MINISTRY OF EDUCATION AND SCIENCE OF UKRAINE
National Aviation University

FUNDAMENTALS OF FLIGHT CONTROL THEORY

Aircraft Automatic Flight Control System Calculation

Term Paper Method Guide
for the students of direction
0914 “Computer-aided automatics and control systems”

Kyiv 2009

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F97 **Fundamentals of flight control theory. Aircraft Automatic Flight Control System Calculation**: Term Paper Method Guide / Authors: V.O. Apostolyuk, O.S. Apostolyuk. – Kyiv: NAU, 2009. – 30 p.

Step-by-step calculation of aircraft flight control systems based on the developed in Simulink non-linear model of the aircraft longitudinal dynamics is provided by this method guide.

These methodological recommendations are intended for students of all specialities of direction 0914 “Computer-aided automatics and control systems”.

F97 **Основи теорії управління польотом. Розрахунок системи автоматичного польоту літака**: Методичні рекомендації до виконання курсової роботи / Автори: В.О. Апостолюк, О.С. Апостолюк. – К.: НАУ, 2009. – 30 с. (англійською мовою)

Подано методичні рекомендації щодо організації та виконання розрахунків систем автоматичного польоту літака на основі моделювання динаміки повздовжнього руху літака у програмному середовищі Сімулінк.

Для студентів спеціальностей, які навчаються за напрямком підготовки 0914 «Комп’ютеризовані системи, автоматика і управління».

INTRODUCTION

High quality research and developments must always be tested either by the experimental study or at least on highly realistic numerical model. The problem of creating such models becomes even more crucial in flight control systems development since experimental studies could be very expensive or even impossible to implement. From this point of view, the development of a non-linear aircraft flight dynamics model, instead of conventional linear transfer function approach, is viewed to be important for many different branches of modern aerospace developments. Such a model then becomes an essential component of the subsequent calculation of the aircraft automatic flight control systems.

Term Paper Goal

There are two major tasks to be completed while performing this term project research:

- Development of a non-linear numerical model of the specific aircraft longitudinal dynamics by using Simulink/MATLAB.
- Calculation of the following aircraft automatic flight control systems: damping, stability, airspeed, and altitude.

Apart from that, student will learn behaviour of the aircraft during uncontrolled flight and how the flight parameters are affected by the respective control loops.

Term Paper Statement

The title of the course work is *“Flight Control Systems Calculation of XYZ”*, where XYZ stands for the name of the chosen aircraft.

Term paper comprises the following specific tasks (corresponding contents sections are given in parentheses):

1. Give the aircraft technical description, specifications, views and projections.
2. Estimate aircraft moment of inertia.
3. Calculate harmonic approximation parameters and produce plots of the aerodynamic coefficients.
4. Assemble and describe the longitudinal flight simulation model in Simulink.

5. Implement, simulate, and choose the best in terms of the corresponding transient process performances gain coefficients for the following control loops: damping, stability, airspeed, and altitude control. Each system must be tested both *with and without wind* disturbances.

All the listed above tasks are performed for the specific aircraft, which is chosen either by the student or issued by the professor.

Term Paper Structure

Term paper structure strongly corresponds to the mentioned above tasks of the term paper statement as follows:

1. Aircraft description
2. Moment of inertia estimation
3. Aerodynamic coefficients
4. Flight simulation model
5. Damping control system
6. Stability control system
7. Airspeed stabilisation
8. Altitude stabilisation
9. Conclusions

Total volume of the term paper is not limited but recommended to be kept below 40 pages.

First section must contain brief description of the chosen aircraft along with its technical specifications and views. For example, plenty of the aircrafts descriptions are available at the www.airwar.ru web-site. Last section, (“Conclusions”), must summarise achieved performances in the developed control systems.

Let us now consider other structural elements of the term paper in greater details along with the performing guidelines.

1. GUIDELINES FOR MOMENT OF INTERTIA ESTIMATION

Apart from the aircraft mass, its inertia in rotational motion, namely moment of inertia I_y, is a very important parameter for the accurate flight simulation. Unfortunately, it is rarely known due to the obvious difficulties in its direct measurement. However, certain approximations can be obtained using the given below approaches. The simplest approach would be to approximate the whole aircraft as a *cylinder* with the corresponding length **h** and diameter **d** shown in Fig. 1.1.

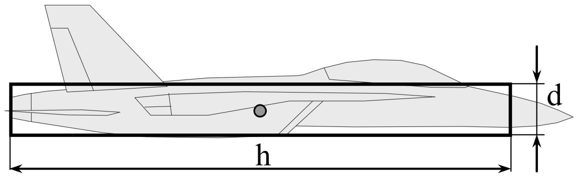

Fig. 1.1. Cylindrical approximation

Needles to say that centres of gravity of the aircraft and the approximation cylinder must coincide. In this case moment of inertial around Y axis is

$$I_y = \frac{m}{12}(\frac{3}{4}d^2 + h^2)\,. \tag{1.1}$$

Here m is the aircraft mass.

More accurate approximation can be obtained by introducing more approximation shapes, such as parallelepipeds, cones, trapezoids for wings, empennage, engines, etc. Moment of inertial approximation can be then done in the steps described below.

1. Calculate volumes V_i of all approximating shapes.
2. Calculate averaged density of the aircraft as

$$\rho_a = m / \sum_i V_i\,, \tag{1.2}$$

where m is the total aircraft mass.

3. Calculate masses of every approximation shape

$$m_i = \rho_a V_i\,. \tag{1.3}$$

4. Calculate moments of inertia I_{0i} of every shape around its own axis of symmetry, which is parallel to the aircraft body-fixed axis Y.
5. Transform each moment of inertia to the body-fixed coordinate system

$$I_{yi} = I_{0i} + m_i r_i^2\,. \tag{1.4}$$

Here r_i is the distance from the body-fixed axis Y to the axis around which corresponding moment I_{0i} is calculated.

6. Calculate total moment of inertia

$$I_y = \sum_i I_{yi}\,. \tag{1.5}$$

The better shapes approximate the aircraft, the better approximation of the moment of inertia will be obtained. Formulae for the shape's volume and moments of inertia could be easily found in corresponding textbooks on mechanics or strength of materials [1].

Finally, if two aircrafts have similar shapes, although different in size and mass, then their moments of inertia are related as

$$I_2 = \frac{m_2 r_2^2}{m_1 r_1^2} I_1\,. \tag{1.6}$$

Here m_i are masses of aircrafts, r_i are the characteristic sizes perpendicular to the axis, around which the moment of inertia is estimated. For example, for the case in Fig. 1.1, r equals to h.

Appropriate method of moment of inertia calculation is chosen based on the level of credibility of the available source data: either shape of the aircraft, or known moment of inertia of some other aircraft, similar in shape.

2. AERODYNAMIC COEFFICIENTS CALCULATION

Essential part of an aircraft flight simulation is proper calculation of its aerodynamic properties. The only reliable way to obtain such characteristics is to perform direct measurement in a wind tunnel. In this case all the data could be used in simulations via the standard interpolation blocks, located in the *Simulink / Lookup Tables* sub-library. Unfortunately, this kind of data is seldom readily available and therefore some other means of aerodynamic properties representation must be used. One should also note, that in order to build really useful flight simulation model, these properties must be defined for every possible incidence or sideslip angles.

