulations.

When selecting an integration algorithm, the following factors must be considered: stability of the integration, accuracy of the approximation, speed of execution of the algorithm, and the time distortion caused by the process. This last item, time distortion, is of greatest importance in vehicle dynamics simulation because it introduces a phase shift, and therefore an equivalent system delay which affects the closed loop response of the simulated vehicle. This constitutes a polluting effect on the whole simulation that must be minimized in order to have a credible model. Another issue is integration accuracy, which is important for weapon delivery and scoring.

The two most significant choices available to the simulation designer for proper integration are the algorithm and the time step size, or quadrature interval. Consider a plot of acceleration that varies as a function of time. Integration by linear approximations of area under the curve will get velocity as a function of time. The area under the curve is divided into equal time intervals and it should be obvious that as these time intervals get smaller, the area approximation becomes more accurate. Therefore, as time step size gets smaller, the integration process becomes more accurate. In the early days of flight simulation, this was a significant issue because computer speeds were so slow - on the order of 10 computation cycles per second or less. Computer speed is less of a problem now and it is up to the simulator designers to make intelligent choices for quadrature intervals. The rigid body dynamics of an aircraft are not impacted much at the 60 Hz rates common today. However, for stiff systems such as helicopter rotor components, aircraft aeroelastic effects, and servo system inner loops, much higher rates (smaller quadrature intervals) are required for stable, undistorted simulation.

A common, but incomplete, rule of thumb used for determining the update rate of a simulation is that it must be 10 to 100 times the highest natural frequency being simulated. For typical rigid body aircraft dynamics with a frequency of about 1 Hz, the update rate should be between 10 Hz and 100 Hz. It turns out that the choice depends more on the particular algorithm chosen and the total system delay, including the visual system (which typically adds 60 ms or more delay). So, 20 to 30 Hz would work fine for simulating aircraft dynamics with most integration algorithms, but 20 Hz carries the liability of 50 ms processing time plus the same amount again for sampling uncertainty. A simulation at 60 Hz involves just 16.7 ms processing time and the output sampling uncertainty can be eliminated if synchronized with a 60 Hz visual system. Therefore, the major consideration is not dynamics but sampling delays and their effect on pilot performance.

Some of the most frequently used integration algorithms are:

Euler (Rectangular)	$X_n = X_{n-1} + Xdot_{n-1}\ dt$
Advanced Euler	$X_n = X_{n-1} + Xdot_n\ dt$
Adams Second Order (Trapezoidal)	$X_n = X_{n-1} + (1.5\ Xdot_{n-1} - 0.5\ Xdot_{n-2})\ dt$
Tustin	$X_n = X_{n-1} + (0.5\ Xdot_n + 0.5\ Xdot_{n-1})\ dt$

These algorithms fall into two general classes. The first class is the predictor, or open form, where the input data is valid for the beginning of the integration interval and the output is predicted for the end of the interval. The Euler and Adams equations shown above are predictor algorithms. The second class is the corrector, or closed form integration scheme, as represented by the Advanced Euler and Tustin equations shown above. This scheme is used when the input is available at the same time that the output is desired. Note that the Adams equation is considered second order because it uses two past values of the derivative. There are other algorithms in use in real time simulation, many of which are only slight variations on the four presented above. Discussions on the nature of integration algorithms can be found throughout the literature (Smith 1977, Ralston 1983, Howe 1989, Dreier 2007, Allerton 2009).

As mentioned earlier, integration algorithms introduce phase lags and a proper choice must be made to minimize this polluting effect in the simulation. Figure 5 illustrates the frequency response of several integration algorithms. The most notable characteristic shown here is the phase distortion introduced by the Euler algorithms. At a vehicle frequency of 6 rad/sec and a sample rate of 10/sec, the normalized frequency parameter is 0.6 and the Euler algorithm induces a phase lag of about 16 degrees.

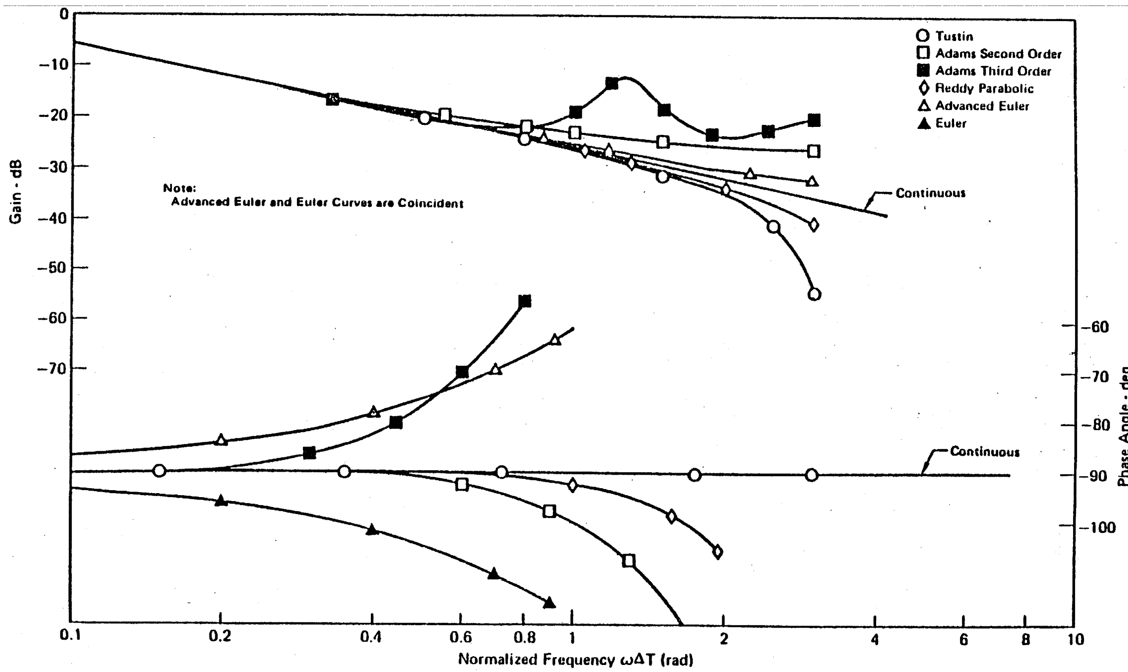


Figure 5
Integrator Frequency Response

When the sample rate is increased to 60/sec, the phase lag for the same vehicle frequency is only about 3 degrees. In many older flight simulators operating at low sample rates, this source of phase lag was overlooked. Pilot complaints about sluggish dynamic response were improperly dealt with by adjusting or 'fudging' damping coefficients elsewhere in the aerodynamic model and satisfactory results were never achieved. The choice of one of the other integration algorithms would have remedied this problem because they do not introduce much phase distortion.

Summary

The processes for computing the EOM for a force and moment model are summarized in figure 6. Forces and moments are summed and accelerations are computed in all six body degrees of freedom. Body angular rates are obtained by integration and then utilized to form transformation matrices to the fixed coordinate systems desired for the model application. Body translational acceleration is integrated to compute velocities. For most aircraft simulations, wind and earth rotation effects are included at this point. The vehicle state parameters are finally computed by integration of the velocities and angular rates and expressed as 3 position and 3 orientation parameters. Additional transformations are then applied to get other parameters needed to support the simulation. The simulation modeler must make some appropriate choices for coordinate systems, transformation methodology, and numerical integration techniques.

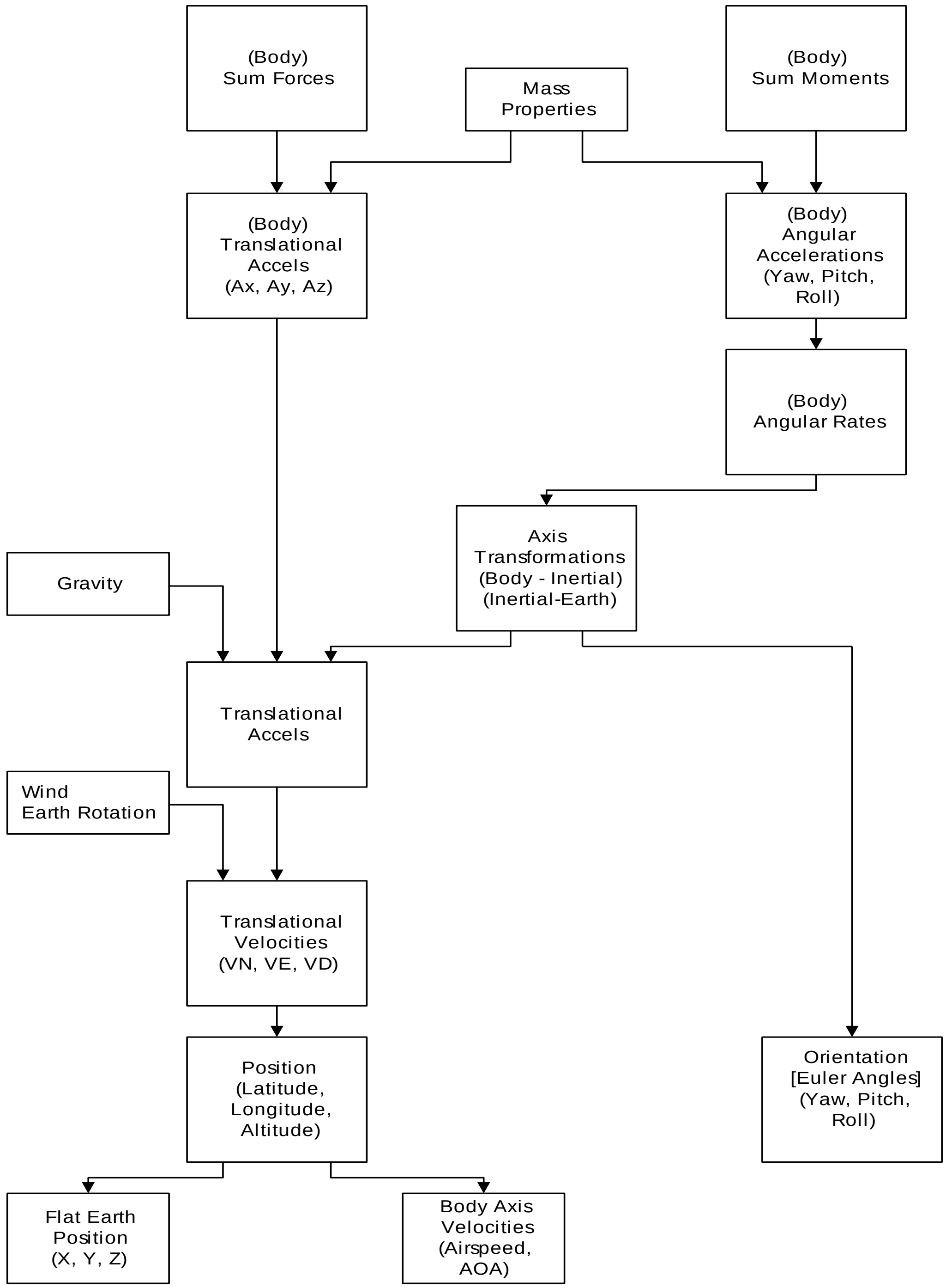


Figure 6
Summary of EOM Process

AERODYNAMIC MATH MODELING

The design goal of a simulator used for flight training is to present the pilot with tasks representative of those he will encounter in the airplane. If these tasks involve learning the flight characteristics of a particular aircraft, then an accurate aerodynamic simulation must be developed to provide meaningful training. A mathematical model of airplane aerodynamics can produce an accurate simulation only if the modeler understands the physical relationship between the components of the model and the characteristics of aircraft in flight. The fundamental components as presented in these lecture notes are illustrated in Figure 7.

