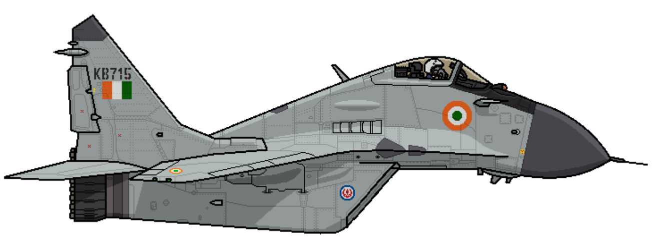
ACCELERATION CONTROL SYSTEM

Prepared by



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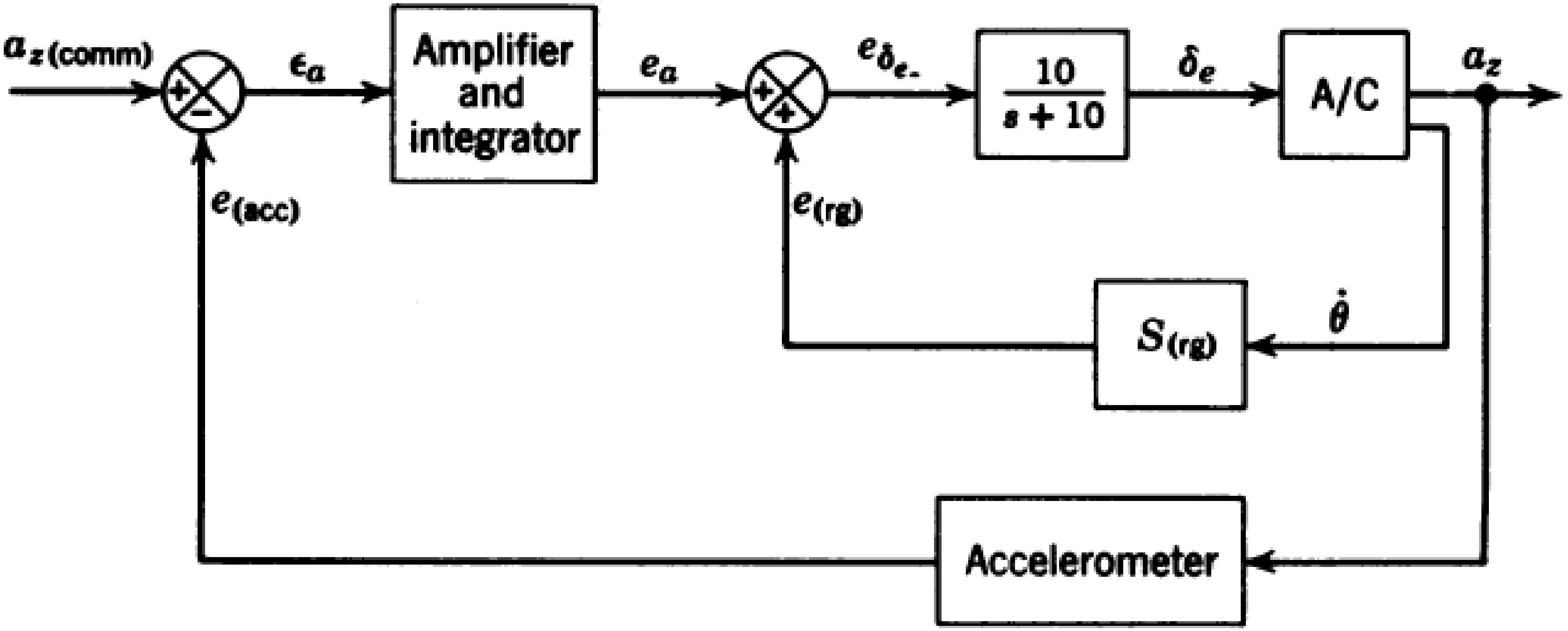
- Acceleration control is one of the most important autopilot in fighter aircraft.
- It helps to limit the acceleration of the aircraft during maneuvering.
- Maximum acceleration during the maneuver may damage the structure of the aircraft as well as damage pilot.
- A pilot may handle upto 4 g of acceleration.
- "pulling 3 G's" is equivalent to experiencing 3 times the normal gravitational force. A person who weighs 150 lbs at 1 G will actually weigh 450 lbs at 3 G's – no kidding!

- The circulatory system is most significantly affected by increased G-forces during flight. Even at 1G, blood pressure in an upright person is highest in the lower extremities (the legs) and lowest intra cerebrally (in the cranium) due to gravity.

- At larger +G forces, this physiologic phenomenon is magnified and a larger discrepancy of blood pressures between cranium and the lower body occurs. At some point, intracranial perfusion cannot be maintained and significant cerebral hypoxia (no blood = no oxygen) follows. The end result is unconsciousness. This is said to be "G lock".

- Other less serious effects of large G forces are musculoskeletal pain (usually confined to the back and neck).

- To avoid the above mentioned problems the acceleration control systems are adopted in most fighter aircrafts.



Block diagram of an acceleration control system

$$m\left(\dot{W}+PV-QU\right)=mg\cos\phi\cos\Theta+\left(-D\sin A-L\cos A\right)-T\sin\phi_T$$

$$For\ longitudinal\ equation\ P=R=V=0$$

$$\sum\Delta F_z=m(\dot{W}+QU)$$

$$ma_z=m(\dot{W}-QU)$$

$$a_z=\dot{w}-U\dot{\theta}$$

$$a_z=U\left(\frac{\dot{w}}{U}-\dot{\theta}\right)=U\left(\dot{\alpha}-\dot{\theta}\right)$$

$$\frac{\dot{w}}{U}=\dot{\alpha}$$

$$\frac{a_z}{\delta_e}=U\left(\frac{\dot{\alpha}}{\delta_e}-\frac{\dot{\theta}}{\delta_e}\right)$$