The following harmonic representation of the drag and lift coefficients is found to be accurate for all possible values of incidence angles [2]:

$$\begin{aligned} C_D(\alpha) &= d_0 + d_1\cos(2\alpha) + d_2\cos(4\alpha)\,, \\ C_L(\alpha) &= l_0 + l_1\sin(2\alpha) + l_2\sin(4\alpha)\,. \end{aligned} \tag{2.1}$$

Here d_i and l_i are some approximation coefficients based on different kinds of data available, which could be then identified.

When aerodynamic coefficients for different incidence angles are measured experimentally and the resulting data is available in a form of an array, then apart from the direct usage of the data via interpolation, any non-linear data fitting algorithm could be used to identify coefficients d_i for drag and l_i for lift. For example, function *FindFit* from *Wolfram Research Mathematica* will do the job.

In case drag and lift plots as functions of incidence angle are available, then the required for the approximation data points could be taken from the corresponding graphs (see Fig. 2.1 and 2.2).

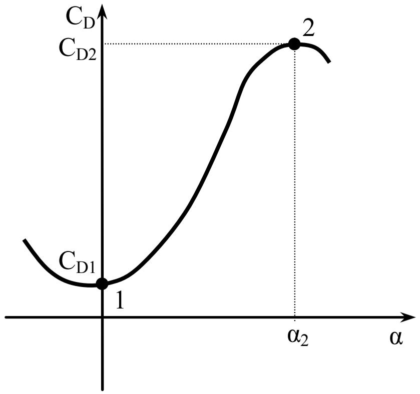

Fig. 2.1. Drag coefficient

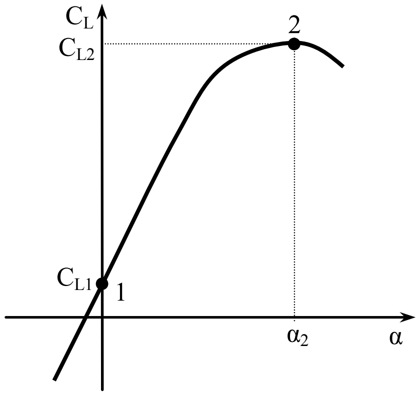

Fig. 2.2. Lift coefficient

Here point **1** corresponds to the incidence α=0, and point **2** is usually chosen at the peak of the function. Note that two chosen points must not be close to each other, since this would significantly reduce the approximation accuracy. With respect to the given points on the graphs and taking only two terms in the approximations (2.1), solutions for the drag and lift coefficients are:

$$d_0 = \frac{C_{D2} - C_{D1}\cos(2\alpha_2)}{\cos(2\alpha_2) - 1}, \quad d_1 = \frac{C_{D1} - C_{D2}}{\cos(2\alpha_2) - 1},$$
$$l_0 = C_{L1}, \quad l_1 = \frac{C_{L2} - C_{L1}}{\sin(2\alpha_2)}. \qquad (2.2)$$

Finally, if the coefficients of the following conventional linear representation of drag and lift are known [3]

$$C_D(\alpha) = C_{D0} + C_{D1} C_L^2 \alpha,$$
$$C_L(\alpha) = C_L^\alpha (\alpha_0 + \alpha),$$

then these coefficients can be used in a straightforward manner to calculate coefficients of the corresponding harmonic representation (2.1)

$$d_0 = C_{D0} + \frac{(C_L^\alpha)^2}{2} C_{D1}, \; d_1 = -\frac{(C_L^\alpha)^2}{2} C_{D1}$$
$$l_0 = C_L^\alpha \alpha_0, \; l_1 = \frac{C_L^\alpha}{2}. \qquad (2.3)$$

Using coefficients (2.2) or (2.3) in representations (2.1) allows modelling of an aircraft aerodynamics for arbitrary incidence angles with sufficiently high accuracy.

3. GUIDELINES FOR FLIGHT SIMULATION MODEL IMPLEMENTATION

In order to implement flight simulation model of the aircraft MATLAB version 7.0 and Aerospace Blockset 1.6 are required (other versions would require minor modifications).

The Aerospace Blockset™ is a collection of block libraries for use with Simulink. The blockset extends Simulink by providing core components for wide range of aerospace systems [4].

The topmost model of an aircraft is shown below in Fig. 3.1. Here the sub-system “Graphs” is considered later. The sub-system “Aircraft” content, which is responsible for the modelling of the aircraft dynamics and all control systems, is shown in Fig. 3.2. Content of the “Aircraft Dynamics” sub-system is given in Fig. 3.3. Block “3DoF (Body Axes)” is the standard motion equations block from the Aerospace Blockset studied earlier. “Engine System” sub-system is just an envelope for the Turbofan Engine System block.

Modelling Aerodynamics

Simulink model for calculating main body aerodynamic forces (“Main Aerodynamics”) and moment for longitudinal motion is shown in Fig. 3.4. Similar diagram in case of the elevator forces and moments is somewhat different (see Fig. 3.5).