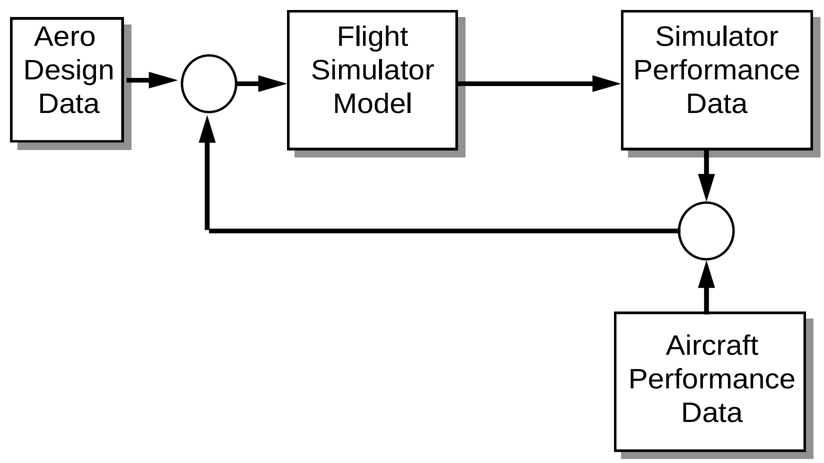


Figure 7
Modeling Components

Aerodynamic math models utilize design data usually obtained from wind tunnel tests which are subject to a number of assumptions and limitations. The creation of an accurate flight simulation requires more than a mere entry of aerodynamic design data into a set of equations. The simulation modeler must understand how to manipulate the model parameters by correlating specific simulator flight characteristics identified by pilot comments and test data with the appropriate terms within the aerodynamic math model. Failure to fully understand this correlation has historically led to "band-aid" software fixes that appear to correct an immediate flight dynamics problem but insidiously cause worse problems somewhere else in the simulation.

Note in Figure 7 that the aerodynamic modeling process consists of not only the model itself but also three distinct data sets. In order to manage the development of a flight simulation model, a clear understanding of the term "data" is required. The creation and validation of a flight simulation model can be considered as a process that connects and reconciles the three data sets shown in Figure 7. The first data set, aerodynamic design data, is necessary to define parameters within the model. The aircraft performance data set is absolutely essential as criteria for validating the model. This is done by comparing it to the simulator performance data set. Each of these data sets come from very different sources and have unique characteristics. Therefore, when the need for "data" is discussed in any flight simulator development program, the specific type of data involved must always be clarified.

These lecture notes present an overview of how aerodynamic characteristics are modeled in flight simulations (the use of design data in the model), and how flight characteristics are identified by specific test techniques (the generation of performance data). The connection or correlation between modeling data and performance test data is described in the Validation and Evaluation lecture notes. The modeling and test data discussion presented herein is based on simplified fixed wing aerodynamics in order to illustrate the concepts involved. The same general concepts can be applied to helicopter simulations. Details of rotary wing aerodynamics and unique test requirements are beyond the scope of this paper but a brief survey is presented. This paper should provide a program manager with an overview of simulator aerodynamic data requirements not only for design data, but more importantly, for flight characteristics data and the corresponding test requirements essential to developing a high fidelity flight simulation.

AERODYNAMIC MODEL CONVENTIONS

The aerodynamic portion of a flight simulator math model computes the aerodynamic forces and moments based on flight conditions and pilot control inputs. These are subsequently summed with all the other forces and moments acting on the airframe in the equations of motion portion of the model. The basic math model structure for a typical six degree of freedom flight simulation is illustrated in Figure 8. Various definitions for model nomenclature and axis systems have been developed as flight simulation applications have evolved. Today, many users recognize the value of common modeling definitions to promote transportability and reusability. A useful guide for achieving commonality is the American National Standard document: "Recommended Practice for Atmospheric and Space Flight Vehicle Coordinate Systems", published by AIAA (ANSI/AIAA 1992).

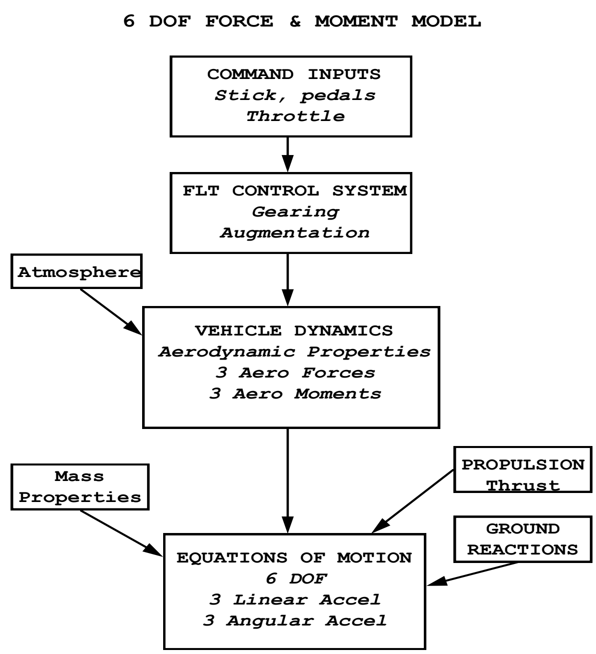


Figure 8

Figure 9 illustrates a conventional right hand axis system originating at the aircraft center of gravity. The longitudinal or X axis is located in a plane of symmetry and is given a positive direction pointing forward or into the wind. A moment about this axis is a rolling moment, L, and the positive direction for a positive rolling moment utilizes the right hand rule (i.e., positive is right wing down). The vertical or Z axis also is in a plane of symmetry and is established positive downward. A moment about the vertical axis is a yawing moment, N, and a positive yawing moment would yaw the aircraft to the right. The lateral or Y axis is perpendicular to the plane of symmetry and is positive out the right side of the aircraft. A moment about the lateral axis is a pitching moment, M, and a positive pitching moment is in the nose up direction. A six degree of freedom aerodynamic model of the aircraft contains three force equations, one for the aerodynamic force along each axis and three moment equations, one for the aerodynamic moment about each axis.

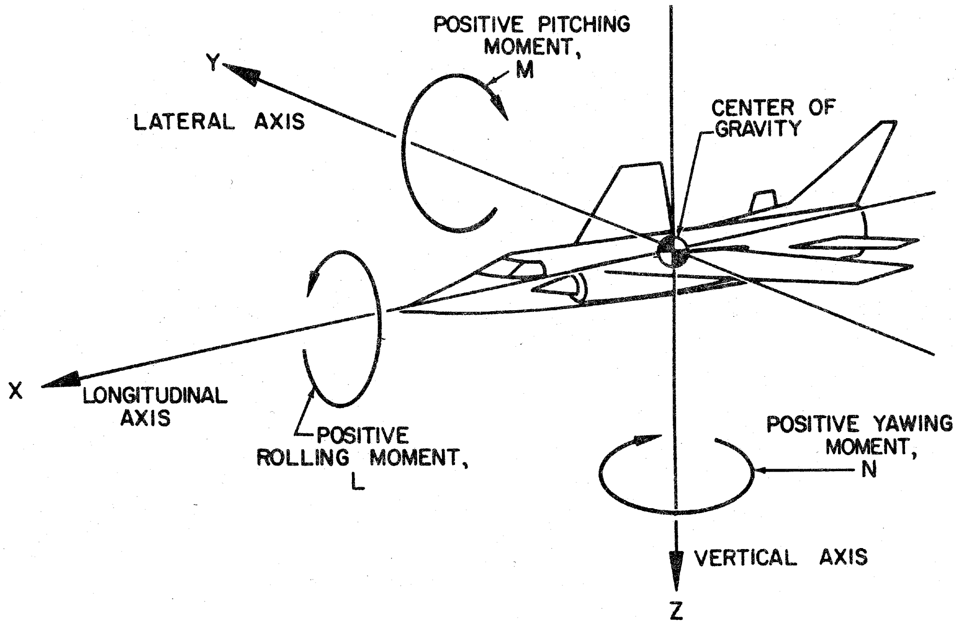


Figure 9
Airplane Body Axis System

There are several coordinate systems used in aerodynamics but the most common ones applied to flight simulations today are the body axis system and the stability axis system. The body axis system illustrated in Figure 9 is a right-handed triad of mutually perpendicular axes whose origin is fixed at the nominal aircraft center of gravity. The three body axes are fixed with respect to the aircraft with the positive X axis directed toward the nose, positive Y directed out the right wing, and positive Z downward. The exact alignment of the X body axis is somewhat arbitrary but simulation models generally use the definition of a fuselage reference line provided in the design data from the airframe manufacturer.

The stability axis system illustrated in Figure 10 is also a right-handed triad of mutually perpendicular axes originating at the center of gravity with the same sign convention as the body axes.

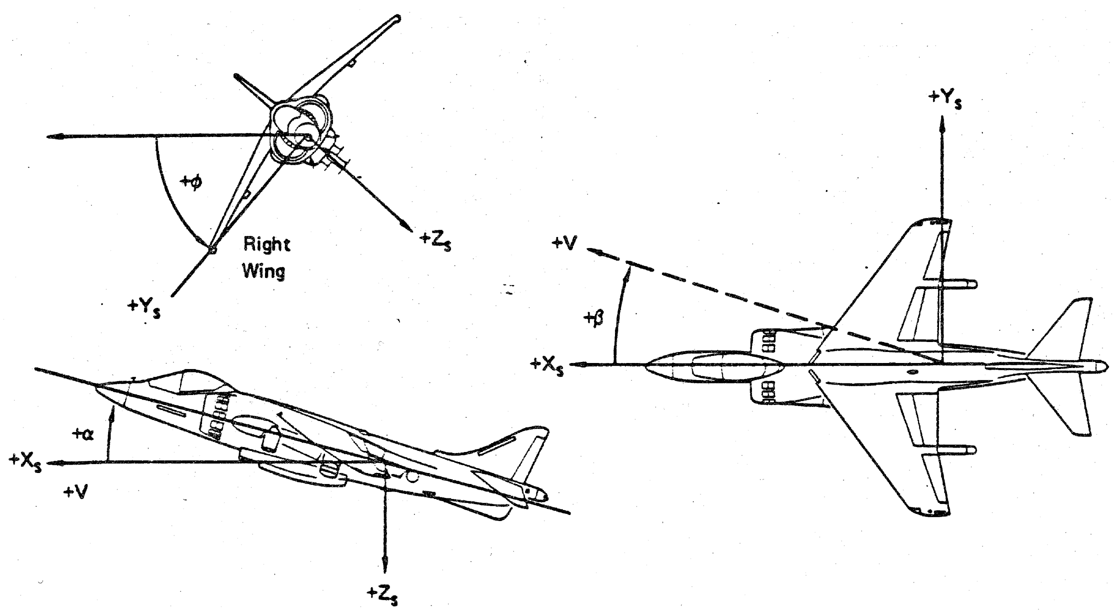


Figure 10
Airplane Stability Axis System

However, the X stability axis differs from the X body axis in that it is inclined in the XZ plane to be aligned with the relative wind. This inclination or angle between the X stability and body axes is defined as the angle of attack (AOA). Since this is a rotation about the Y body axis, the Y stability and Y body axes are coincident. The stability axis system was originally developed to simplify aerodynamic analysis. Typical wind tunnel test results are measured perpendicular and parallel to the relative wind and it has been conventional to calculate stability derivatives from subsonic flow theory with reference to stability axes. The orientation of aerodynamic forces in the stability axis is shown in Figure 11.