$$\frac{\dot{\alpha}}{\delta_e}=\frac{-\left(0.1s^2+15s\right)}{s^2+0.9s+8}$$

$$\frac{\dot{\theta}}{\delta_e}=\frac{-\left(15s+6\right)}{s^2+0.9s+8}$$

$$\frac{a_z}{\delta_e}=\frac{-77.7\left(s^2-60\right)}{s^2+0.9s+8}$$

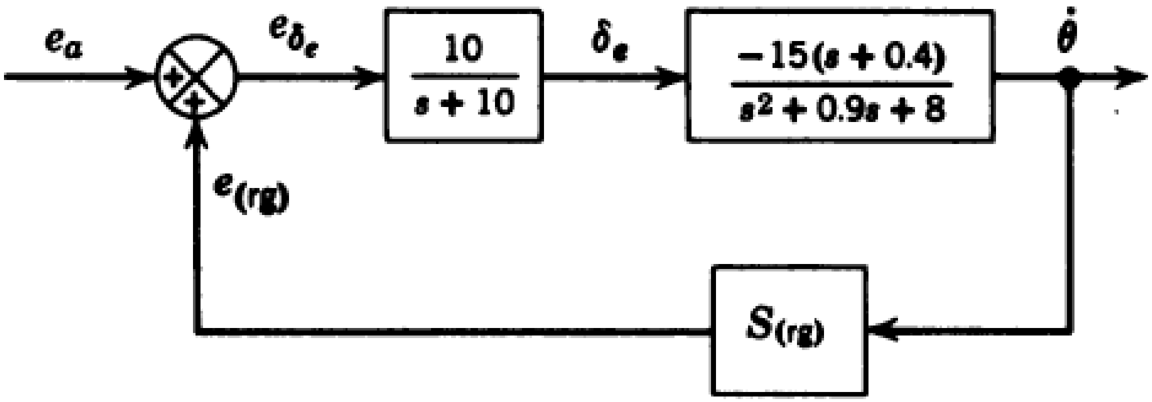
Factoring

$$\frac{a_z}{\delta_e} = \frac{-77.7\left(s+7.75\right)\left(s-7.75\right)}{s^2+0.9s+8}\,\frac{ft/\sec^2}{rad}$$

But the units for the transfer function of the elevator servo are deg/volt, and for the accelerometer they are volt/g. The units of the above equations can be changed to g/deg by dividing by 32.2 (ft/sec2)/g and 57.3 deg/ra

$$\frac{a_z}{\delta_e} = \frac{-77.7\left(s+7.75\right)\left(s-7.75\right)}{s^2+0.9s+8} \times \frac{ft/\sec^2}{rad} \times \frac{1}{\left(32.2\right)} \times \frac{g}{ft/\sec^2} \times \frac{1}{\left(57.3\right)} \times \frac{rad}{\deg}$$

$$\frac{a_z}{\delta_e} = \frac{-0.042\left(s+7.75\right)\left(s-7.75\right)}{s^2+0.9s+8}\,\frac{g}{\deg}$$



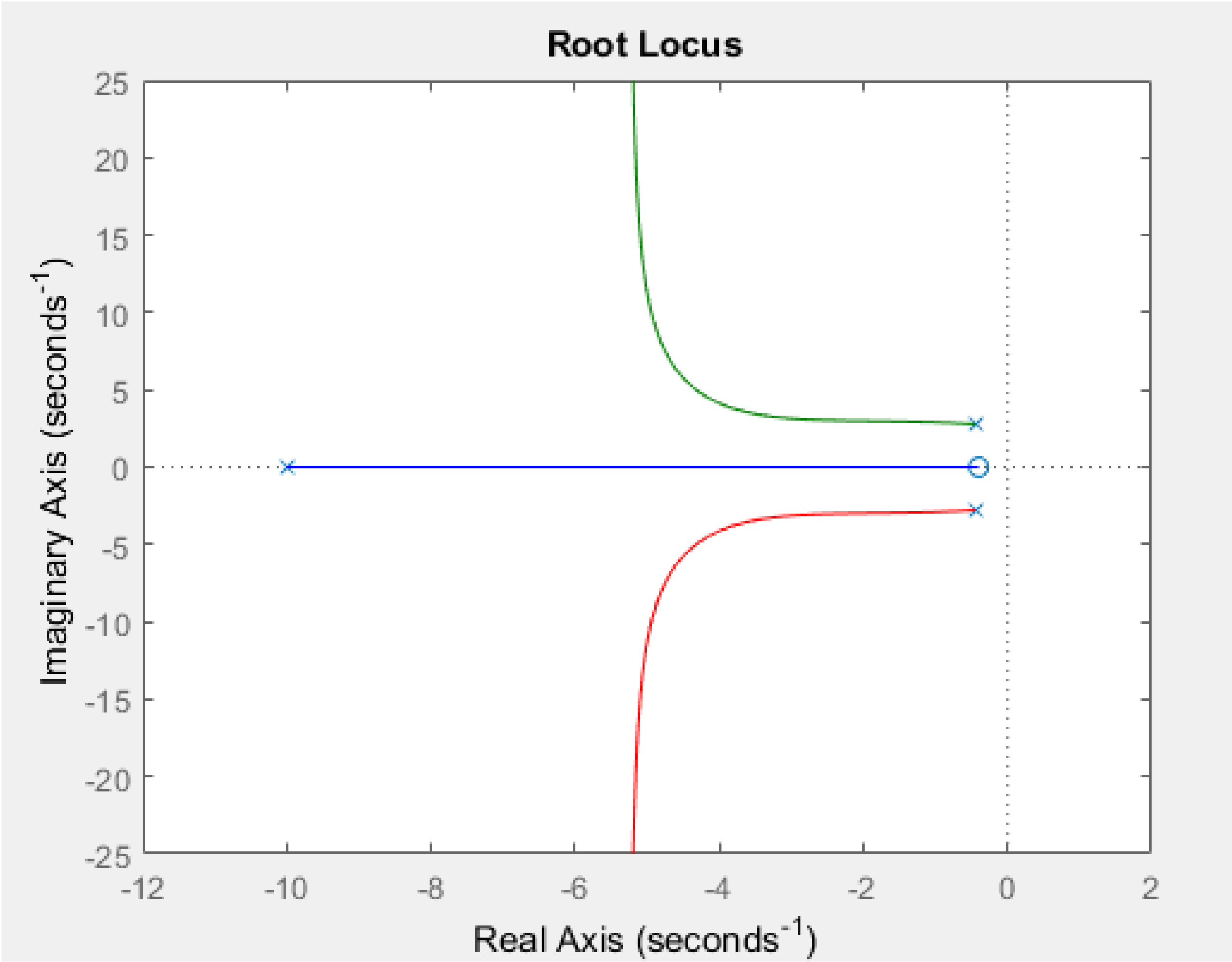
Block diagram for the Inner loop of the acceleration control system