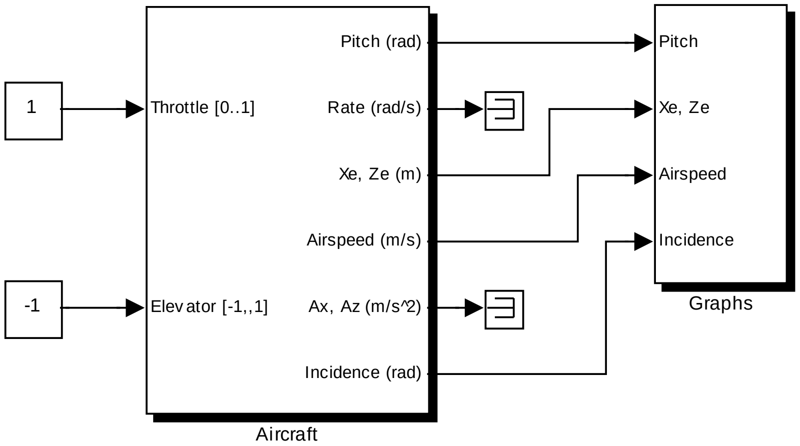

Fig. 3.1. Top level model

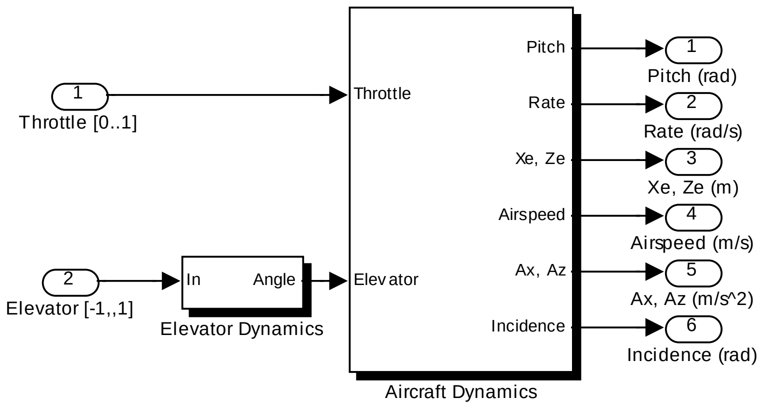

Fig. 3.2. Implementation of the “Aircraft” sub-system

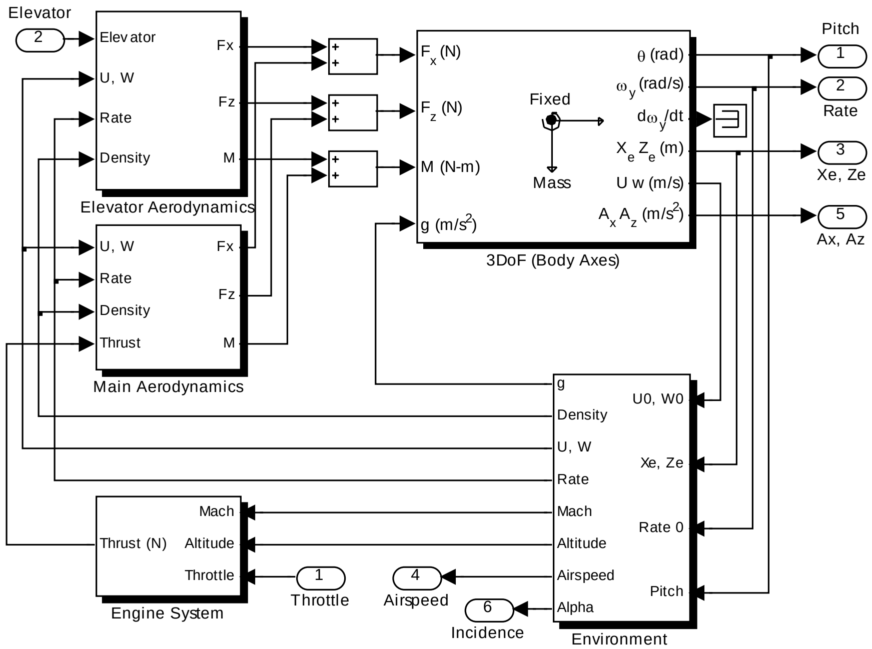

Fig. 3.3. “Aircraft Dynamics” sub-system implementation

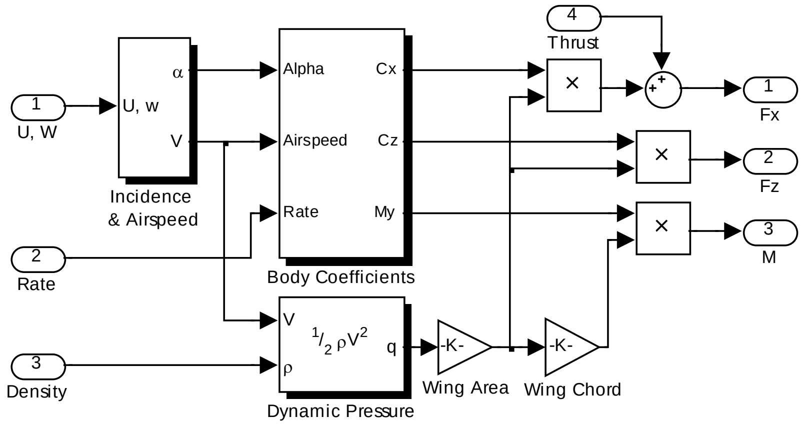

Fig. 3.4. “Main Body Aerodynamics” sub-system

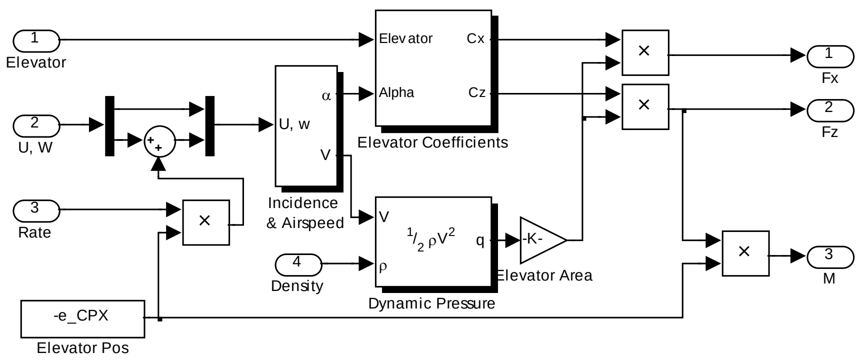

Fig. 3.5. “Elevator Aerodynamics” sub-system

Here *Elevator* is the angle of the elevator deflection given in radians, and e_CPX is the X coordinate of the elevator centre of pressure in the body-fixed frame, assuming that $\vec{r}_e = \{e\_CPX, 0, 0\}$.

Coefficients calculation sub-systems are shown in Fig. 3.6 and 3.7.

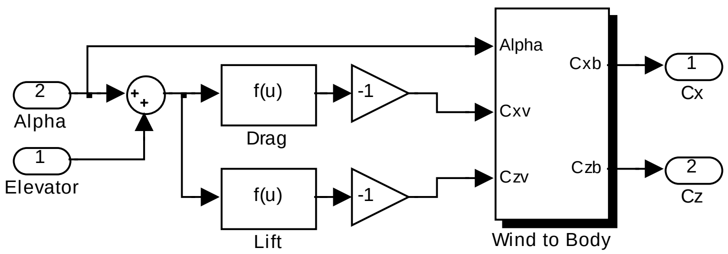

Fig. 3.6. “Elevator Coefficients” sub-system

Standard blocks *Fcn* (stands for “Function”) located in the *Simulink / User-Defined* Functions sub-library implement calculation of the *Drug* and Lift coefficients according to the formulae (2.1). Parameter *Expression* is

```
x_D0 + x_D1*cos(2*u) + x_D2*cos(4*u)
```

Here x_D0, x_D1, and x_D2 – are the variable holding values of the corresponding coefficients in the drag representation.

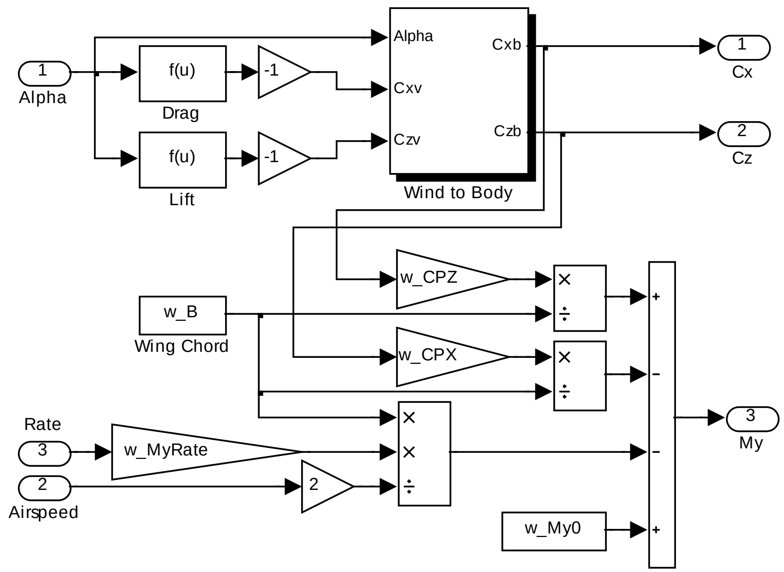

Fig. 3.7. “Body Coefficients” sub-system

Similarly to the drag coefficient, lift coefficient is implemented as

```
x_L0 + x_L1*sin(2*u) + x_L2*sin(4*u)
```

Here x_L0, x_L1, and x_L2 – are the variable holding values of the corresponding coefficients in the lift representation (2.1). Note, that parameter prefix “x” must be later replaces with “w” for the “wing” and with “e” for the “elevator”.

Transformation from wind coordinate system to body-fixed is represented by the “Wind to Body” sub-system shown in Fig. 3.8. Calculation of the “Dynamic Pressure” is demonstrated in Fig. 3.9.