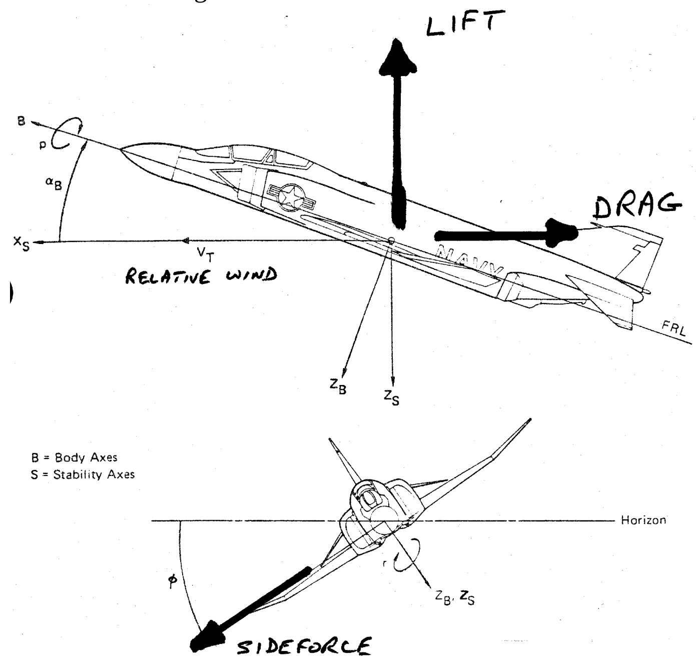


Figure 11
Stability Axis Aero Forces

AERODYNAMIC COEFFICIENTS

The aerodynamic forces and moments produced by an aircraft are the result of airflow over the various components of the airframe, i.e., fuselage, wing, vertical tail, horizontal tail, flaps, or combinations thereof. The customary practice in simulation work is to describe static and dynamic features of a particular aircraft in terms of dimensionless aerodynamic coefficients. These coefficients must be multiplied by the appropriate dimensional factors to obtain the corresponding force and moments for use in the equations of motion. In addition, many of these aerodynamic coefficients are functions of one or more variables such as Mach number, angle of attack, control surface position, etc.

The classical definitions of the aerodynamic coefficients originated in an era in which many of the dynamic effects encountered by modern supersonic aircraft were either unknown or regarded lightly. Moreover, computational tools for the analysis of the equations of motion were limited at that time, consequently every effort was made to reduce the complexity of the equations by simplifying assumptions. This causes some confusion regarding precise definitions of the coefficients and basic limitations on their use. These difficulties are particularly acute in the field of flight training simulators where the whole range of aircraft operating conditions, static and dynamic, must be simulated and the classic aerodynamic coefficients, based on small perturbations about a given steady-state situation, are inadequate. These inadequacies can be overcome by developing additional coefficient terms to suit the phenomenon, such as rotary balance coefficients for spin characteristics. Detailed development of the classical coefficients can be found in the literature (Dommasch 1951, Perkins 1949, Connelly 1958, Anderson 2000, Blakelock 1965, Roskam 1979, Etkin 1982, Stevens 2003, Phillips 2004). The basic idea is that the aerodynamic forces and moments acting on an aircraft in flight may be arbitrarily considered as being products of the dimensionless coefficients Cx, Cy, Cz, Cl, Cm, Cn; dynamic pressure Q; and appropriate geometric terms to satisfy the dimensional requirements. The six body axis aerodynamic equations look like this:

$FX = C_x QS$ = *X axis aerodynamic force*
$FY = C_y QS$ = *Y axis aerodynamic force*
$FZ = C_z QS$ = *Z axis aerodynamic force*
$MX = C_l QSb$ = *aerodynamic moment about x axis*
$MY = C_m QSc$ = *aerodynamic moment about y axis*
$MZ = C_n QSb$ = *aerodynamic moment about z axis*

where:

S = *wing area*
b = *wing span*
c = *mean aerodynamic chord*
$Q = 0.5\, \rho\, V^2$ (Q is dynamic pressure)
ρ = *air density*
V = *aircraft velocity*

Aerodynamic coefficients of necessity describe the forces and moment components in a given axis system and care must be exercised to employ the coefficients in a manner consistent with their definition. Subsonic tunnels use primarily stability axis balance rigs, whereas practically all transonic and supersonic tunnels use internal, body-mounted gauges giving data in body axes. Most tunnel facilities, however, have made provision for converting data from axis system to axis system expeditiously. The use of different reference axes adds to the diversity of meanings possible for a given coefficient. Aerodynamic coefficients are normally written with subscripts to designate the axis and aircraft component associated with the particular term. Greek symbols are commonly used to express angles (angle of attack: alpha, sideslip: beta, control deflection: delta, etc.). Standard nomenclature for aeronautical symbols is defined in the ANSI/AIAA reference.

For convenience, subscripts and Greek letters will not always be used in this text. Instead, these terms will be abbreviated and expressed in much the same way as they appear in software code.

Subsonic coefficients used in simulators are typically provided in the stability axis system. The X and Z forces in this axis system are referred to as drag and lift, respectively, and these data are provided as drag coefficient (CD) and lift coefficient (CL). Drag acts along the negative X stability axis and lift acts along the negative Z stability axis. The stability axis aerodynamic force and moment equations are:

*FXS = -CD*Q*S = stability axis X force*
*FYS = CY*Q*S = stability axis Y force*
*FZS = -CL*Q*S = stability axis Z force*
*MXS = CRS*Q*S*B = stability axis roll moment*
*MYS = CMS*Q*S*C = stability axis pitch moment*
*MZS = CNS*Q*S*B = stability axis yaw moment*
(Note: C_{ls} expressed as CRS to distinguish from CL)

Next, these terms must be transformed to the body axis system for summation with other forces and moments in the equations of motion. This transformation requires a rotation through the angle of attack (AOA) as follows:

*FXB = FXS*cos(AOA) - FZS*sin(AOA)*
FYB = FYS
*FZB = FZS*cos(AOA) + FXS*sin(AOA)*
*MXB = MXS*cos(AOA) - MZS*sin(AOA)*
MYB = MYS
*MZB = MZS*cos(AOA) + MXS*sin(AOA)*

Note that the Y force and moment terms are equivalent in either axis system as explained earlier.

Origin of Coefficients

In general, the dimensionless coefficients CD, CY, CL, Cl (or CR), CM, CN are functions of the steady state flight conditions, the perturbations in linear and angular velocity, and the control positions. The literature explains how each force and moment component can be expanded in Taylor series to develop coefficient terms representing small perturbations about a steady state condition. Therefore, a coefficient such as CLAOA is interpreted as the change in lift coefficient due to small changes in angle of attack. Each small perturbation coefficient is also referred to as a stability derivative because each term is analyzed for its effect on aircraft characteristics.

In a simulator used for piloted flight purposes, the small perturbation restrictions imposed by classic theory are generally too severe. Large excursions of the variables can and do occur and the resultant nonlinear behavior of the forces and moments are an important feature of the aircraft response. The full spectrum of flight conditions must be simulated faithfully over an altitude range from sea level to service ceiling and for every conceivable combination of configurations and control settings.

Simulator Implementation

An approach to the problem of unrestricted simulation of aircraft characteristics is to represent each of the basic aerodynamic coefficients as a function of all of the pertinent variables. To keep the functional relationships manageable, each coefficient is made a function of only the most influential variables affecting its value, such as Mach number or angle of attack, or both. Thus the small perturbation coefficients are "strung together" to collectively represent the flight regime of interest. Secondary effects of other variables are incorporated as a series of independent corrective terms similar to the Taylor's series expansion of the classic theory. To give an adequate representation of these effects over the full range of flight conditions, however, the corrective coefficients may themselves be non-linear functions of one or more variables.

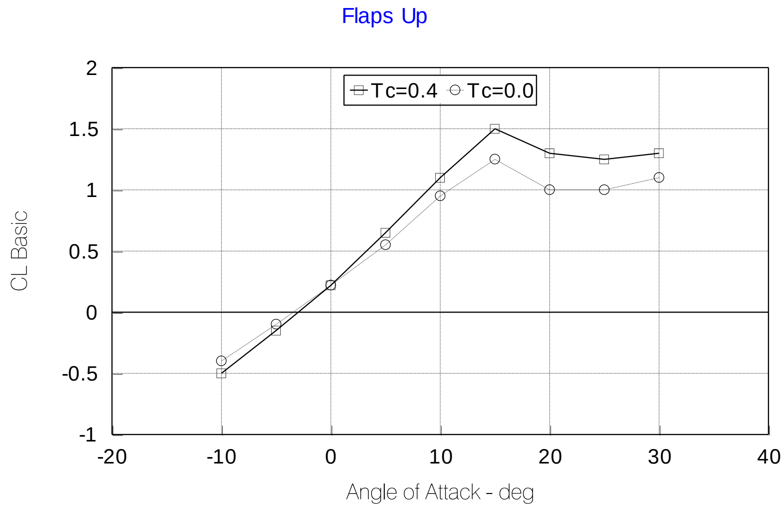
The variables that predominately affect aerodynamic coefficients are angle of attack and Mach Number (ratio of aircraft velocity to the speed of sound). Sideslip angle (defined by the ratio of side velocity to total velocity) has important effects, but since the normal flight condition is with zero sideslip, these effects are generally incorporated as corrective terms. Changes in the effective value of coefficients are also caused by aeroelastic distortion of the aircraft, and these effects are usually presented as a function of altitude. Power effects can also be significant, particularly for propeller driven aircraft due to high energy slipstream effects.

Data Handling

Typical aerodynamic coefficient data are shown in Figure 12, which is a plot of the variation of basic lift coefficient with angle of attack for the T-34C airplane. Note that there are two curves: one each for thrust coefficient at zero and at 0.4 (high power). Also note that while a portion of the data exhibits a linear variation with angle of attack, the overall character of the curves is non-linear. Inexperienced simulation modelers may try to represent aerodynamic data such as this with equations generated by curve fitting routines, usually to reduce any computational burden or complication that might be imposed by using data tables. Data tables are a better choice because the computational burden is minimal with modern systems if properly implemented. More importantly, the tables provide direct access to individual coefficient values, which is a tremendous benefit during model debugging or engineering investigations.

When data sets like the example shown are implemented in the simulator model, a table look-up algorithm must be supplied in the simulation software to interpolate the value for basic lift coefficient as a function of the two variables, angle of attack and thrust coefficient. Most flight simulator aerodynamic models contain a large number of these data tables and the tables can become quite large depending on the nature of the coefficient being represented. A typical example is the OFT for the T-45A jet trainer where the aerodynamic model contains over 110 data tables and many are functions of two or three variables. These tables contain about 100,000 data points. This is not a large number of data points when compared to the original aerodynamic database generated by the airframe manufacturer that contained about 10 times this number of points. The contents of aerodynamic data tables must be carefully managed to maintain the integrity of the model, therefore, careful choices must be made in reducing the data set sizes. Also, aerodynamicists frequently create tables that are functions of three or more variables but it may be desirable for the simulation modeler to reduce the number of variables because the computational implementation may become burdensome. Data table size and look-up algorithm execution time must be considered when selecting a computation system for flight simulation. This example shows that a key task for the simulation modeler is to study the mountain of data generated for the airplane and configure it for practical real time simulation implementation.

Figure 12
T-34C Lift Coefficient Data



Aerodynamic Force and Moment Equations

A basic set of aerodynamic coefficients is presented in the equations listed in Table 1. This basic set would not be adequate for any particular simulation and additional coefficients are usually required for unique airplane characteristics. For instance, the incremental aerodynamic effects of external stores such as fuel tanks, bombs and rockets would be required when appropriate. Also, simulation of any aircraft with reversible control systems would require some additional equations containing hinge moment coefficients in order to model the aerodynamic forces felt at the pilot's controls.