$$\frac{\dot{\theta}(s)}{\delta_e} = \frac{-15(s+0.4)}{(s^2+0.9s+8)}$$

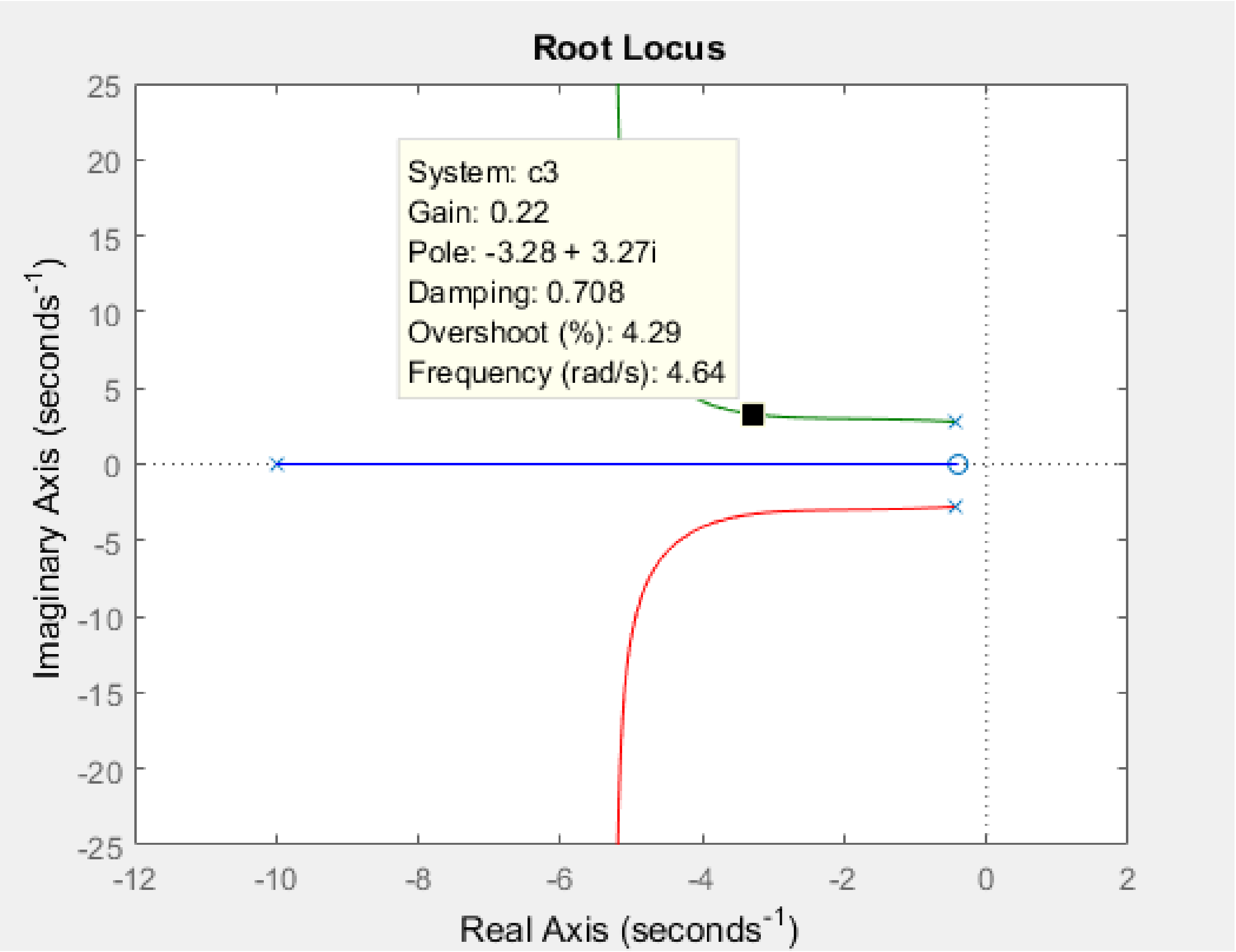
$$Servo\,TF: \quad \frac{\delta_e}{e\delta_e} = \frac{10}{s+10}$$

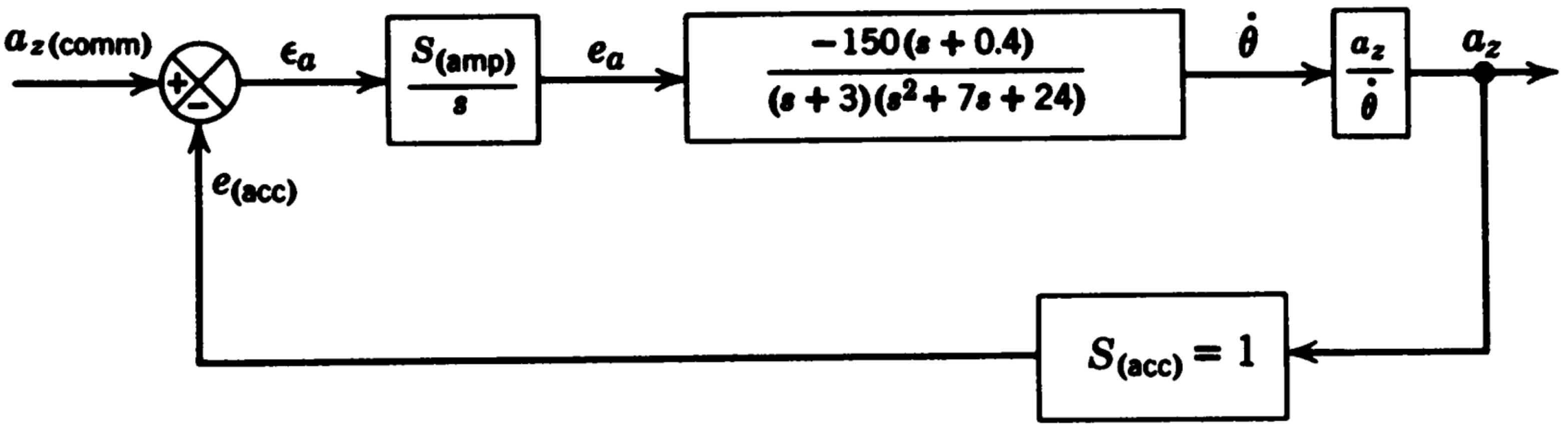
$$\frac{\dot{\theta}(s)}{e\delta_e} = \frac{-150(s+0.4)}{(s+3)(s^2+7s+24)} \frac{\text{deg/sec}}{volt}$$