Modelling Environment

Environment model usually includes atmosphere, gravity, and wind disturbances. Specific implementation is shown below in Fig. 3.10.

Atmosphere and gravity models are the standard blocks in the *Environment* sub-library of the Aerospace Blockset. Let us now have a close look at the “Wing Models” sub-systems included in this implementation (see Fig. 3.11).

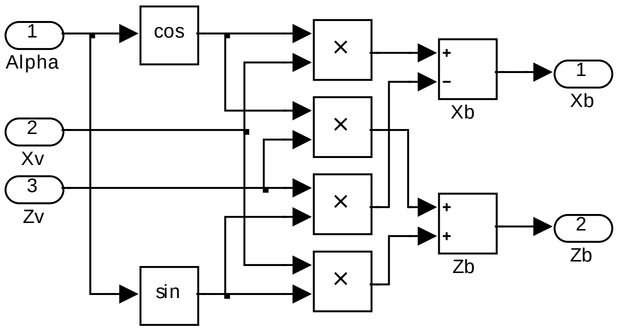

Fig. 3.8. “Wind to Body 3DoF” transformation sub-system

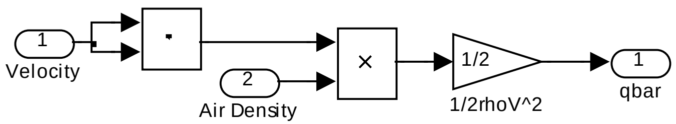

Fig. 3.9. “Dynamic Pressure” implementation

Wind model contains the following three essential components: shear wind, wind gusts, and turbulences models, which are implemented by the standard blocks from the *Environment* sub-library.

The “Dryden Wind Turbulence Model” block requires altitude, airspeed, and direction cosine matrix as inputs. The latter two calculated in the service sub-system shown in Fig. 3.12 below.

Modelling Actuators

Model of aircraft dynamics takes inputs to its controls such as engine throttle and aerodynamic control surfaces. In reality, an introduced input will NOT be instantaneously transferred to the receiving block. Every actuator system has its own dynamics that must be properly modelled.

Certain control surfaces, such as rudder and ailerons, can deflect in any direction with the same maximum angle. In a sense, such actuators

are symmetric with respect to their deflection angles. Sub-system implementing this kind of actuator is shown in Fig. 3.13.

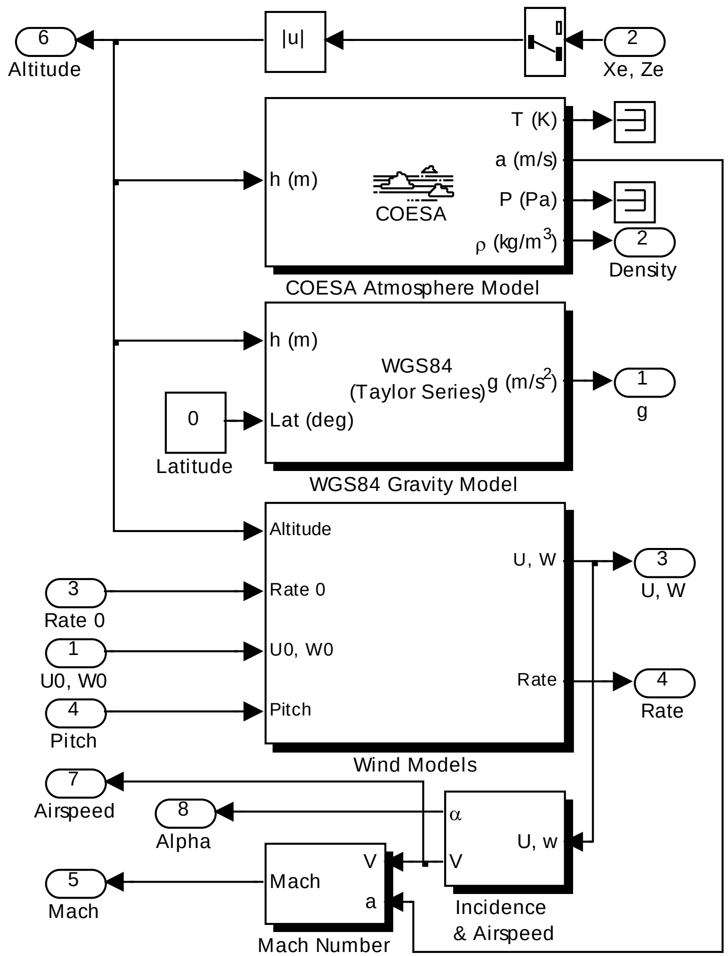

Fig. 3.10. “Environment” sub-system

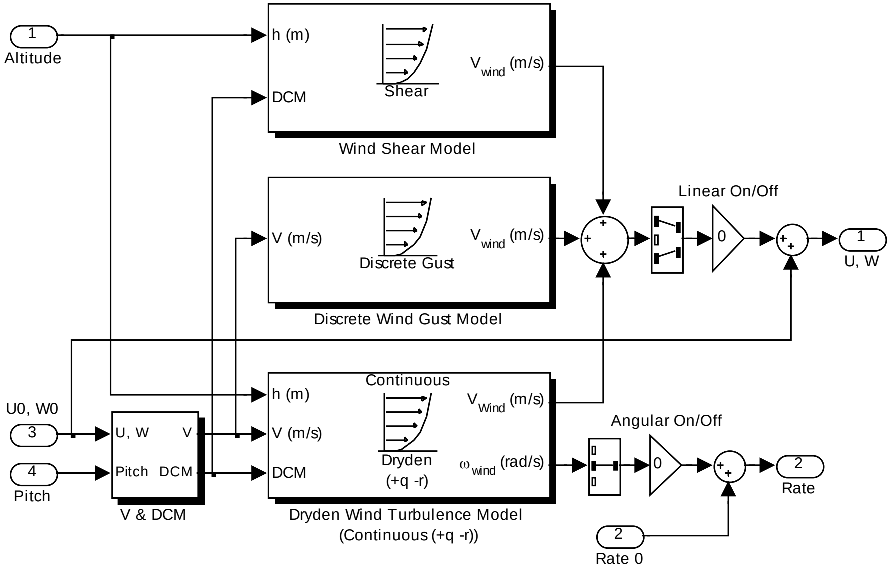

Fig. 3.11. “Wind Models” sub-system

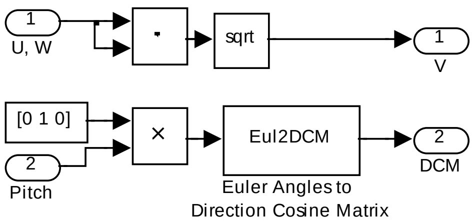

Fig. 3.12. “V & DCM” sub-system implementation

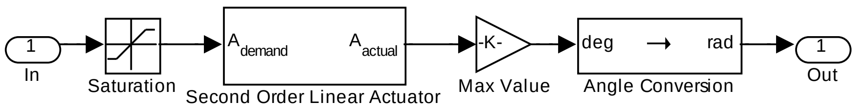

Fig. 3.13. “Symmetric Actuator” sub-system

Here “Upper Limit” and “Lower Limit” parameters of the *Saturation* (from *Simulink / Discontinuities*) block must be set to 1 and -1 correspondingly instead of defaults. This block will cut out any values exceeding the predefined range of [-1,1] for the input signals. “Max Value” gain is set to maximum deflection angle in degrees. Finally block *Angle Conversion* from *Aerospace Blockset / Utilities / Unit Conversions* sub-library transforms the signal from degrees to radians.