Table 1
BASIC FIXED WING AERODYNAMIC COEFFICIENTS
Stability Axis System

Lift Force

CL = CL(AOA) + CLDE*DE + CLDF*DF
(*or CL0+CLAOA*AOA)*

Drag Force

CD = CD(CL) + CDDF*DF + CDDG*DG + CDDSB*DSB

Side Force

CY = CYBETA*BETA + CYP*P + CYR*R

Roll Moment

CRS = [CRDA*DA + CRDR*DR] + CRBETA*BETA + (CRP*P + CRR*R)*B/2V

Pitch Moment

CMS = [CMDE*DE + CMDF+DF] + CM(CL) + (CMQ*Q + CMADOT*ADOT)*C/2V

Yaw Moment

CNS = [CNDR*DR + CNDA*DA] + CNBETA*BETA + (CNR*R + CNP*P)*B/2V

where:

AOA=angle of attack	*DE=elevator deflection*	*B=wing span*
ADOT=AOA rate	*DA=aileron deflection*	*C=mean aero chord*
BETA=sideslip	*DR=rudder deflection*	*V=true airspeed*
P = roll rate	*DF=flap deflection*	
Q = pitch rate	*DSB=speed brake deflection*	
R = yaw rate	*DG=landing gear position*	

Detailed derivations of the physical basis for each aerodynamic coefficient are covered extensively in the literature. A very brief outline of the physical significance of each of the basic coefficients is presented in Table 2.

Table 2
PHYSICAL BASIS OF AERODYNAMIC COEFFICIENTS

DEGREE OF FREEDOM	COEFFICIENT	PHYSICAL MEANING
Lift	CL0	Lift at zero AOA
	CLAOA	Lift curve slope
	CLDE	Effect of elevator deflection
	CLDF	Effect of flap deflection
Drag	CD: CD0	Drag at zero lift (parasite drag)
	CD: CDCL	Drag due to lift
	CDDF	Effect of flap deflection
	CDDG	Effect of retractable landing gear
	CDDSB	Effect of speedbrake deflection
Side Force	CYBETA	Effect of sideslip (affects crosswind behavior)
	CYP	Effect of roll rate
	CYR	Effect of yaw rate
Roll Moment	CRDA	Roll control power
	CRDR	Roll due to rudder
	CRBETA	Effect of sideslip (a cross coupling term called dihedral effect)
	CRP	Damping due to roll rate (primary effect)
	CRR	Damping due to yaw rate (secondary effect)
Pitch Moment	CMDE	Elevator control power
	CMDF	Effect of flaps
	CM: CM0	Moment at zero AOA (bias term)
	CM: CMAOA	Effect of AOA or CL ("spring" term)
	CMQ	Damping due to pitch rate (primary effect)
	CMADOT	Damping due to AOA rate (secondary term)
Yaw Moment	CNDR	Rudder control power
	CNDA	Yaw due to aileron (usually adverse yaw)
	CNBETA	Effect of sideslip ("spring" term)
	CNR	Damping due to yaw rate (primary effect)
	CNP	Damping due to roll rate (secondary effect)

The purpose of Table 2 is to provide a basic feel for the effect of each coefficient on the behavior of the airplane. It should be noted that typical aircraft pitch and yaw axis responses resemble second order spring mass damper systems as illustrated conceptually in Figure 13. Therefore, the CMAOA and CNBETA terms are equivalent to spring constants and the CMQ and CNR terms are equivalent to damping terms. The roll axis response resembles a first order dynamic system and the roll damping term, CRP, is equivalent to the system time constant.

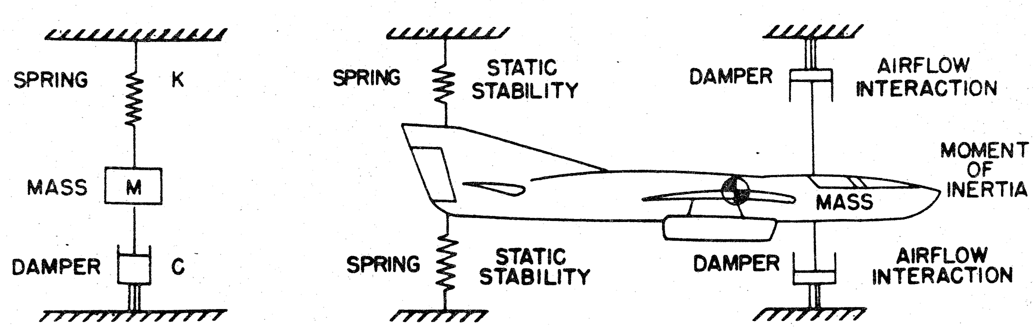


Figure 13
Airplane in Flight is Analogous to Spring-Mass-Damper System

One should not get the impression that the aerodynamic coefficients presented thus far are in universal use in the exact form that they are presented here. For example, it is not uncommon to encounter data for which degrees have been used surreptitiously as an angle measure instead of radians. Stability derivatives can also be expressed in a dimensional form instead of the non-dimensional approach described here. Aerodynamicists exercise a large degree of local autonomy with the result that there is a staggering variety of coefficients currently in use for simulation purposes.

The foregoing treatment of aerodynamic coefficients is not based on the viewpoint of the aerodynamicist or wind tunnel expert, but as an overview for those unfortunate people who are forced to rely on these coefficients in their simulation work. The tacit assumption is made that the aerodynamicists can produce the required aerodynamic data in convenient, consistent form. Usually these data can appear in a number of forms ranging from single constants, linear analytical expressions, to complex multivariate tables or curves. It is up to the simulator modeler to adapt these data formats to his particular software and system complexity requirements.

The accuracy of coefficient data must be considered with caution. Coefficients derived solely from wind tunnel and analytical methods have estimated accuracies from 5 percent for static coefficients like CD and CL to a terrible 50 percent for dynamic coefficients like CNP and CMQ. In the past, once a new aircraft commenced flying, the manufacturer typically became so preoccupied with system integration problems that he neglected to verify in detail by actual flight test the aerodynamic coefficients on which the design was based. Modern aircraft development programs have become highly dependent on flight simulation and so the refinement of coefficients from flight test data is becoming more commonplace. Sophisticated software tools for system or parameter identification are applied to correlate and adjust coefficient values so the model matches flight test results. More will be said of this in the Validation and Evaluation discussion.

HELICOPTERS

Helicopter modeling and data requirements are mentioned briefly here to point out some important differences from fixed wing requirements. Helicopters are modeled differently and additional flight test techniques must be used to identify their unique characteristics.

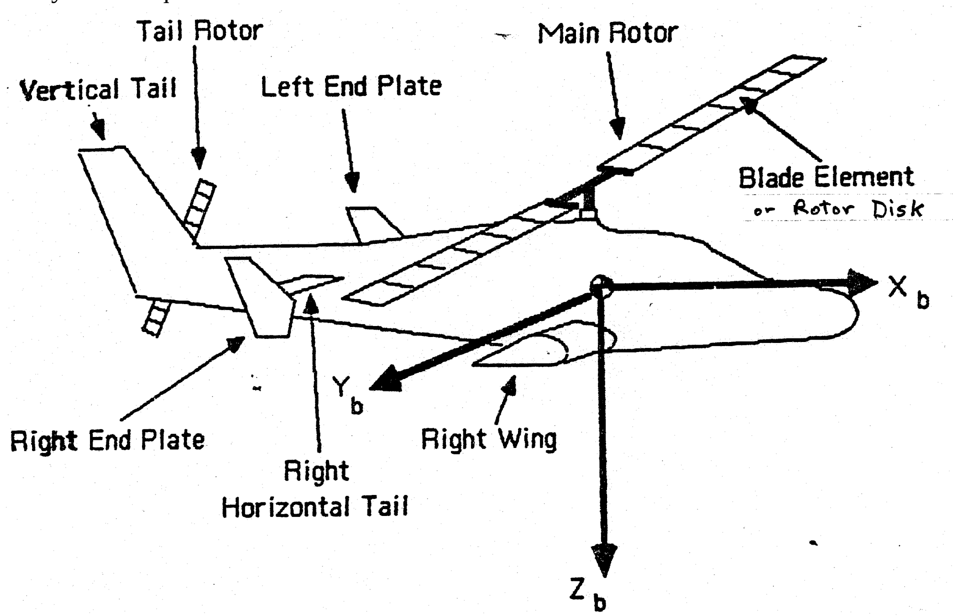


Figure 14
Helicopter Modeling Components

Fixed wing aerodynamic models usually treat the aircraft as a single entity but helicopter models are structured as a collection of parts. As shown in Figure 14, the aerodynamic characteristics of each component (rotors, fuselage, stabilizer, vertical tail, etc.) are modeled separately and then summed to get total body axis aerodynamic forces and moments. Except for the rotors, these data are usually obtained directly from wind tunnel test results. Modeling of the rotor is not straightforward, to say the least, and it presents additional simulation complexities due to such phenomenon as retreating blade stall, unsteady aerodynamic effects, blade elastic modes, non-uniform inflow distribution, and the effects of rotor downwash on the fuselage and empennage components. Dreier 2007 describes the assembly process for a rotary wing simulation model.

The basic theory for rotor aerodynamics can be found in textbooks (Gessow 1967, Prouty 1990). The rotor system aerodynamic forces are functions of not only vehicle speed and angle of attack, but also blade rotational speed and local blade section velocity. Blade section velocity varies with radial distance from the hub and is also dependent upon current blade azimuth position. This results in non-uniform velocity distribution, or inflow, over the area of the rotor disk. To get uniform load distribution and stable flight, each rotor blade must have its angle of attack varied continuously with azimuth; this is referred to as 'cyclic pitch'. In addition, blades are free to flap vertically and to swing back and forth (called lead-lag) about the hub attachment point. Rotor blade pitch angle typically varies along the radius due to a design twist profile and the blades may have elastic bending modes. In forward flight, a reversed flow region may develop on the retreating blades. A stalled condition referred to as vortex ring state may develop on the main rotor in settling flight or on the tail rotor in crosswinds and sideways flight. Compressibility effects occur at the blade tips and the continuous variation of blade angle of attack induces a dynamic aspect to the lift generated that is not typical of steady fixed wing flight. Flight transitions between hover and forward flight exhibit translational lift effects that must be represented somehow to achieve a high fidelity piloted simulation. The rotor autorotation state must also be represented accurately, especially for effective training in simulators.

Rotor aerodynamics modeling in real time is a difficult problem and several methods are used in current simulators. These methods can be classified as four basic types:

1 - Perturbation models (no separate rotor representation)
2 - Rotor disk models (also called Bailey models)
3 - Rotor map models
4 - Blade element models

Perturbation Models. Perturbation models are intended for small deviations about a steady state trim point. There is no distinct model of the rotor system; instead, rotor effects are lumped into the aero coefficients corresponding to the six rigid body vehicle degrees of freedom. This type of model can be made to operate over a range of flight conditions by stringing the sets of coefficients together with table look up routines. Perturbation models are very useful in supporting simulations for conceptual studies that need VTOL-like flight characteristics such as air traffic control automation studies, or highly augmented control system applications (Whalley 1992). Another appropriate application is for tactical players in Weapons System Trainers. Perturbation models will never provide adequate fidelity if man in the loop flying qualities of the unaugmented aircraft are important to the simulation application because too many dynamic elements are missing.

Disk Models. The most common approach applied to training simulators is the quasi-static disk method where the rotor is treated as a whole disk rather than as a set of blades. This eases the computational requirements considerably so that general purpose minicomputers can be utilized. The rotor disk method, commonly called the Bailey model, was first developed in the 1940's. The disk method assumes a uniform inflow velocity over the whole rotor disk and applies momentum theory to determine local velocity distribution. Blade forces are determined by analytically summing the contribution of each blade as a function of azimuth. The process yields analytical expressions for the output parameters of thrust, drag, sideforce, torque, and flapping angles (disk orientation). These are computed as functions of the disk velocity components, advance ratio and inflow ratio, and the control inputs. The major aerodynamic components of the rotor disk are illustrated in Figure 15.

Rotor Map Models. The rotor map approach is another disk method, which uses large data tables that are functions of three variables (advance and inflow ratios, collective position) to model the output parameters. The rotor map is usually derived from the output of a much more sophisticated non-real time model of the rotor system developed by the airframe manufacturer. One advantage of the rotor map over the analytical approach is that data table values could be refined easily during simulator testing to improve static performance fidelity.

