**Aircraft Simulation Programming**

Calculating Aerodynamic Stability Coefficients

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# Abstract

Write the abstract

# Introduction

# Related Work

# System Model

# Problem Statement

**Rewrite but look at equations first**

When deriving the equation of motion for an aircraft, changes in aerodynamic forces and moments are expressed in terms of stability derivatives e.g. . For example, the aerodynamic forces in the X-direction can be expresses in the follow manner as a Taylor series expression:

The problem in determining the stability derivatives or their associate coefficients is that it can be very costly in terms of money and time using methods such as flight tests, wind tunnel testing or Computation fluid Dynamics (CFD). In some cases the derivative or coefficient may not directly be able to measure from these methods. Instead a quick method is to use analytical expressions which are less accurate.

# Solution

# Analysis

## 

Figure - Definition of forces, moments and velocity components

## If Mach number is less than 0.3 in a flow, the density changes in the flow will usually be negligible

Subscript zero indicates the reference flight condition – explain what tis means

Calculation given for aft tail configuration

## Estimate Longitudinal Stability Coefficients

|  |  |  |  |
| --- | --- | --- | --- |
|  |  |  |  |
| ***u*** |  |  |  |
|  |  |  |  |
| ***q*** | *0* |  |  |
|  | *0* |  |  |
|  | *0* |  |  |

**Need to put and are neglagable and why**

## Derivatives for change in forward speed (u)

Expressing in the coefficient form:

(7‑1)

Where &

*Subscript 0 represents the reference condition.*

The coefficient can be estimated from the drag coefficient verses Mach number for the Mach number of interest.

If low speed flight conditions are being considered then terms related to Mach number can be ignored i.e. . for gliding flight and can also be a good approximate for jet powered aircraft. For variable pitch propeller and piston engine power plant can be approximated by assuming it being equal to the negative reference drag coefficient i.e.

Expressed in coefficient form:

(7‑2)

Where

can be neglected at low flight speeds however can become come very significant near the critical march number for the aircraft.

Expressed in coefficient form:

(7‑3)

At low speed the coefficient is 0 and at high speeds aero elastics bending of the aircraft can cause large changes in the magnitude of the coefficient.

## Derivatives for change in pitch velocity (q)

Both the wing and horizontal tail have an effect on the pitching motion of the aircraft, in which the wing contribution is very small compared to the tail.

When calculating the q derivatives for pitch it is common practice to increase the tail effect by 10% to allow for the wing and body. However, if the wing is highly swept, of low aspect ratio or may not have a tail i.e. flying wing then the wing may have a significant effect on the q derivatives.

A change in tail angle of attach will create a pitching motion q and thus a change in lift on the tail:

qltail

ltail

ltail

α

u0

CG

q

Horizontal Tail

Ltail

ltail

Figure ‑ Aerodynamic force due to pitch rate

ltail is the horizontal length from the aircraft cg to horizontal tail centre of pressure.

From Figure 7‑1,

The tail efficiency can have values in the range of 0.8 -1.2 and depends on the location of the tail surface.

\* (7‑4)

The pitching moment due to change in lift can be calculated using the following:

If , then,

\* (7‑5)

***\*Equations represent tail contribution only. The coefficient of the complete aircraft, as a general rule is to increase and by 10%.***

Also note the and stability coefficients are related in the following way:

(7‑6)

## Derivatives for change in Angle of Attack ()

The derivatives describe the changes in forces and moments to the aircraft when the angle of attack is changed. Normally an increase in angle of attaches results in increased lift and drag couple together with a negative pitching moment.

Assuming that the thrust coefficient is independent of

When the drag is given by a parabolic polar in the form:

**Where e is span efficiency factor**

Then,

(7‑7)

The derivative can be expressed in term of lift and drag, again assuming the angle between the thrust line and X-axis is very small.

Therefore,

(7‑8)

The total pitching moment of the aircraft can be obtained by the summation of the individual contributions i.e. fuselage, wing and tail:

The fuselage contribution assumes that the power plant effects are included.

For the wing contribution:

Given the form

For the tail contribution:

Given the form

Therefore the total pitch coefficient with respect to change in angle of attack can be given as:

(7‑9)

Define & – distance fromwing leading edg

## Derivatives for the rate in time change of Angle of Attack ()

The stability coefficients and are due to the lag in down wash at the tail. As the wing angle of attack changes so does the downwash from the wing. Thus as a result the downwash will change over time till it reaches the tail.

If an aircraft has a forward speed of and the tail is a distance of from the wing then, the incremental time for change in the trailing vortex at the tail would be:

The angle of attack at the tail with respect to the changing down wash can be calculated as follows:

The change in the lift force of the tail can be calculated as follows:

If then,

(7‑10)

Where the rate of change of the downwash angle with respect to the angle of attack can be estimated from the wing characteristics as follows:

The pitching moment due to the changing downwash on the tail can be calculated by the following:

If then,

(7‑11)

**Equations represent tail contribution only. The coefficient of the complete aircraft, as a general rule is to increase and by 10%.**

Also note the and stability coefficients are related in the following way:

(7‑12)

## Derivatives for change in Elevator angle ()

Change in lift due to the elevator deflection can calculated as follows:

The moment due to the elevator defection can be expressed as:

If then,

If

Where  **flap effectiveness parameter and can be determined from the following chart:**

**diagram**

Therefore,

(7‑13)

(7‑14)

When comparing equation ??? and ??? the following relationship can be obtained:

(7‑15)

## Estimate Lateral Stability Coefficients

|  |  |  |  |
| --- | --- | --- | --- |
|  |  |  |  |
| ***β*** |  |  |  |
| ***p*** |  |  |  |
| ***r*** |  |  |  |
|  |  |  |  |
|  |  |  |  |

### Derivative due to rolling rate (p)

The coefficients , and are the result of the rolling angular velocity p. As the aircraft rolls there is a linear change in the velocity distribution over the wing, vertical and horizontal tail section causing a change in the local angle of attack.

The change in lift distribution across the wing results in a rolling moment that opposes the motion and is proportional to the roll rate.

Figure - Velocity distribution due to rolling – need to fix

**p**

***y is a position along the span of the wing***

From if the forward velocity is , then the local change in angle of attack can be assumed as:

The incremental lift force created by the rolling motion can be expressed as

The incremental roll moment can be calculated by multiplying the above equation by the incremental moment arm y.

The rolling moment can be determined by integration the total moment contribution across the wing.

Or as a coefficient,

If the rolling damping coefficient is,

From the above equation it is evident that as the wing span increases so does the roll damping. Generally for aircraft the wing is assume to contribute mostly to the roll dampening, however will some low aspect ratio wings other components may start to significantly add to the roll dampening for example, vertical and horizontal tail.

**py**

### Derivatives due to the yawing rate (r)

The coefficients , and are the result of the yawing angular velocity r.

# Simulation & Experiment

Conclusions

References

Appendix

Formulas

Dimensional to non-dimensional

|  |  |  |
| --- | --- | --- |
| Dimensional | non-dimensional | Divisor |
| X,Y,Z |  | QS |
| M |  | QS |
| L,N |  | QSb |
|  |  |  |