$$\frac{\dot{\theta}(s)}{e\delta_e} = \frac{-150(s+0.4)}{(s+3)(s^2+7s+24)} \frac{\text{deg/sec}}{volt}$$



$$\frac{\dot{\theta}(s)}{e\delta_e} = \frac{-150(s+0.4)}{(s+3)(s^2+7s+24)} \frac{\text{deg/sec}}{volt}$$





Block diagram for the outer loop for the acceleration control system *we already know that*

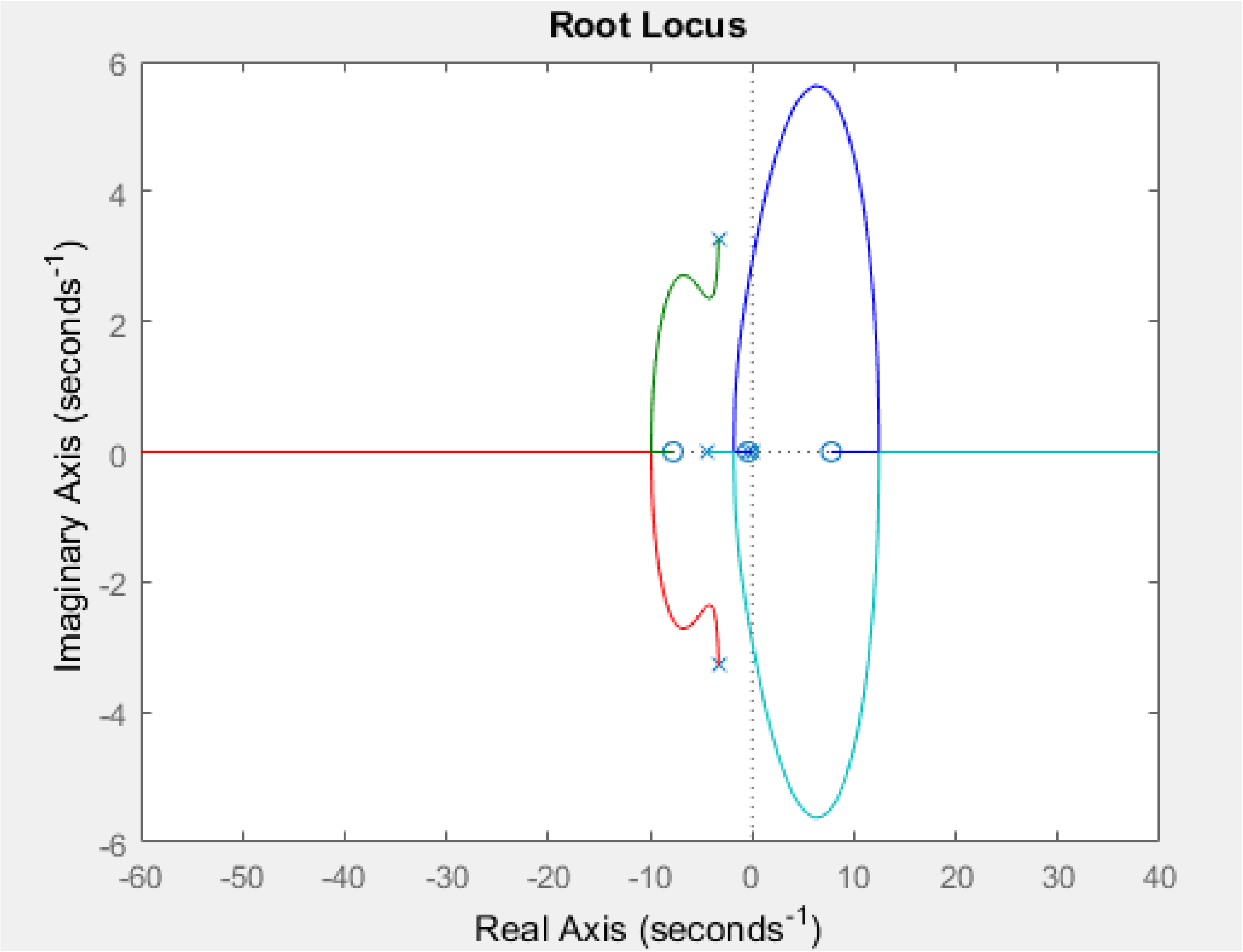
$$\frac{a_z(s)}{\delta_e(s)} = \frac{-0.042(s+7.75)(s-7.75)}{s^2+0.9s+8}\frac{g}{\deg}$$