This rotor modeling method was a product of Singer-Link and many flight training simulators were delivered with it over the past 20 years (Briczinski 1984).

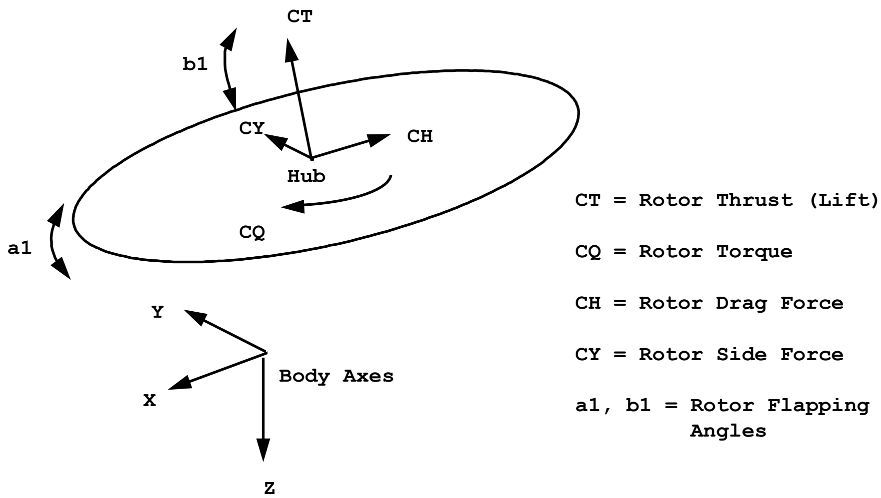


Figure 15
Aero Components of Rotor Disk

Since disk models are quasi-static in nature, explicit dynamic rotor effects such as blade flapping, coning, lead-lag, and elasticity are not accounted for. Therefore, dynamic response of a rotor system is not well represented by disk models. Some of the limiting assumptions include: small flapping and inflow angles, reverse flow region ignored, uniform inflow distribution with no dynamics, negligible compressibility and stall effects. In spite of these limitations, disk models work fairly well for tail rotors since tail rotor flapping frequencies are high enough to be ignored. All rotor disk models typically must be "customized" by the addition of small data tables and special functions to try to alleviate these shortcomings and to try to match flight test results. All rotor disk modeling methods exhibit limited dynamic accuracy which degrades the simulation of maneuvering and transient response characteristics.

Blade Element Models. The blade element method is a physically based modeling approach that offers better dynamic accuracy than the disk methods. Local lift and drag forces are computed by sections along each rotor blade and then summed to get total rotor aerodynamic forces and moments. This technique allows for the inclusion of the combined contributions of aerodynamic effects, blade lead-lag, blade mass properties and inertial effects. Blade flapping acceleration is computed explicitly and integrated to get flapping velocities and angles. Computational requirements are high since each blade is typically broken into five or more elements and the computation interval is 20 or more increments per revolution for each element. In addition, the high frequency content of the flapping dynamics requires a high computation rate. Experience has shown that azimuth step size should be kept below 15 degrees to assure computational stability for the whole operating range of the helicopter rotor system. The high frequency content and the large number of elements and azimuthal increments require a large number of simultaneous computations at a high iteration rate (over 100 Hz). Recall that rigid body aircraft dynamics are typically computed at 30 to 60 Hz. Originally, special purpose processors with parallel computing architecture were required to accomplish this in real time. Modern processors that can handle real time blade element models are available at reasonable cost.

Blade element models were initially developed for evaluating handling qualities at specific flight conditions in engineering simulations because of their dynamic accuracy. GENHEL, developed by Sikorsky in conjunction with NASA, is the classic reference model for implementing the blade element method (Howlett 1981). Several companies have successfully implemented blade element models on training simulators. Two examples are the USMC CH-46E and CH-53E Operational Flight Trainers. General implementation details include: 6 to 10 segments per blade, high iteration rates (180 Hz - equates to 8-10 degrees azimuth step size), and Ada software (later converted to other languages such as C++). The computer hardware initially utilized DSP or RISC technology but were subsequently rehosted on less specialized processors with no difficulty. Basic flight fidelity validation was laborious but straightforward since model parameters were direct representations of physical rotor system components. The Automatic Fidelity Test capability in the CH-53E OFT was very beneficial in expediting the recurring flight testing during the debugging process. Creech & Hildreth 2006 describes the work performed to successfully replace a disk rotor model with a blade element model for the AH-1W Weapon Systems Trainer.

Beyond Blade Element Models - Improvements in computer technology and rotor aerodynamic modeling expertise have made improved dynamic fidelity more readily available in flight simulators. However, more improvements are needed. More sophisticated validation methods based on frequency domain techniques must become more widely utilized so that complex helicopter models can realize their full fidelity potential. A neglected modeling area, especially for training simulators, is dynamic inflow and its impact on vertical response. Dynamic inflow accounts for the fact that the induced velocity change at the rotor does not occur instantaneously. The dynamic lag associated with the acceleration of a large air mass results in an angle of attack (AOA) change at the rotor blade. For a collective input, the AOA perturbation is initially due only to the blade pitch angle change; this initial AOA change is reduced by the change in inflow as the vehicle responds. The net effect is that steady state thrust is lower than the initial peak value. Simulation modelers have attempted to capture this characteristic even within disk models, but many resort to simplistic crude first order time constants that are not realistic. More effective dynamic inflow implementations have been developed as described in Peters 1988 and Dreier 2007. Ballin 1991 describes work at NASA Ames Research Center with both the dynamic inflow modeling and frequency domain validation methods. Other helicopter modelers should benefit from the experience documented in these references. Research at Georgia Institute of Technology with dynamic inflow modeling plus elastic blade modeling is documented in He 1992. This work confirmed the benefits of dynamic inflow modeling but showed only minor benefits from including elastic blade effects.

Another area requiring more development is the simulation of helicopter flight in turbulence, especially in disturbed airwakes downwind of buildings and ship superstructures. Some issues and modeling approaches are discussed in more detail in Costello 1992 and Riaz 1993. Blade element models are capable of sampling flow variations across the disk but local flow field modeling for real time applications lacks the granularity and accuracy to provide representative simulation to pilots. Naval helicopter pilots are obvious beneficiaries of improvements to airwake modeling and research efforts are continuing. The implementation of a complex ship airwake model in a UH-60A blade element simulation at NASA Ames is described in Bunnell 2001. The unsteady (time varying) airwake model was derived using computational fluid dynamics techniques, and the simulation was used in an experiment to define operating limits for the UH-60A with an LHA type ship. The experimental results showed promise and suggest further analysis in the JSHIP program (Wilkinson 2001,VanderVliet 2001).

SUPPORTING AERO MODEL COMPONENTS

Simplified Flight Dynamics Models

In some simulator applications, flight dynamics play a secondary role to the main mission of the simulator. Examples include airborne adversary aircraft for air combat or dynamic flight-like eyepoint movement for visual system data base demonstrations. For these applications, the full 6 degree of freedom (DOF) aero force and moment model depicted in Figure 8 is too cumbersome to implement and operate, therefore, some simplified representation of flight dynamics is desired. Some typical simplified fixed wing model structures are described below along with typical applications.

5 DOF Performance Model - Air combat simulators must provide a multitude of adversary aircraft with realistic performance characteristics. Therefore, adversary models must be structured to conserve computer assets and easily utilize different aircraft data sets. The flight control functions are essentially eliminated so that computer algorithms can command the desired maneuvers as directly as possible. Figure 16 illustrates this type of model, sometimes referred to as a 5 DOF Performance Model. Note that the input commands are simply bank angle, normal acceleration (g), and airspeed. The vehicle dynamic response is based on the aerodynamic parameters significant to air combat maneuvering: lift, drag, thrust, and angular rate limits. Since sideforce is usually not included, the vehicle exhibits 5 DOF motion assuming coordinated turns. Body axis yaw rate can be computed via the following relationship: $r = (g \cos\theta \sin\phi) / V$

Modeling data required for these types of model include aircraft mass characteristics lift and drag aero coefficients, angle of attack limits, and angular rate limits. Engine data needed are thrust, fuel consumption, and spool-up dynamics. Validation data requirements are primarily concerned with performance tests such as accelerations, decelerations, climbs, sustained and instantaneous g capability and turn performance, corner velocity, and fuel consumption. A common problem with developing this type of model is reducing the number of coefficient terms while preserving essential flight fidelity. Research efforts described in Anderson/VPI 1993 demonstrated that validation data such as specific excess power (typical criteria for aircraft threat models) can be applied to work backward to define the essential model parameters using commercial systems identification software tools. This method is an efficient way to produce models that are practically self-validating.

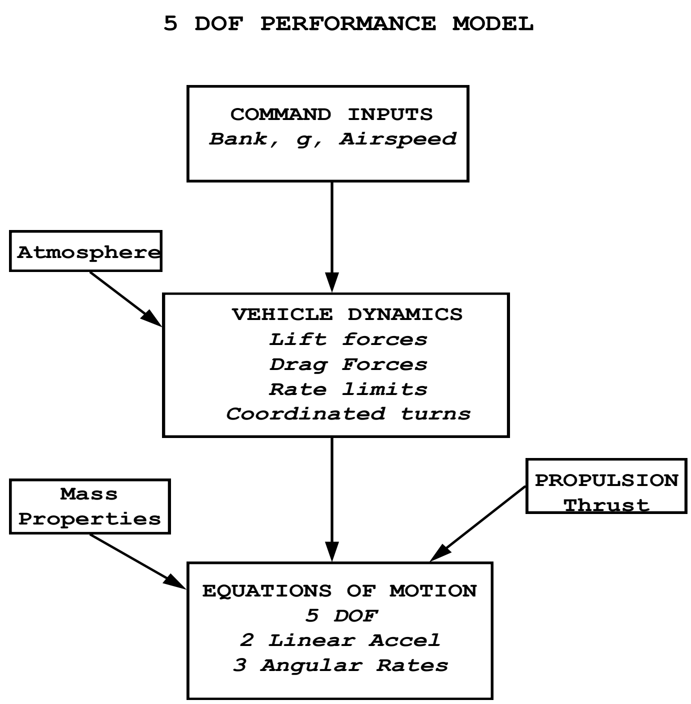


Figure 16

5 DOF Rate Model - (Also called a Kinematic Model) Simulators that include tasks such as formation flight and air-to-air refueling must drive an airborne moving model in a realistic fashion. For this application, the aerodynamic forces and moments can be eliminated to simplify the model as shown in Figure 17. Control inputs can be provided by a joystick or by software algorithms. The vehicle dynamics module simply imposes

rate limitations representative of the modeled aircraft and body axis yaw rate is derived from the coordinated turn relationship described above. These rate models are very compact and convenient for driving secondary airborne moving models but they should not be used for pilot-in-the-loop situations because the simplified handling qualities do not feel realistic to experienced pilots. Aircraft data are needed to identify the appropriate rate and airspeed limits.

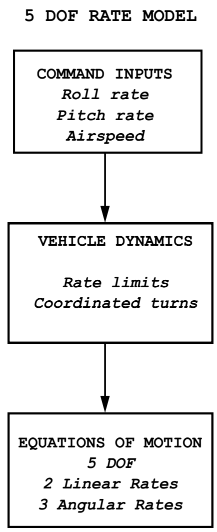


Figure 17

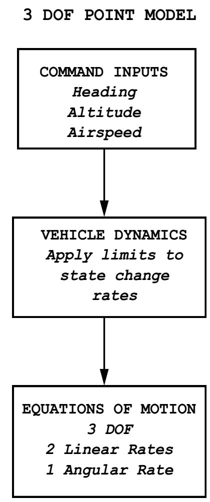


Figure 18

<u>3 DOF Point Mass Model</u> - Simulation of airborne vehicles such as beyond-visual-range targets can be accomplished with very simple models consisting of a "point mass" where vehicle heading, altitude, and airspeed are the only parameters of concern. As shown in Figure 18, these parameters are the command inputs and the vehicle dynamics module simply applies realistic limits to changes in these commanded states. Modeling data consists of rate limits, particularly the variation of turn rate with altitude and airspeed since heading changes are commanded directly by the control algorithm. Validation data is needed to verify turn rates, airspeed limits and climb rates. These models are sufficient as long as aircraft attitude is not important such as for basic targets for detection by long range radar. More sophisticated vehicle models must be utilized when attitude sensitive factors such as radar cross section and infrared signature must be included in the target aircraft characteristics.