Finally, such control surface as elevator has different angles of deflection in different directions. For example, typical values are 35° leading edge down and 15° leading edge up.

Aerospace Blockset also contains Second Order Nonlinear Actuator block. However, this block functionality corresponds to the considered above “Symmetric Actuator” sub-system that should be used instead.

Aircraft Model Parameters

Most of the blocks in the model above are parameterised using external variables (see table 1). Some of these parameters usually are given in the aircraft technical specifications, while the others are obtained using calculations presented above.

Initialization of these variables can be placed in a standard MATLAB m-file, which then should be executed prior to running simulations.

4. GUIDELINES FOR RESULTS VISUALIZATION

During any modelling good and easily perceivable representation of simulation results is a paramount. Typical visualisation sub-system is shown in Fig. 4.1. This sub-system takes coordinates of the aircraft, its airspeed, pitch and incidence angles and displays them using standard visualisation tools from *Simulink / Sinks* sub-library and animation tool provided by the Aerospace Blockset.

Flight Trajectory Plotting

The aircraft flight trajectory can be easily visualised by using standard *XY Graph* block from *Simulink / Sinks* sub-library. Aircraft X coordinate is directed to its first input, and taken by its absolute value Z coordinate (to obtain altitude) is directed to its second input.

Table 1

Aircraft model parameters (F-15 "Eagle")

Variable	Description	Value
a_Mass	Aircraft mass [kg]	20000
a_Iyy	Moment of inertia around Y axis [kg*m^2]	168000
a_Pmax	Maximum total engines thrust at see level [N]	210000
w_S	Main wing area [m^2]	55.7
w_B	Main wing chord [m]	5.2
w_L	Main wing span [m]	13.1
w_CPX	X position of the centre of pressure [m]	0
w_CPZ	Z position of the centre of pressure [m]	0
w_My0	Constant moment coefficient	0
w_MyRate	Rate damping moment coefficient	0.01
w_D0	Main drag harmonic coefficient d0	1.16566
w_D1	Main drag harmonic coefficient d1	-1.00578
w_D2	Main drag harmonic coefficient d2	-0.12529
w_L0	Main lift harmonic coefficient l0	0.18674
w_L1	Main lift harmonic coefficient l1	1.4885
w_L2	Main lift harmonic coefficient l2	0.19916
e_S	Elevator effective area [m^2]	10.5
e_CPX	X position of the elevator centre of pressure [m]	-6
e_Min	Minimal elevator deflection angle [degrees]	35
e_Max	Maximal elevator deflection angle [degrees]	15
e_D0	Elevator drag harmonic coefficient d0	1
e_D1	Elevator drag harmonic coefficient d1	-1
e_D2	Elevator drag harmonic coefficient d2	0
e_L0	Elevator lift harmonic coefficient l0	0
e_L1	Elevator lift harmonic coefficient l1	1.4
e_L2	Elevator lift harmonic coefficient l2	0

Block parameters are as follows: "x-min" (0), which is the initial X position of the aircraft, "x-max" (10000 metres), which is the maximum reachable during simulation position, "y-min" (0), which is the minimal altitude, and "y-max" (2000 metres), which is the maximum flight altitude.

Needless to say, that in every specific case of simulation these parameters can be set accordingly. Extraction of the X and Z components of the aircraft position is done by using *Mux* block from *Simulink / Signal Routing* sub-library. *Abs* (absolute value) block can be found in *Simulink / Math Operations*.

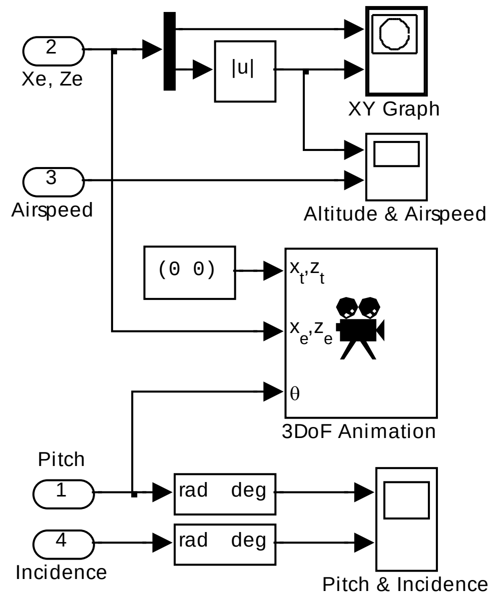

Fig. 4.1. Visualisation sub-system

Visualising flight parameters

All other flight parameters can be visualised by using standard *Scope* block from *Simulink / Sinks* sub-library. However, certain modifications to the Scope parameters are still required. In order to use the same scope for several quantities, one must set appropriate "number of axes" (2). Finally, it is highly recommended to remove limit for the plotting points at the second tab of the Scope parameters (see Fig. 4.2).

For that purpose, just uncheck the "Limit data points to last" checkbox. One should also note, that by checking the second checkbox "Save data to workspace", simulation results could be saved to the Matlab workspace and later plotted by using standard *plot* function.

This may be required to have the capability to export the plotted graphs in an appropriate format to other software or to MS Word documents.

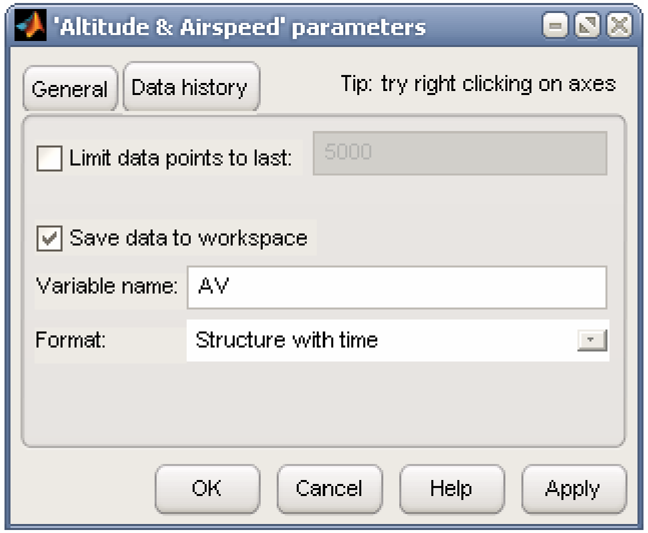

Fig. 4.2. Removing limit for the plotting points

In order to plot the exported to workspace data, the following commands based on different kinds of data available from the Matlab command line:

```
>> plot( AV.time, AV.signals(1,1).values )
>> plot( AV.time, AV.signals(1,2).values )
```

The first command plots altitude (first input) and the second – for the airspeed.

After the graph is plotted, X and Y axes labels must be inserted, describing the quantity and its dimension.

While copying the plots to the MS Word document it is important to make sure that the background colour is set either to white (“force white background” option) or transparent. Exporting format must be metafile.

5. GUIDELINES FOR CONTROL SYSTEMS CALCULATION

Benefits of the presented above approach to non-linear flight simulation become apparent when newly developed control systems must be tested in numerical experiment, which certainly should be as close to reality as possible.

“Uncontrolled” Flight Simulation

First of all let us study results of flight simulation without any service control systems or autopilots. Assuming initial velocity of 120 m/s, initial altitude 1000 m, no wind, throttle input 0.5, elevator input -0.5, simulated incidence angle, pitch, flight altitude, and airspeed as functions of time are shown in figures 5.1-5.4 respectively.