$$\frac{\dot{\theta}(s)}{\delta_e} = \frac{-15(s+0.4)}{(s^2+0.9s+8)}$$

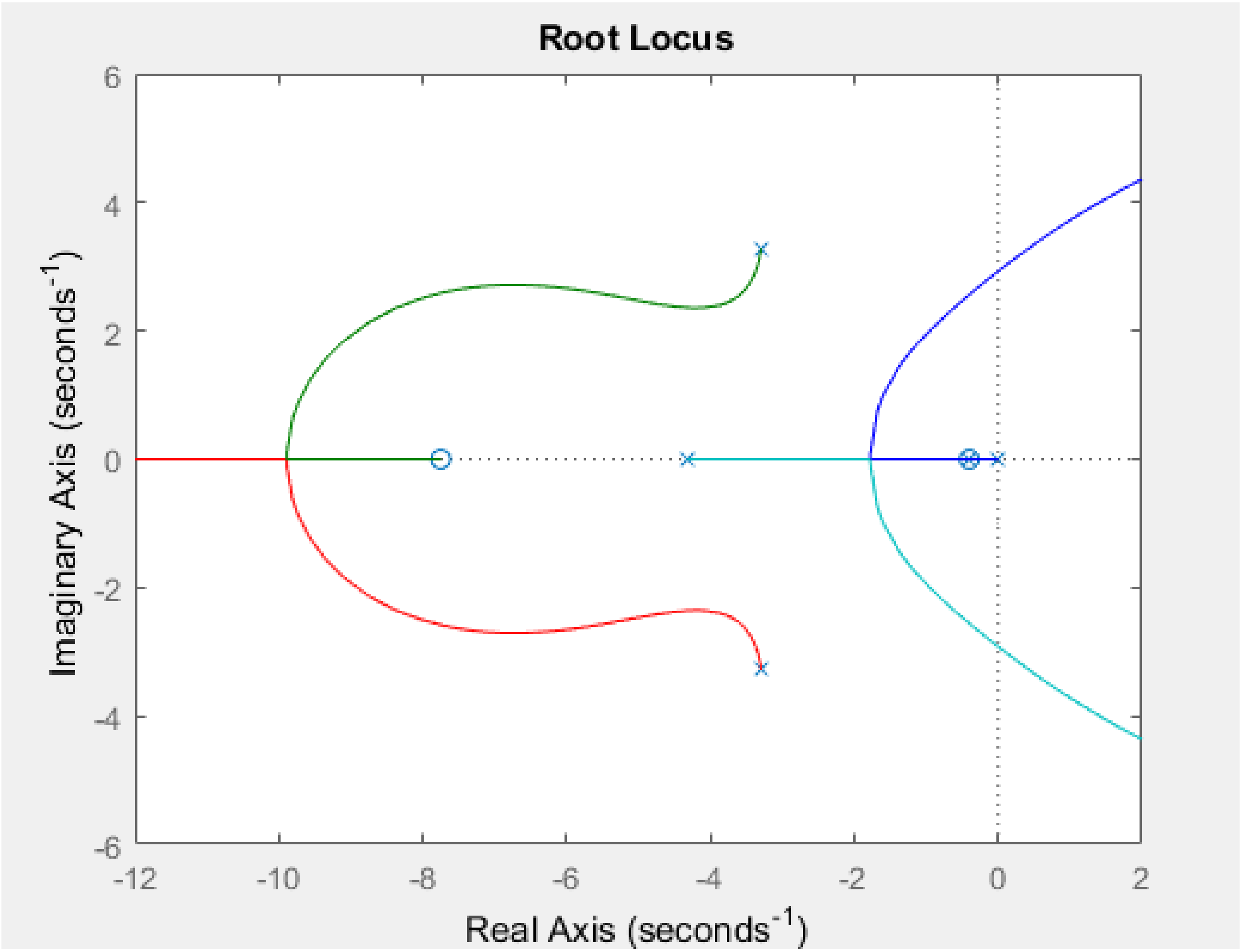
$$\frac{a_z}{\delta_e} \Big/ \frac{\dot{\theta}(s)}{\delta_e}$$

$$\frac{a_z(s)}{\dot{\theta}(s)} = \frac{-0.042(s+7.75)(s-7.75)}{-15(s+0.4)}\frac{g}{\deg/\sec}$$

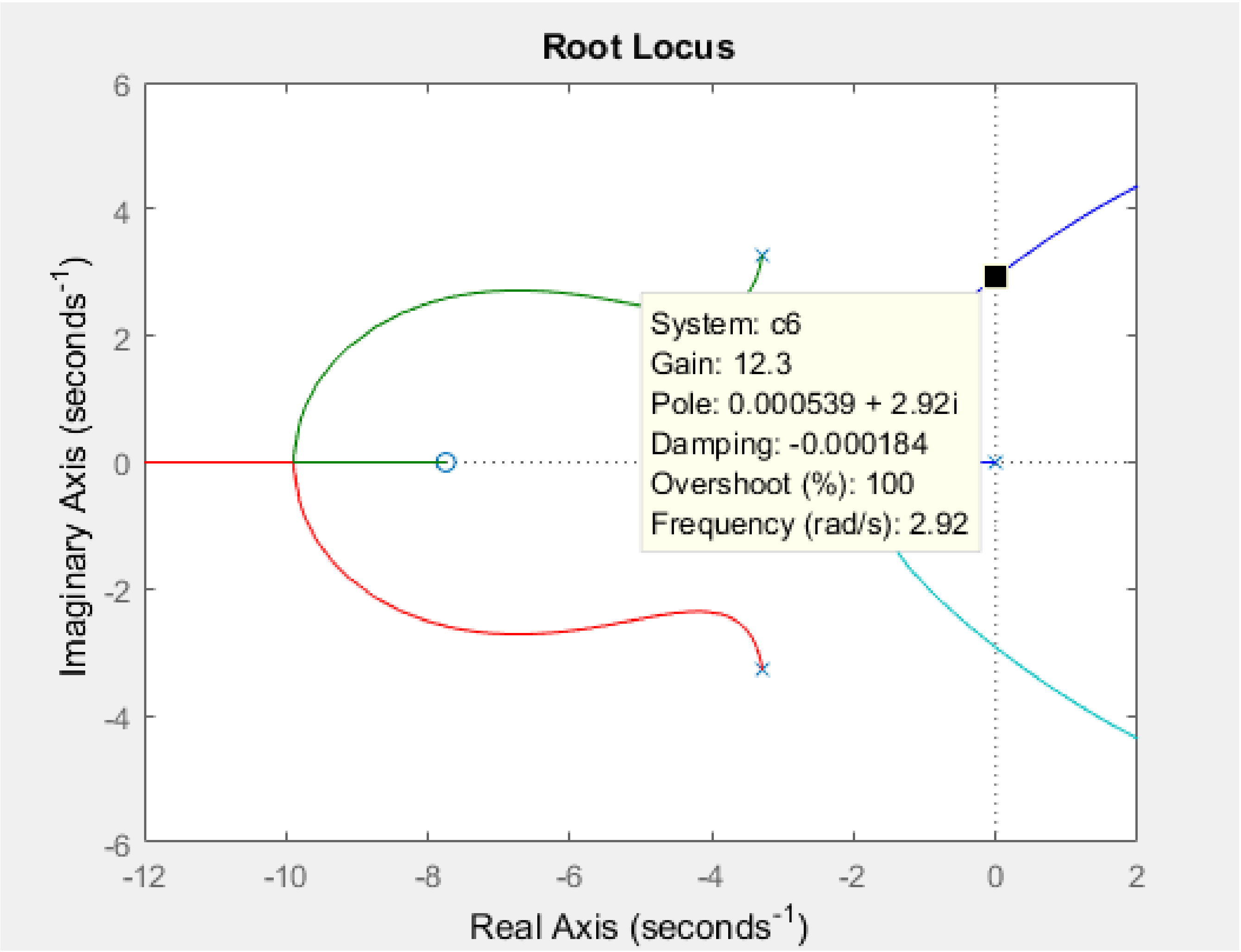
$$\frac{a_z(s)}{a_{z(command)}(s)} = \frac{-0.93(s+7.75)(s-7.75)}{(s^2+2.2s+2.4)(s^2+7.8s+24)}$$