Other Model Components

The flight simulation model structure illustrated in Figure 8 contains some other components besides the flight controls and vehicle dynamics that should be mentioned.

<u>Atmosphere</u> - The basic atmospheric parameters, temperature, pressure, and density, are generated in this model. The ICAO 1976 Standard Atmosphere Model is a commonly used reference or basis for the equations implemented here. Other atmospheric phenomenon are also modeled here: winds, gusts, turbulence, wind shear, storm cells, and airwakes (or burble) due to freestream flow obstructions such as ships, formation or tanker aircraft, and land features. Modeling of these phenomena ranges from extremely crude to very sophisticated and a careful review of the literature is recommended before utilizing any given model. Some of the issues pertaining to flight simulation for ship operations are summarized in Galloway 1990. Wind shear

encounter during takeoff and landing operations is a critical issue, especially for commercial airline operations. The FAA defines simulation test criteria and provides wind field modeling data in FAA AC 120-41.

Engines - A full thermal cycle modeling approach is generally too complex for most real time flight simulator applications. Therefore, real time engine models usually are structured to match the manufacturer's steady state performance data via table look-up routines and the transient response is modeled to emulate the function of the fuel controller and to ensure that cockpit engine instruments exhibit the correct response. Rolfe 1986 describes this approach in more detail.

Ground Reactions - The interaction of aircraft landing gear with the landing surface is a very complex and very dynamic phenomenon. The model must accommodate sudden large force changes, tilting surfaces, various surface friction conditions, and steering and braking effects. Model dynamic parameters and the model iteration rate must be structured to ensure that computational stability is maintained. Modeling of ground reactions is typically hampered by lack of data on the dynamic characteristics of key components (struts, tires) and on the effects of pilot steering and braking techniques.

Modeling Tools

A number of software products are on the market today that support model development and rapid prototyping for all types of vehicle dynamics applications, not just automobiles. These modeling tools feature graphical user interfaces usually with a core set of basic predefined functions appropriate for the application, e.g., for control systems tools are provided for integration, summing, rate limiting, backlash, etc. The user assembles the complete model in building block fashion and links all the components. The modeling tools include data gathering and analysis capabilities that facilitate debugging and design choices during both the build up and final testing process. A “code generation” feature can then convert the model into a transportable higher order language such as C, Ada, or FORTRAN. Construction of vehicle dynamics models can progress at a very rapid pace with these tools but users should always be aware of all assumptions and limitations applied to each and every one of the ‘building blocks’ contained in the modeling tool. Recently, the developers of a modeling environment for helicopters found that basic assumptions for such things as axis rotation and rotor hinge construction had to be revisited every time a new and different type of helicopter was modeled on the system. It is hoped that model tool developers have been through many iterations such as this and applied lessons learned to produce robust modeling aids.

GROUND VEHICLE MODELING

At first glance, ground vehicle modeling may seem to be less difficult than aircraft modeling since ground vehicles are constrained by contact with a surface. Surface vehicle travel is largely a two dimensional relationship. If simple ground vehicle movement is all that needs to be represented in a simulation then a relatively simple kinematic model will suffice. However, if a more detailed representation of ground vehicle dynamics is desired, the model most address ground contact which generates normal and frictional forces between vehicle and terrain that tend to produce high frequency vertical components of motion. These vertical forces act through complex mechanical assemblies such shock absorbers, tire wheels, and suspension systems (Figure 19) and they typically displace the whole vehicle in small, independent amounts in the vertical direction.

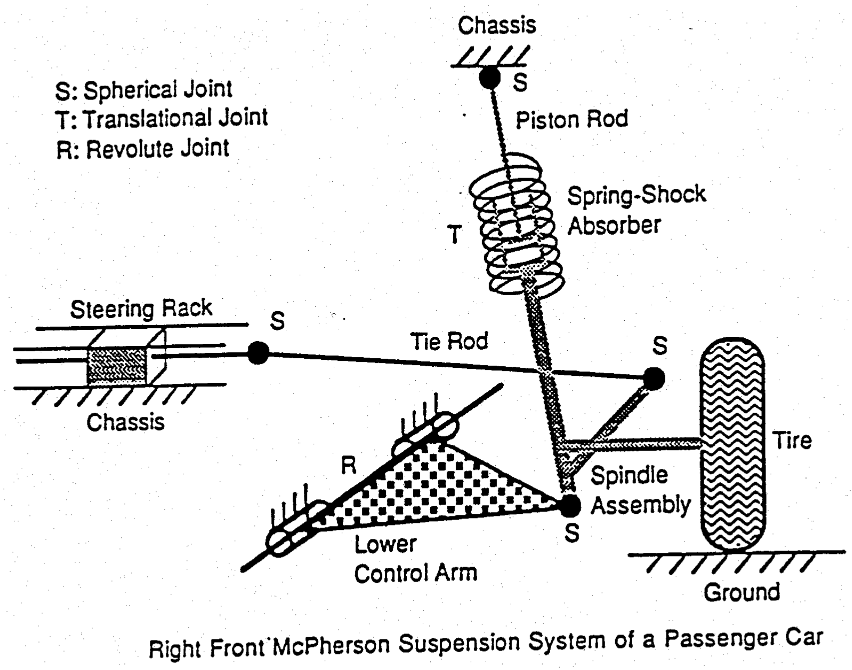


Figure 19

Accounting for these forces makes ground vehicle modeling significantly more complicated than fixed wing aircraft. In the past, ground vehicle dynamics were modeled by writing force and moment equations for each element in the system, summing these forces and moments, then solving for accelerations and integrating. Developing these equations and then implementing them in computer code was a tedious process, especially for systems with large numbers of degrees of freedom (more than six).

Current practice is to use so-called multi-body codes where the various bodies are described by their parameters and the modeling tool writes the equations and generates the computer code, in any of several high order languages.

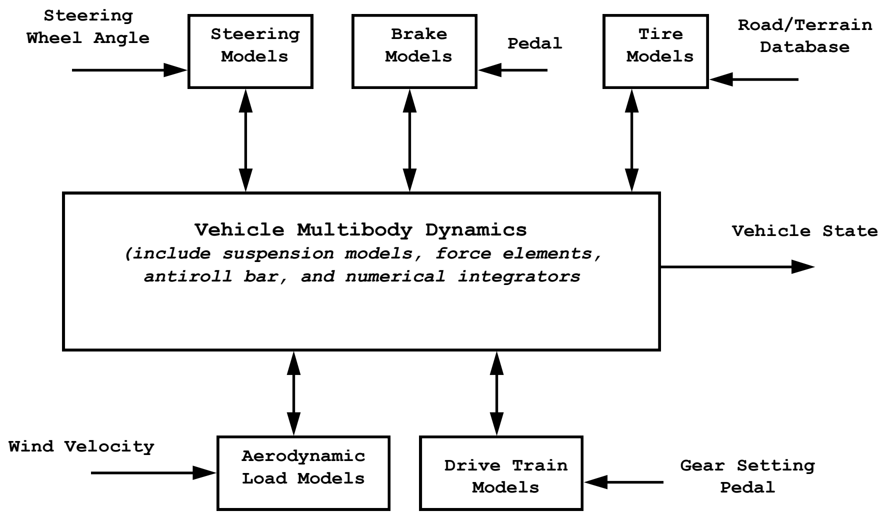


Figure 20
Vehicle Subsystem Modeling

These tools generate executable code that will run in real time for driver in the loop simulation. Two of these tools are AUTOSIM, developed by the University of Michigan Transportation Research Institute (UMTRI), and Real Time Recursive Dynamics (RTRD), developed by the University of Iowa. These modeling tools can be used for any multibody simulations or real time control applications such as robotics.

AUTOSIM has been used at UMTRI to generate simulations of automobiles up to 18 degrees of freedom (Sayers 1991 & 1993). These include: six rigid body (chassis) degrees of freedom, four degrees of freedom for the front and rear suspension systems, four slip angle degrees of freedom and four wheel spin degrees of freedom. Tank models have been developed using AUTOSIM with up to 21 degrees of freedom: six rigid body, one turret, and 14 road wheel degrees of freedom.

The RTRD is used on the IOWA Driving Simulator (IDS), which is a driver in the loop simulator. The RTRD has been used to develop a 14-body model of the HMMWV and a 15-body A-car simulation. A point of contact for RTRD is Dr. John Kull at the University of Iowa. RDRT is a fundamental part of the National Advanced Driving Simulator that is being built at the University of Iowa for the U.S. Department of Transportation National Highway Traffic Safety Administration.

SHIP MODELS

The nature of modeling surface ship dynamics will be mentioned briefly here. For simple applications, surface ship motion is typically represented with two-dimensional kinematic equations in the same manner as for simple ground vehicles. That is, kinematic relationships are utilized for longitudinal velocity and yaw(heading). When the surface ship is a large aircraft landing platform in a flight simulator application such as a CV or LHA, then the kinematic equations are embellished to add deck pitch, roll, and heave dynamics. These dynamics are typically generated by sum of sines equations where the amplitude and phasing for each axis is set to match landing deck characteristic motion as a function of sea state. Smaller landing platforms such as destroyers and frigates apply these sinusoidal drivers to five axes: pitch, roll, yaw, heave, and sway.

Man-in-loop simulations for controlling ships have essentially the same structure as flight dynamics models except that vertical motion is primarily determined by very large counterbalancing values of weight and displacement (buoyancy). Both hydrodynamic and aerodynamic forces must be considered as well as combined environmental influences of water current, wave action, and wind. An extensive body of knowledge exists [Fossen 94, Brutzman 94] and further discussion is beyond the scope of these notes.

FLIGHT TEST DATA

The aerodynamic math models in flight simulators utilize coefficients that are normally developed analytically during the preliminary design process to predict aircraft characteristics. These coefficient data have reasonable accuracy for steady state lift and drag estimates but are relatively inaccurate for dynamic characteristics due to limitations in wind tunnel measurement accuracy and assumptions based on small linear perturbations. To overcome these limitations, the fidelity of simulator flight characteristics must be validated by comparison with aircraft flight characteristics data. Once this comparison is made, various analytical techniques can be applied to adjust the coefficients in the simulator math model to improve the fidelity. The best source of validation or criteria data is directly measured aircraft flight test data.

Classical Flight Test Methods

All aircraft are designed to meet certain mission requirements, and in the case of military and commercial aircraft, these requirements can be very specific. For instance, tactical military aircraft intended for air-to-ground attack missions must meet certain performance requirements for payload, speed, fuel consumption, and maneuverability. In addition, it must exhibit satisfactory flying qualities so as not to degrade the pilot's ability to accomplish his mission. That is, that portion of the pilot's workload associated with just flying the airplane must be small in proportion to more significant mission tasks such as launch, enroute navigation, weapons delivery, evasive maneuvering, aerial refueling and recovery. The ultimate standard for judging an airplane's mission suitability is the opinion of the professional pilot after he has flown and thoroughly evaluated the airplane in appropriate mission scenarios. However, professional pilot opinion must be substantiated by quantitative measures to establish if certain contract performance guarantees were met and to provide guidance to engineers in correcting any deficiencies. Flight test data are particularly important for sorting out the effects of complex aerodynamic phenomenon so that the true cause of a piloting problem can be identified. Flight test techniques have been developed to properly document significant airplane flying qualities and performance characteristics and the basic techniques are described in reference manuals such as those prepared by the military test pilot schools (USNTPS & USAF: current updated editions)). Variations in test techniques are developed when necessary to test unique aircraft features (e.g., vectored thrust, use of computer generated inputs) or to enhance safety of flight.