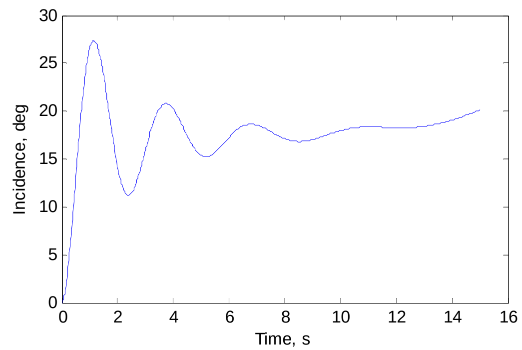

Fig. 5.1. Incidence angle as a function of time

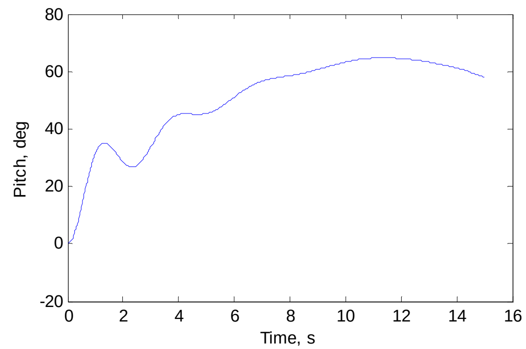

Fig. 5.2. Pitch angle as a function of time

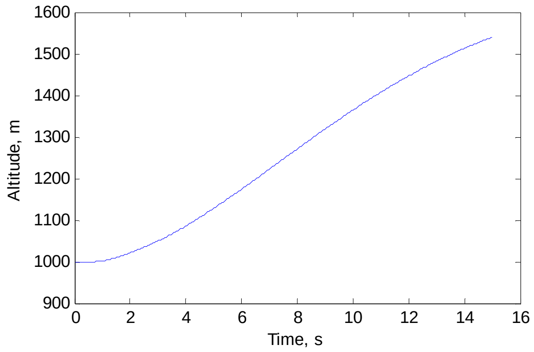

Fig. 5.3. Flight altitude as a function of time

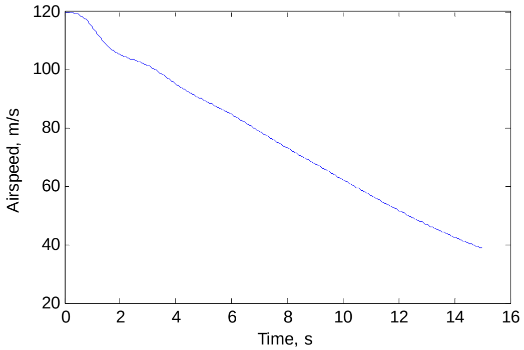

Fig. 5.4. Flight airspeed as a function of time

Short periodic oscillations of an un-damped aircraft are easily observed at the above shown graph. Long periodic (phugoid) oscillations are also could be seen at the pitch angle plot when the short-periodic oscillations are settled (time > 10 s).

Damping Control System

In order to eliminate short periodic oscillations, damping control system is used. It takes pitch angular rate as an input and provides elevator control according to the following control law:

$$\delta_e^{\omega} = K(H,V)\frac{T \cdot s}{T \cdot s + 1}\omega_y . \tag{5.1}$$

Here $K(H,V)$ is the general damping gain that must be adjusted at different altitudes and velocities, T is the time constant. Implementation of such a system is shown below in Fig. 5.5.

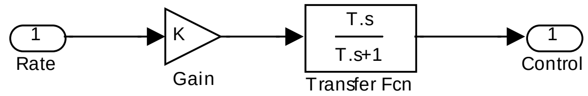

Fig. 5.5. “Damping Control” sub-system (K=1, T=1)

This system is then added to sub-system shown in Fig. 3.2 (see Fig. 5.6). Resulting transient process for the incidence angle is shown in Fig. 5.7. Now transient process settles after half-period of oscillations. This is achieved by means of proper choice of the gain coefficient and time constant.

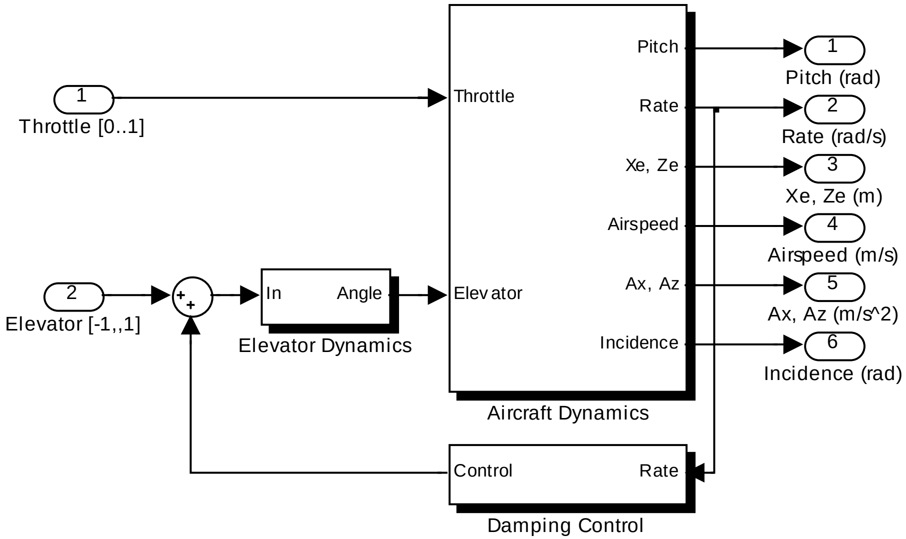

Fig. 5.6. Damping control added to the aircraft model

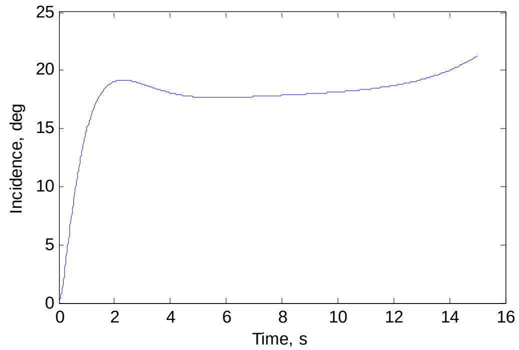

Fig. 5.7. Damped incidence angle

Stability Control System

Modern military aircrafts are designed with its centre of pressure positioned in front of the centre of gravity that makes them statically unstable and highly agile at the same time.

In order to provide aircraft controllability during the flight, stability control system must be used. Such system takes either incidence angle or vertical overload measurements to provide corresponding elevator control. This system could be also considered as a system that limits overload experiences by the aircraft. The simples control law is given by the following expression:

$$\delta_e^n = -K_n \frac{1}{g} A_z \,. \tag{5.2}$$

Here A_z is the vertical acceleration, g is the acceleration due to the gravity, K_n is the gain factor that being properly chosen will provide necessary stability. Vertical overload obtained during flight simulations without stability control system is shown in Fig. 5.8 (initial velocity 200 m/s, throttle 0.5, elevator -1). Peak overload is about 9g, which is dangerous and within the breaking limit even for the military jet fighters. Stability control system could be added to the existing damping control system (see figures 5.9 and 5.10).

After introducing the stability control system vertical overload for the same flight parameters is then limited to less than 4g (see Fig. 5.11), which is totally acceptable for most of the aircrafts and pilots.