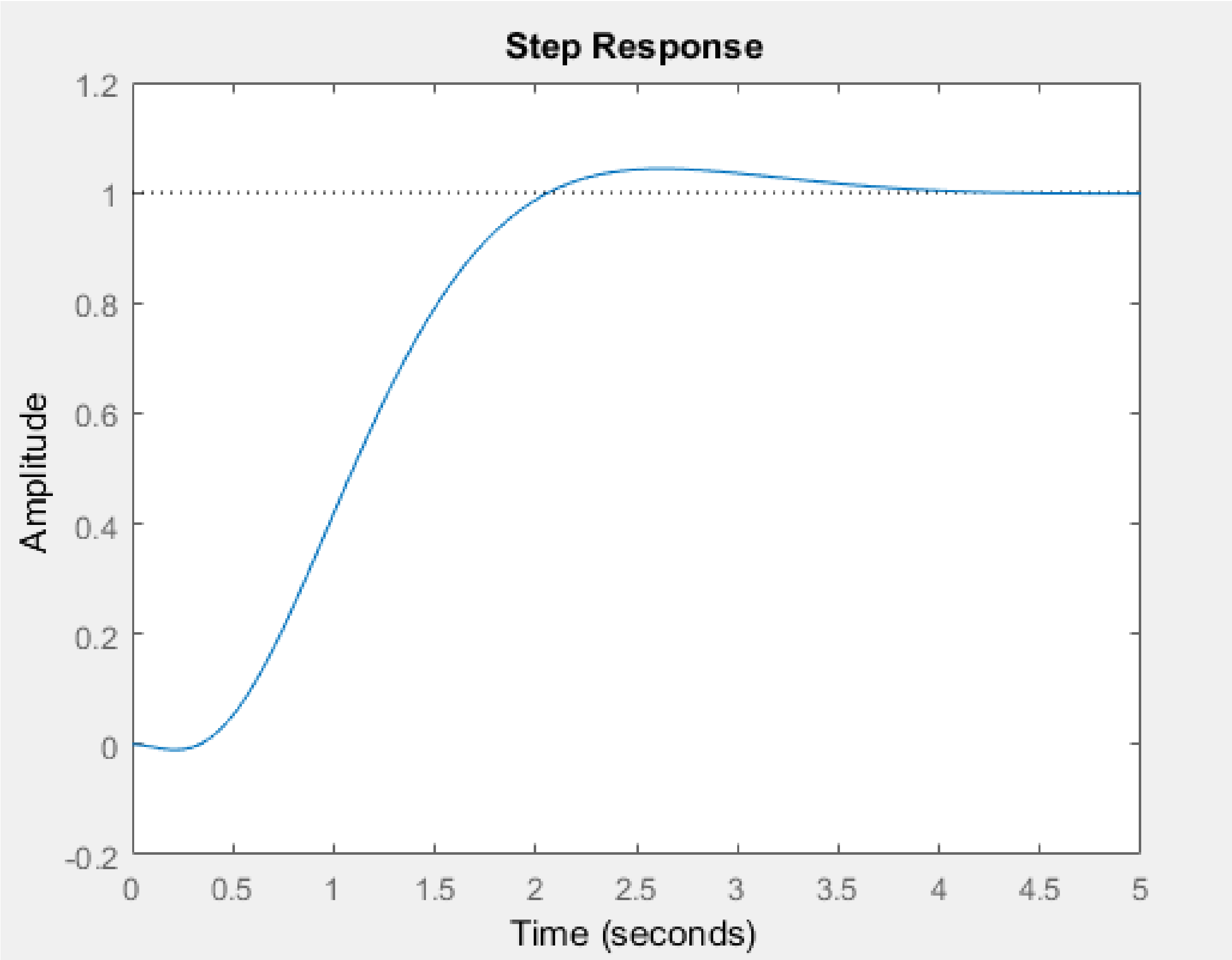
Root Locus
Imaginary Axis (seconds^-1)
Real Axis (seconds^-1)
6
4
2
0
-2
-4
-6
-60
-50
-40
-30
-20
-10
0
10
20
30
40