These flight test techniques are equally valid for assessing the characteristics of flight simulators. In this manner, pilot opinion of simulator flight fidelity can be substantiated in engineering terms. The ideal situation is to use aircraft test data for direct comparison to identical tests in the simulator so that differences between the airplane and simulator will be readily apparent. It is important for simulator professionals to understand the nature of flight test techniques and data so that they can be used effectively to evaluate and correct deficiencies in a simulator aerodynamic math model. There must be continuous liaison between the flight test data generators (engineers and pilots) and the data users (simulator engineers) throughout a simulator development to ensure that proper data are generated, correctly interpreted, and correctly applied in the simulator validation tests.

Airplane flight testing falls into two broad categories: <u>performance</u> testing and <u>flying qualities</u> (or stability and control) testing.

<u>Performance</u> testing is concerned with characteristics resulting from the airframe and powerplant combination. The aerodynamic lift and drag characteristics of the airplane generally define the power or thrust requirements at the various conditions of flight while the powerplant generally defines the power or thrust available.

<u>Flying qualities</u> testing is concerned with those **stability** and **control** characteristics that influence the ease of safely flying the airplane during steady and maneuvering flight while executing mission tasks.

The stability of an aircraft is defined by its static and dynamic stability characteristics. Aircraft **static stability** is the characteristic tendency of an aircraft to return to a particular equilibrium condition after having been disturbed from that condition. **Dynamic stability** is determined by the motion characteristics after the disturbance occurs.

Controllability involves the ability to disturb the equilibrium condition and the ability to change from one equilibrium condition to another. To a pilot, stability and controllability are closely related and a balance of the two is desirable so that the aircraft will tend to remain put when not intentionally being maneuvered and yet easily maneuvered when so desired. Airplane and simulator handling qualities are the manifestation of several items, most notably all of the stability derivatives used in the aerodynamic math model, the flight control system characteristics, the aircraft inertia properties and center of gravity location, and many cases, the power setting of the powerplant(s). In addition, transport delays introduced by digital processing in all simulators and in some aircraft flight control systems can have significant effects on handling qualities.

Performance Tests

A typical set of performance tests provides the basis for the mission planning data which appears in generalized form in an airplane's flight handbook. Flight simulators are expected to replicate these performance characteristics in order to provide representative pilot training for planning takeoff and climb performance, cruise fuel consumption, maneuvering performance for tactical aircraft, stall airspeeds, and landing performance. Therefore, it is important to understand the pilot techniques required to perform these tests in the airplane and simulator in order to obtain valid, repeatable results.

There are three basic test conditions in which the pilot will operate his airplane while doing performance testing. Each test condition requires special flight techniques and utilizes different primary flight instruments for pilot reference. These conditions are stable equilibrium, unstable equilibrium, and non-equilibrium test conditions. The equilibrium test conditions represent tests in which there is no rate of change of acceleration in any direction. A stable equilibrium condition is one in which, if the aircraft is disturbed, it will return to its initial condition. An unstable equilibrium point is one in which, if the aircraft is disturbed, it will continue to diverge from its equilibrium test point. A non-equilibrium test indicates a condition in which there is a rate of change of acceleration along some flight axis.

A stable equilibrium test point represents a condition where altitude, thrust (or power), and flight path angle are constant and acceleration is zero. Stable equilibrium data points are obtained in both level and turning flight when operating in the stable portion of the thrust or power required curve. The test technique for obtaining stable equilibrium data is to adjust altitude first, power second, and then wait till the aircraft stabilizes at the equilibrium flight speed. It is important to point out that altitude must be maintained precisely at the desired test level and that power must not be adjusted once set. The primary flight instruments for pilot reference when obtaining data points under stable equilibrium conditions are altimeter, vertical speed, heading for straight flight and bank for turning flight. In airplanes equipped with automatic flight control systems that incorporate attitude, altitude and heading hold modes, stable equilibrium data points are more easily obtained by using these modes. An example of stable equilibrium testing includes <u>speed/power</u> tests where the airplane is trimmed at various airspeed/altitude combinations to determine thrust required and fuel consumption.

Unstable equilibrium data points are more difficult to obtain but can be obtained very rapidly if proper technique is followed. For the unstable equilibrium data points, indicated airspeed is held constant. Altitude, engine RPM, and bank angle may be adjusted as required by the test being conducted. Unstable equilibrium data points are associated with the unstable portion of the thrust or power required curve where power must be increased to stabilize at slower airspeeds. To obtain data points under these conditions, the exact test airspeed is established first. Throttle is then adjusted in order to climb or descend to the desired test altitude. The vertical speed indicator is an important instrument in achieving equilibrium conditions and the pilot also utilizes the airspeed indicator and heading for straight flight or bank angle for turning flight. Automatic flight control systems offer little advantage over manual control in obtaining unstable equilibrium data points. These test conditions are encountered in tests of high performance military aircraft for thrust limited <u>level turn performance</u> and for <u>speed/power</u> data in the landing configuration.

Non-equilibrium test points are usually the most difficult to obtain. They preclude the pilot having stable conditions or being able to trim to maintain any desired condition. The pilot does, however, have some schedule which he can follow and which he can use to assist him in correcting to achieve a satisfactory flight path or flight test condition. Some non-equilibrium tests such as <u>acceleration</u> and <u>deceleration</u> runs are performed at a desired constant altitude. Others, such as climbs, are performed according to a desired flight schedule of airspeed or Mach number versus altitude. Stall airspeeds are determined via a very slow deceleration (less than 1 knot/second). The primary reference instruments for non-equilibrium tests will be dictated by the specific test being performed. Automatic flight control system equipment can be a great aid in obtaining non-equilibrium condition data, but the degree to which it can be employed will depend upon the specific test and the ability of the hold modes to perform their design functions. Good heading hold and altitude hold modes are extremely valuable in obtaining level acceleration and deceleration data. Climb and descent tests can be aided using Mach or airspeed hold modes if they can maintain the desired schedule accuracy (+/- 5 kt or +/- .01 Mach). Availability of these autopilot capabilities in flight simulators (or equivalent driver routines) is extremely beneficial for testing to enhance data repeatability and provide pilot relief for an otherwise tedious and lengthy test process.

Flying Qualities Tests

As mentioned earlier, the airplane must possess a certain measure of both stability and controllability in order to exhibit satisfactory flying qualities. The optimum "blend" depends on the total mission of the airplane. A certain degree of stability is necessary if the airplane is to be easily controlled by a human pilot. However, too much stability can severely degrade the pilot's ability to perform maneuvering tasks. The attainment of an optimum blend of stability and controllability is the goal of the airplane designer. When the optimum blend is attained, flying qualities greatly enhance the ability of the pilot to perform the intended mission.

The airplane is a dynamic system, i.e., it is a body in motion under the influence of forces and moments producing or changing that motion. In order to investigate the motion of the airplane, it is necessary to establish first that it can be brought into a condition of equilibrium, i.e., a condition of balance between opposing forces and moments. Then the stability characteristics of the equilibrium condition can be determined. The airplane is statically stable if restoring forces and moments are created which tend to restore it to equilibrium when disturbed from equilibrium. Thus, static stability characteristics must be investigated from equilibrium flight conditions, in which all forces and moments are in balance. The direct in-flight measurement of certain static stability parameters is not feasible in many instances. Therefore, flight test methods measure parameters which only give indications of static stability. However, these indications are usually adequate to establish conclusively the mission effectiveness of the airplane and are more meaningful to the pilot than the numerical value of the stability derivatives. Typical static stability measures include the amount of control displacement and control force required to stabilize at flight conditions offset from the existing trim state. While static instability about any axis is generally undesirable, if not completely unacceptable, excessively strong static stability about any axis may degrade controllability to an unacceptable degree. For some pilot tasks such as dive bombing, neutral static stability may actually be desirable because of the increased controllability which results. The optimum level of static stability depends on the mission of the airplane.

The pilot makes changes from one equilibrium flight condition to another through one or more of the airplane's modes of motion which characterize dynamic stability. These changes are initiated by excitation of the modes by the pilot and terminated by suppression of the modes by the pilot. These modes of motion may also be excited by external perturbations. Dynamic stability characteristics are measured from nonequilibrium flight conditions during which the forces and moments acting on the airplane are not in balance. The characteristics of the modes of motion of the airplane determine its dynamic stability characteristics. The most important characteristics are the frequency and damping of the motion. The frequency of the motion is a measure of the "quickness" of the airplane. The damping of the airplane modes of motion has a profound effect on flying qualities. If too low, the airplane motion is too easily excited by inadvertent pilot control inputs or by atmospheric turbulence. If too high, the airplane motion following a control input is slow to develop and the pilot may describe the airplane as sluggish. The mission of the airplane again determines the optimum dynamic stability characteristics. However, the pilot always desires some level of positive damping of all the airplane's modes of motion.

Static and dynamic stability prevent unintentional excursions into dangerous ranges (with regard to airplane strength) of dynamic pressure, normal acceleration, and sideforce. The stable airplane is resistant to deviations in angle of attack, sideslip and bank angle without action by the pilot. These characteristics not only improve flight safety, but also allow the pilot to perform maneuvering tasks with smoothness, precision and a minimum of effort.

Controllability may be defined as the capability of the airplane to perform at the pilot's wish, any maneuvering required in total mission accomplishment. The characteristics of the airplane should be such that maneuvers can be performed precisely and simply with a minimum of pilot effort. The pilot's opinion of controllability is shaped by several factors. The most apparent of these factors are the initial response of the airplane to a control input and the total attitude change that results. In addition, the cockpit control forces and deflections required to accomplish necessary pilot tasks are extremely important. These factors depend on the static and dynamic stability of the airplane and the characteristics of the flight control system. The complexity or degree of difficulty that the pilot encounters during maneuvering tasks is directly dependent on the stability characteristics of the airplane.

The nature of aircraft behavior makes it convenient to separate the test and analysis of longitudinal flying qualities from lateral-directional flying qualities. Longitudinal flying qualities involve response exhibited in the aircraft's plane of symmetry. This plane of symmetry divides the airplane into two essentially symmetrical halves and contains components of motion only along the X and Z axes and about the Y axis. Airplane motion in the plane of symmetry, i.e., longitudinal and pitch motion, generally results in insignificant motion about the lateral (X) or directional (Z) axes. On the other hand, motions about the lateral and directional axes tend to be tightly coupled together but they do not usually excite significant motion about the longitudinal axis. Therefore, longitudinal flying qualities are generally investigated apart from lateral directional flying qualities.

Longitudinal Tests

Longitudinal flying qualities must be investigated from equilibrium and nonequilibrium flight conditions. From equilibrium flight conditions, the static longitudinal stability characteristics may be determined. These characteristics are:

<u>Static Longitudinal Stability</u> - Variation of longitudinal control forces and elevator positions with airspeed variations from trim in unaccelerated flight (longitudinal control force and elevator position stability).

<u>Maneuvering Longitudinal Stability</u> - Variation of longitudinal control forces and elevator positions with normal acceleration at a constant airspeed (longitudinal maneuvering stability, or "stick force per g" and "elevator position per g"). Also of interest is the variation of normal acceleration with angle of attack at a constant speed.

There are two longitudinal modes of motion that are suppressed in equilibrium flight. These modes of motion exhibit the <u>dynamic</u> <u>longitudinal stability</u> characteristics of the airplane and they are determined from nonequilibrium flight conditions. The longitudinal modes of motion are called the "airplane <u>short</u> <u>period</u>" and the "<u>long period</u>" or "phugoid" motions, and the parameters of interest for each mode are the frequency and damping of the characteristic motion.