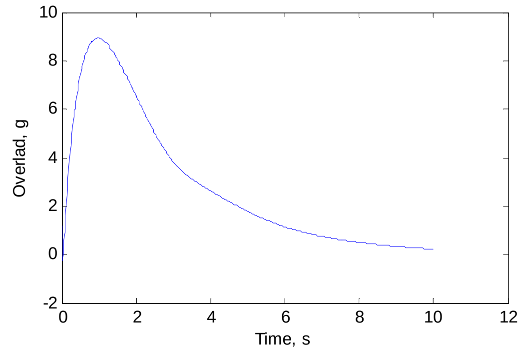

Fig. 5.8. Vertical overload without stability control

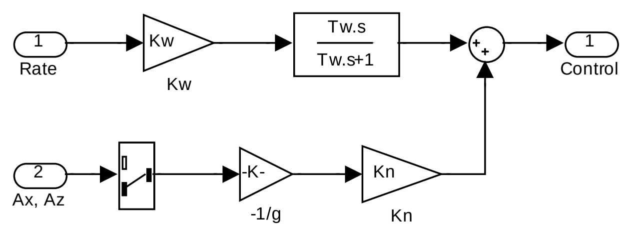

Fig. 5.9. Stability control loop is added to the damping control (Kn=0.2)

Airspeed Control System

Aerodynamic forces and therefore all of the aircraft flight characteristics depend on its airspeed. From this point of view, efficient airspeed stabilisation is of utmost importance. Airspeed can be controlled either by controlling *pitch* angle or by engine *thrust*. The simplest airspeed control law for the latter case is given by the following expression

$$\delta_t^v = (V_0 - V)(K_{v0} + \frac{K_v}{s}) \,. \tag{5.3}$$

Here V_0 and V are the target and current airspeeds respectively, K_{v0} and K_v are the gains that must be appropriately chosen to provide airspeed stabilisation at V_0. Implementation of this law is shown in Fig.

5.12 (target airspeed is 150 m/s). This sub-system is added to the aircraft model as shown in Fig. 5.13.

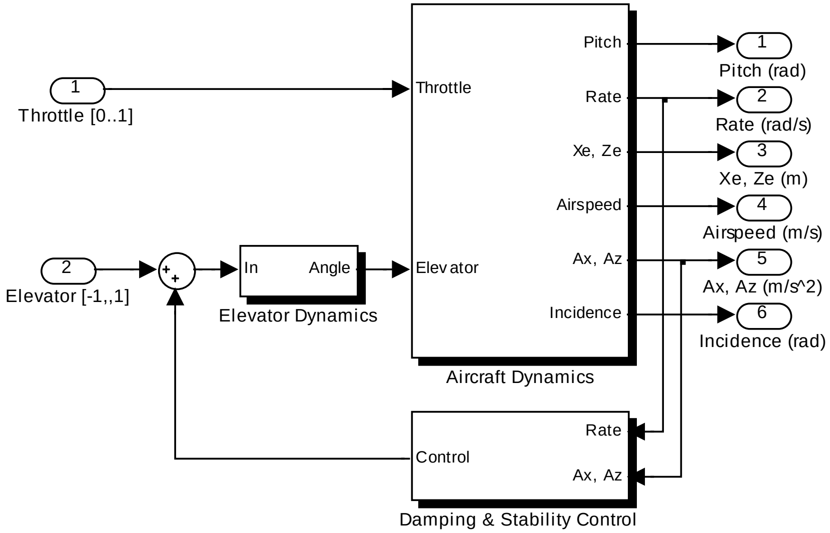

Fig. 5.10. Resulting model with added stability control

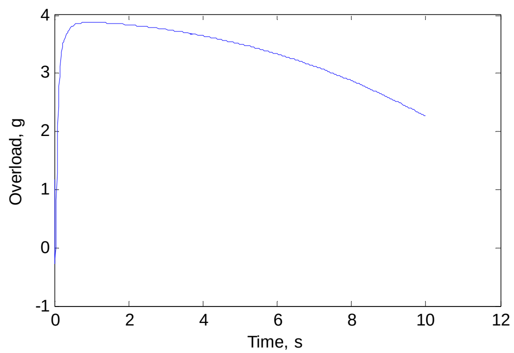

Fig. 5.11. Overload limited by the stability control system

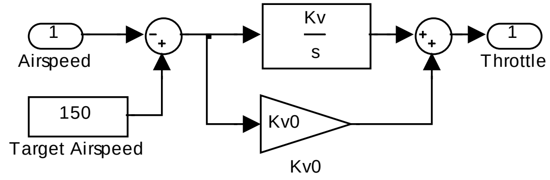

Fig. 5.12. "Airspeed Control" sub-system (K_v=0.1, K_{v0}=0.001)

At the same time, no throttle is provided from outside. Note, that saturation element limiting throttle has limits from 0 to 1.

Results of simulation after airspeed control system is added to the mode are shown in the figures 5.14 (without winds) and 5.15 (with winds). Initial airspeed is 120 m/s.

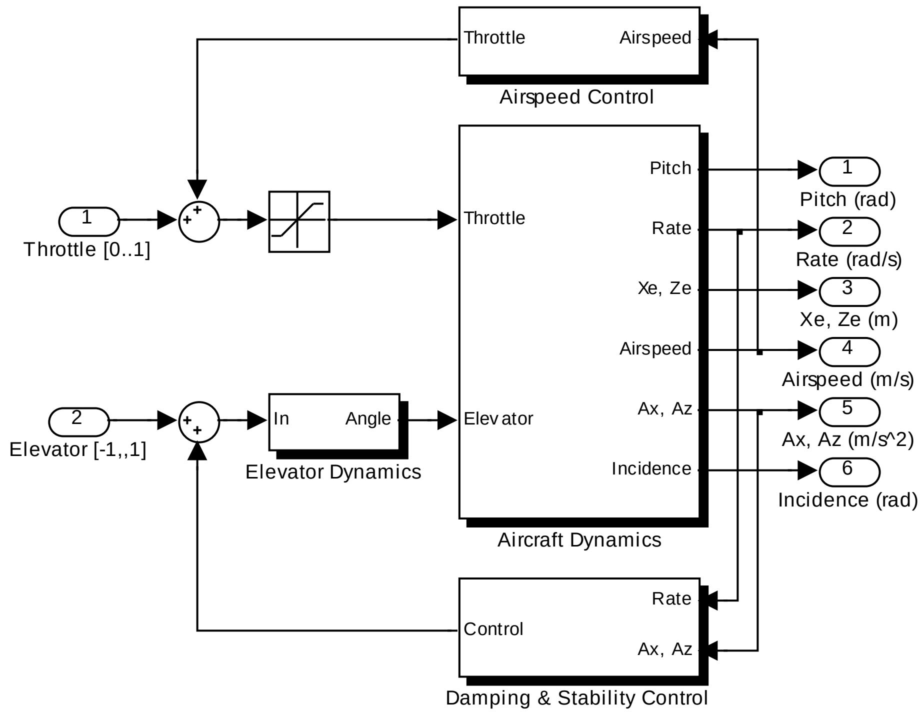

Fig. 5.13. "Airspeed Control" added to the model

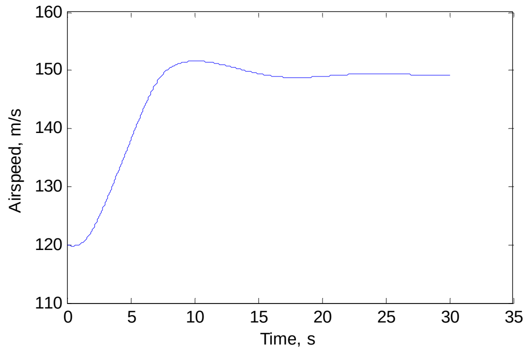

Fig. 5.14. Aircraft airspeed stabilised at 150 m/s (without wind)