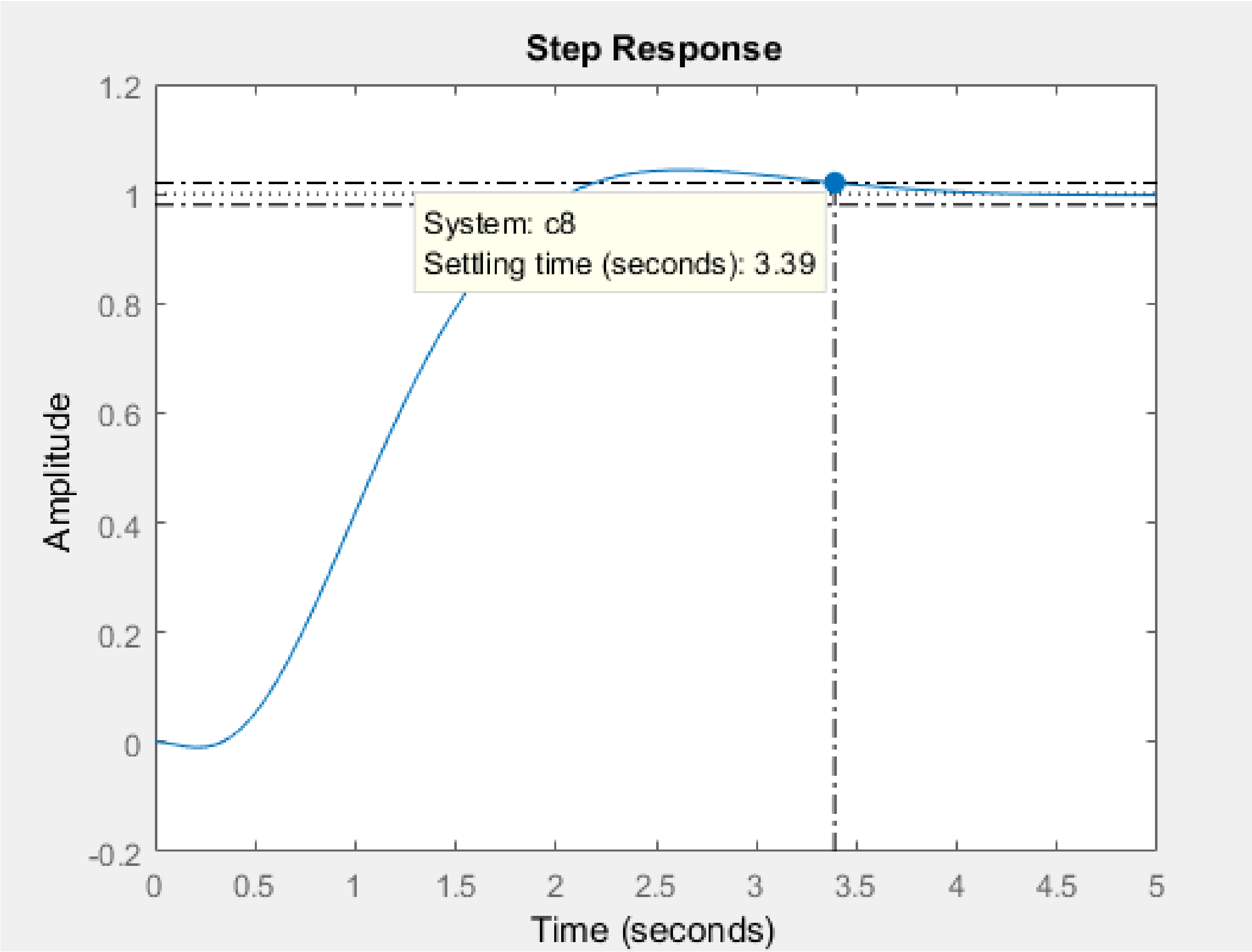


Root Locus
Imaginary Axis (seconds^{-1})
Real Axis (seconds^{-1})
6
4
2
0
-2
-4
-6
-12
-10
-8
-6
-4
-2
0
2



Root locus of the outer loop of the acceleration control system


Step Response
1.2
1
0.8
0.6
0.4
0.2
0
-0.2
Amplitude
0
0.5
1
1.5
2
2.5
3
3.5
4
4.5
5
Time (seconds)


Step Response
System: c8
Settling time (seconds): 3.39
Amplitude
Time (seconds)
1.2
1
0.8
0.6
0.4
0.2
0
-0.2
0
0.5
1
1.5
2
2.5
3
3.5
4
4.5
5

- One of these is that the accelerometer cannot distinguish between the acceleration due to gravity and accelerations caused by aircraft motion.

- The acceleration of gravity can be balanced out so that in straight and level flight at normal cruise air speed and altitude the output of the accelerometer would be zero. However, at different angles of attack the accelerometer output would not be zero.

- For example, if the angle of attack changed by 10 deg from the value at which the accelerometer was nulled, the output would correspond to ± 0.5 ft/sec2

- The accelerometer can be adjusted so that it is insensitive to accelerations that are less than 1 ft/sec^2, thus eliminating this problem.