<u>Short Period Mode</u> - usually characterized by oscillatory angle of attack and pitch attitude motions whose periods range from 2 to 4 seconds. This mode can be excited either by a one cycle sine wave elevator control input called a doublet or by a step elevator input. The short period mode influences the pilot's opinion of initial airplane response to his control inputs and external disturbances.

<u>Long Period Mode</u> - characterized by oscillations in airspeed and altitude which appear as roller coaster type flight path excursions with periods on the order to 40 to 60 seconds. The long period mode affects the pilot's ability to maintain long term trim conditions in cruising flight and it is usually excited for test purposes by temporarily displacing the elevator to establish a specified deviation from trim airspeed.

Lateral-Directional Tests

Lateral-directional flying qualities are also investigated from equilibrium and nonequilibrium flight conditions. From equilibrium conditions, such as a steady heading sideslip, the static lateral-directional characteristics may be determined. These characteristics are:

1. Variation of directional control forces and rudder positions with sideslip angle in steady heading flight at a constant trim airspeed (static directional stability characteristics).

2. Variation of lateral control forces and aileron positions with sideslip angle in steady heading flight at a constant trim airspeed (static lateral stability characteristics or dihedral characteristics).

3. Variation of bank angle with sideslip angle in steady heading flight at a constant trim airspeed (sideforce characteristics).

Dynamic lateral-directional flying qualities are investigated from nonequilibrium flight conditions. This requires study of the characteristics of the three lateral-directional modes of motion - the Dutch roll mode, the spiral mode, and the roll mode - which are suppressed in equilibrium flight. Two of the lateral-directional modes differ from the longitudinal modes in that the pilot does not usually deliberately excite the Dutch roll or spiral modes. Excitation of these modes is not required to maneuver the airplane under normal flight conditions. However, the Dutch roll and spiral modes are continually inadvertently excited by the pilot or by external perturbations. Therefore, the characteristics of these modes greatly affect the pilot's opinion of the airplane during all phases of mission accomplishment.

Dutch Roll Mode - a second order response generally involving both lateral and directional motion, the characteristics of this mode to be investigated are the frequency and damping of the motion, the relative magnitude of the lateral part of the motion to the directional part of the motion, or simply, the "roll to yaw ratio", and degree of excitation of the Dutch roll mode during uncoordinated, aileron only turns.

Spiral Mode - a first order motion which may be convergent, divergent, or neutral. That characteristic of the motion is investigated as well as the time required for the amplitude of the first order motion to double or half.

Roll Mode - deliberately excited by the pilot via lateral stick inputs in order to make bank angle changes required in all phases of mission accomplishment. The characteristics of the roll mode have a significant influence on the pilot opinion of the airplane. The roll mode is an essentially first order response and is usually heavily damped. Therefore, the characteristics of the roll mode to be investigated are: the roll mode time constant; steady state roll rate obtainable with various lateral control inputs; and the nature and amount of yawing motion generated during rolling maneuvers.

Other Tests

These are additional flying qualities tests, which do not easily fit in the categories described above, and some of these will be mentioned now. The flight control system mechanical characteristics and rates of operation of the secondary controls (trim, flaps, speedbrake, etc.) must be evaluated because of their inherent influence on flying qualities. Steady state trim angle of attack and control settings are very useful for simulator validation and can be easily obtained whenever equilibrium flight conditions are established such as during speed/power performance tests. Trim changes associated with aircraft configuration changes (landing gear, flaps, power, etc.) mostly affect longitudinal characteristics and tests are conducted for both open loop (hands off) and closed loop (hold altitude or airspeed constant) response. The flying qualities of the airplane about all axes during approaches to and beyond aerodynamic stall are investigated to reveal any unsafe characteristics. Evaluations of airplane spin characteristics are conducted only with extra safety precautions and only if essential to mission effectiveness. Multi-engine aircraft are tested for minimum control airspeeds with failed engines under static and dynamic conditions. Ground handling characteristics (steering and braking) plus the ability to lift the nose wheel during takeoffs and hold it off in ground effect during landings are also important parts of flying qualities evaluations.

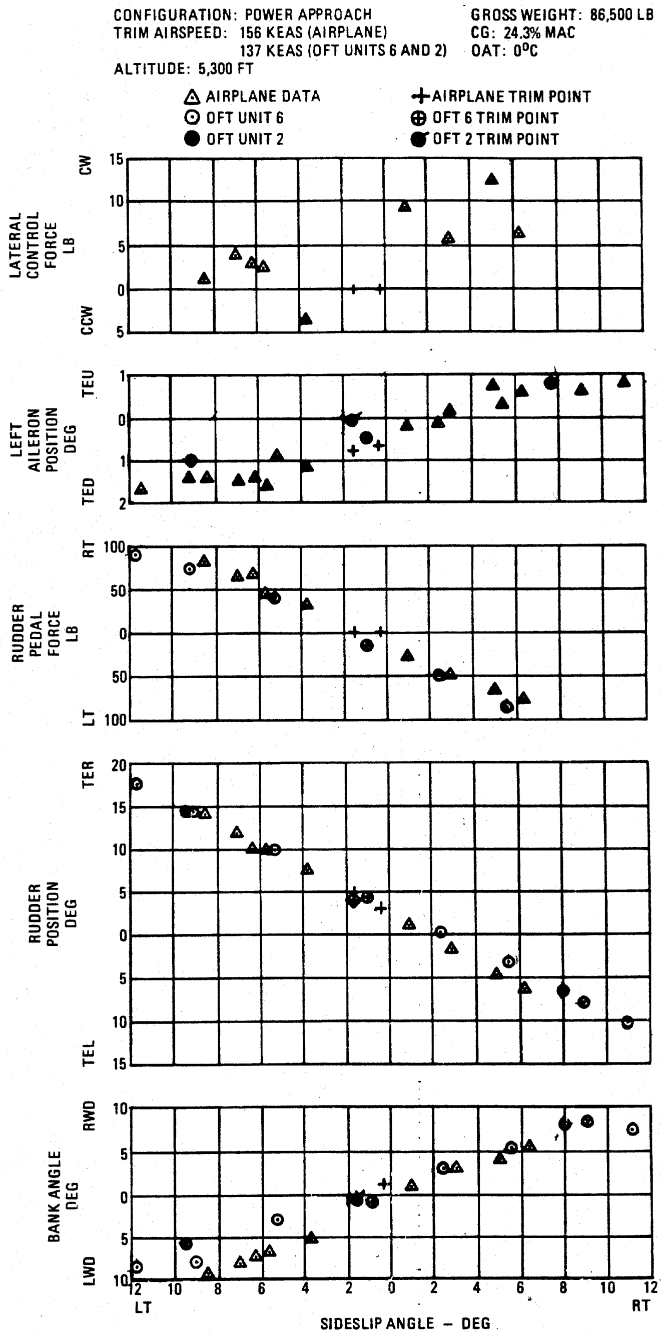
Flight Test Data for Simulators

A typical test matrix outlining the minimum requirements for such a program is presented in Table 3. These tests are conducted at normal operating weights and center of gravity locations and at airspeed/altitude conditions distributed evenly throughout the flight envelope. Flying qualities tests are conducted as appropriate with control augmentation and/or stability augmentation on and off. Test aircraft are usually equipped with special flight test instrumentation to record the necessary parameters. This is desirable especially for unstable vehicles like helicopters. However, if such instrumentation is not available, a competent flight test team using portable test devices or handheld measuring devices such a tape measures can obtain a useful amount of data, force gauges and stop watches. In fact, variations between similar airplane types, particularly in control system mechanical characteristics, make it almost imperative to do some tests in several aircraft in order to substantiate average airplane characteristics. The use of handheld and production cockpit instrumentation provides considerable flexibility in going from one cockpit to another and allows for rapid data generation should significant data gaps appear during simulator testing.

Table 3
TYPICAL FLIGHT FIDELITY TEST MATRIX
Test Conditions: Normal Operating Conditions

TEST	PARAMETERS OF INTEREST
Control System Mechanical Characteristics	Breakout, including friction, centering, freeplay, oscillations, rates of operation
Static Trim Points	Trim settings, power settings, AOA
Level Accelerations, Decelerations	Time history of airspeed and fuel used; speedbrake in/out for decelerations
Climb Performance	Rate of climb, fuel consumption
Longitudinal Stability a. Static b. Maneuvering	Longitudinal stick position and force gradients, AOA gradients; flight path (rate of climb) stability in landing configuration (static only)
Dynamic Longitudinal Stability	Frequency and damping of short period and phugoid modes; time histories of small step and sinusoidal inputs
Trim Changes	Effects of landing gear, flaps, power changes, speedbrakes, runaway trim (open/closed loop)
Longitudinal Control Effectiveness	Nose wheel lift-off, ground effects
Static Lateral-Directional Stability	Sideslip, bank angle, rudder and aileron control positions and forces in steady heading sideslips
Dynamic Lateral-Directional Stability	Frequency and damping of Dutch roll mode; character of spiral mode
Lateral Control Effectiveness	Full and partial deflection rolls; time histories of bank angle, roll rate, sideslip
Stall Characteristics	Variation of airspeed, buffet, control force and position, rate of climb/descent in slow decels
Asymmetric Power	Static and dynamic minimum control speeds
Mission Tasks	Observe dominant characteristics especially during closed loop tracking tasks; engine dynamic response; ground handling characteristics

A typical set of flight test data is shown in Figure 21 to illustrate two important points about test data. First, the test conditions must be thoroughly documented. The header contains enough information to recreate this particular test, but a simulation modeler usually needs more details, such as trim angle of attack, all trim control and power settings, etc, to analyze any fidelity problems when trying to match these data. Such additional information is sometimes hard to get unless the flight test program is aware of this need and endeavors to capture it. The second point is that typical flight test data exhibits a fair amount of scatter. Good engineering judgment is required to interpret the scatter and decide which data points to accept and which should be ignored. This engineering judgment can only be derived from knowledge of the test techniques used and previous experience in judging what is important for matching simulator performance to the simulator design purpose.



STATIC LATERAL-DIRECTIONAL STABILITY
(Configuration PA)
P-3C Airplane, BuNo 160290
and
P-3C OFT, Device 2F87(F) (Units 6 and 2)

Figure 21
Sample Flight Test Data

Flight Test Data for Helicopters

Rotary wing flight test techniques are similar to fixed wing tests for forward flight characteristics. However, additional tests are required for such characteristics as hover performance and control response, trim control position in low speed flight (in all directions), critical azimuth for tail rotor authority, and vertical climb performance, as shown in Table 4. A typical flight test program for helicopter simulator validation is outlined in reference 22. Flight test techniques for helicopter performance and stability and control are described in U.S. Navy Test Pilot School Manuals (USNTPS 1987 & 1991).

Table 4
CRITERIA DATA UNIQUE TO ROTORCRAFT

- *ENGINE*
    - *START/STOP/ROTOR ENGAGE*
    - *GOVERNOR CHARACTERISTICS*

- *SLOW SPEED PERFORMANCE & FLYING QUALITIES*
    - *SIDEWARD FLIGHT*
    - *REARWARD/ FORWARD FLIGHT*
    - *CRITICAL AZIMUTH*

- *IN-FLIGHT PERFORMANCE*
    - *HOVER*
    - *VERTICAL CLIMB*
    - *TRIM CONTROL POSITIONS*
    - *POWER EFFECTS*

- *STABILITY & CONTROL*
    - *CONTROL RESPONSE (ALL 4 AXES)*

- *ROTOR CHARACTERISTICS*
    - *AUTOROTATION*
    - *BLADE STALL*
    - *POWER SETTLING*
    - *VIBRATION*
    - *GUST RESPONSE*

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