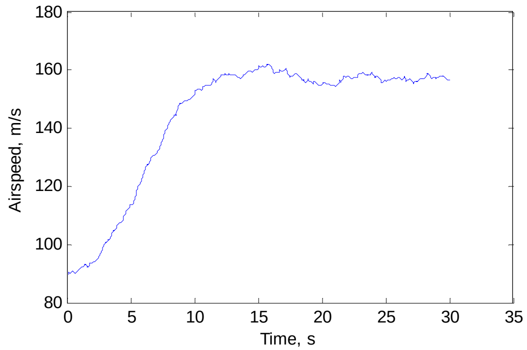

Fig. 5.15. Airspeed stabilisation with wind

From the shown above results one can see that settling time for the implemented airspeed stabilisation is about 15 s. After that time aircraft flight becomes very close to the ideal steady flight mode.

One should also note, that presence of disturbances, such as winds and turbulences, noticeably degrades accuracy of the system. And since in reality this is always the case, some more advanced control laws, which may be derived using statistical dynamics, should be applied instead.

Altitude Control System

Finally, altitude control system can be now added to the aircraft model. Altitude control law in its simplest form is

$$\delta_e^h = (H - H_0)K_h + (\theta - \theta_h)K_p . \qquad (5.4)$$

Here H_0 is the target altitude, H is the current altitude, θ is the current pitch angle, θ_h is the pitch angle of a horizontal flight at the target altitude (determined experimentally), K_h and K_p are the gain factors that are used to implement desired qualities of the system. When altitude control system is added, external input to the elevator control is zero, and the elevator is affected by altitude, damping, and stability control systems only. At the same time, airspeed control system completely controls throttle (no external input is provided). Altitude control sub-system implementation is shown in Fig. 5.16, and Fig. 5.17 shows the aircraft model with this system added. Initial altitude is 1000 m. After 20 seconds of flight, when airspeed is stabilised, target altitude is set to 1300 m. Simulation results with and without winds are shown in the graphs (see figures 5.18 – 5.19).

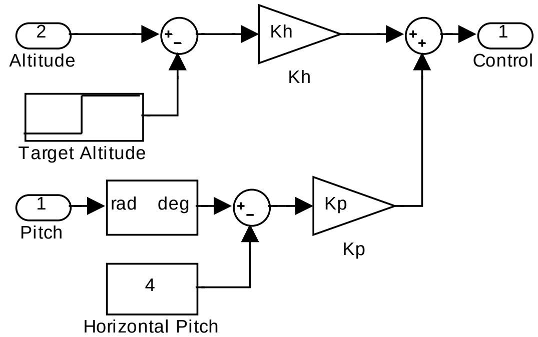

Fig. 5.16. “Altitude Control” sub-system implementation
(K_p=0.1, K_h=0.01, θ_h=4)

One should note that changing altitude causes airspeed to drop. However, this variation has been compensated by the airspeed control system. Adding wind although degenerates quality of the altitude stabilisation, accuracy is still within the acceptable tolerance.

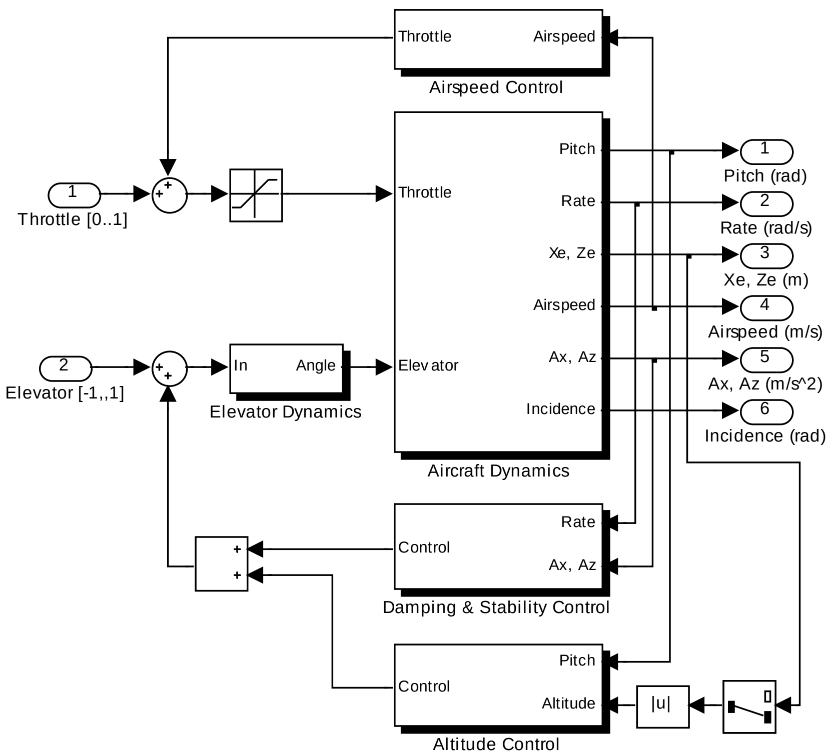

Fig. 5.17. Aircraft model complete with the altitude control

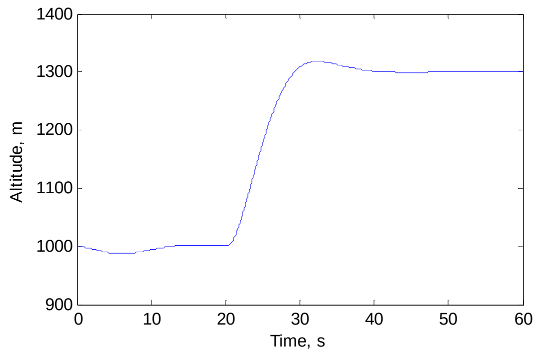

Fig. 5.18. Altitude transient process (no wind)

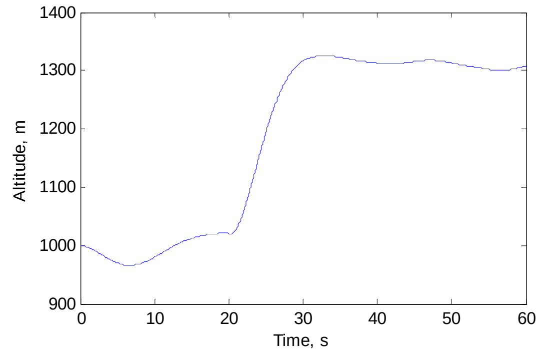

Fig. 5.19. Flight altitude with presence of wind

Presented above flight simulation with Simulink technique allows to model flight dynamics of different types of aircrafts. Although jet aircraft model is presented here, minor modifications will allow implementing models of other kinds of aircrafts as well.

RECOMMENDED LITERATURE

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Навчальне видання

ОСНОВИ ТЕОРІЇ УПРАВЛІННЯ ПОЛЬОТОМ

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Методичні рекомендації до виконання курсової роботи
для студентів напрямку 0914 «Комп'ютеризовані системи, автоматика і управління»
(англійською мовою)

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