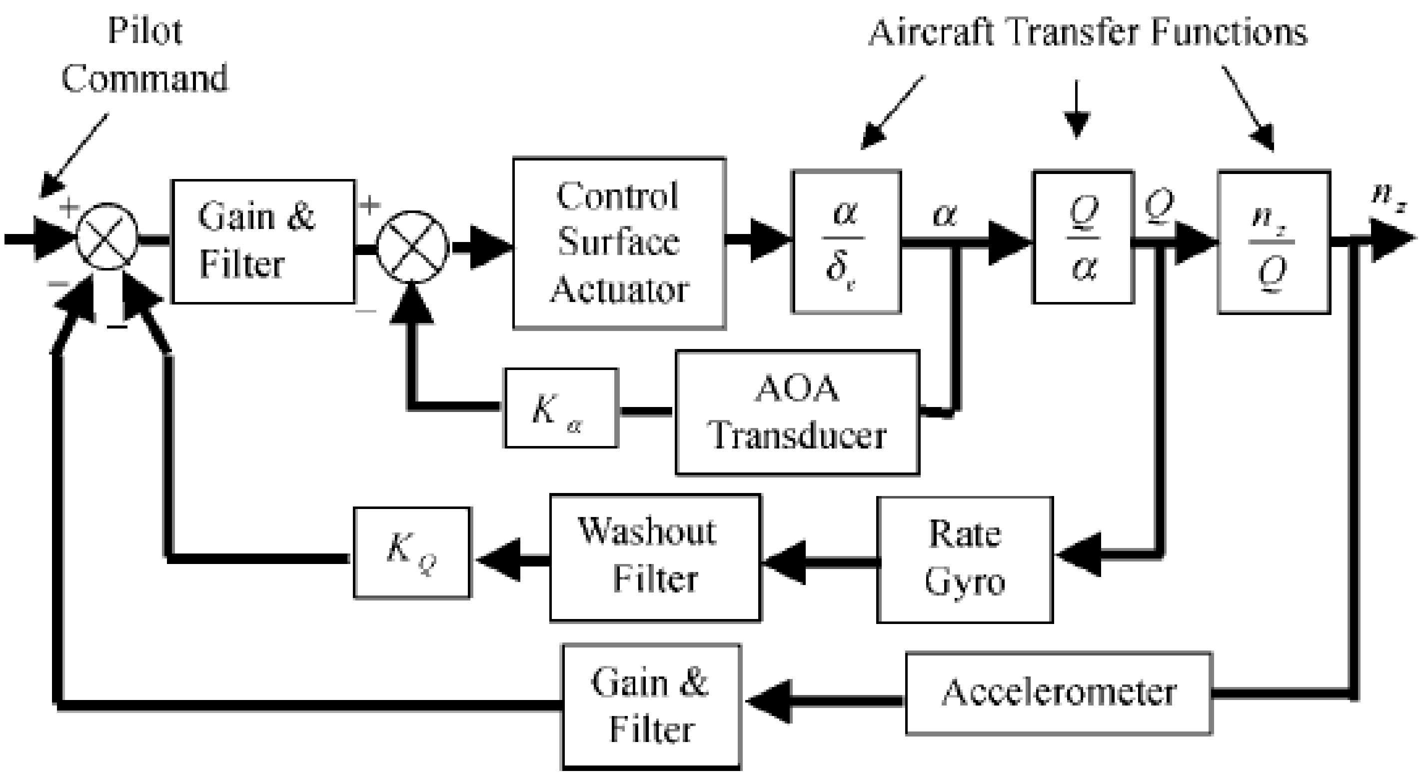
- The another problem which would probably harder to overcome is unwanted acceleration arise because of the turbulence.

- It creates a noise in the measurement, so filter must be introduced in the loop to over come the problem.

- As a result of the above problems the accelerometer loop is not employed in all flight condition. However, during the tactical maneuvers like pull up or pull down maneuvers, the accelerometer loop is engaged.

- It will helps to limit or control maximum allowable acceleration of the aircraft as well as pilot.

- Most importantly the acceleration control system is used for the control surface-to-air guided missile.



Simplified F-16 longitudinal Fly by wire system

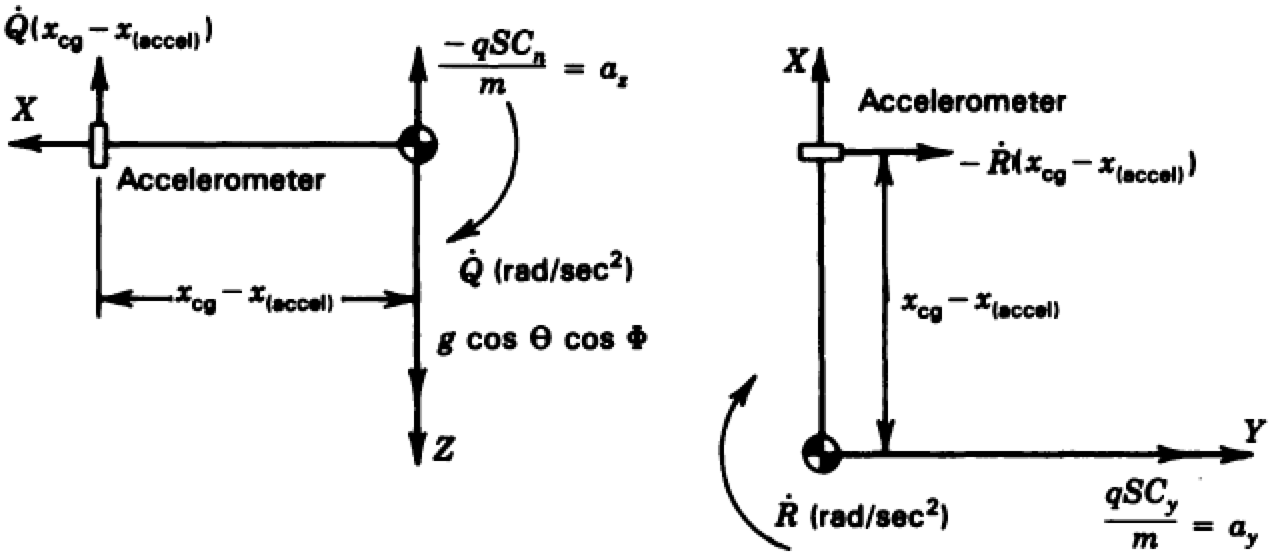
x_a (ft)	Static-Loop-Sensitivity and Numerator Factors
0	$0.03333(s + 6.432)(s - 13.14)$
5	$0.006042(s + 9.171)(s - 50.82)$
6	$0.0005847(s + 10.68)(s - 450.7)$
6.1	$0.00004005(s + 10.90)(s - 6448.2)$
7	$-0.004872(s + 14.73)(s + 39.23)$
15	$-0.04852(s + 3.175 \pm j6.925)$

- The interesting factors of the transfer function is the two zero at the numerator S+6.432 and (S-13.14).
- Because of the NMP zero at s=13.4, the normal acceleration response to a negative step elevator command (aircraft nose-up) will be initially negative acceleration, followed by the expected positive acceleration.
- The physical explanation for the non minimum phase behavior is that the elevator control surface is deflected trailing edge upwards to produce positive acceleration, this create a downward increment force on the tail.
- Due to this the CG of the aircraft may drop momentarily during the pitch up, so the normal acceleration may briefly become negative before it builds up positively.

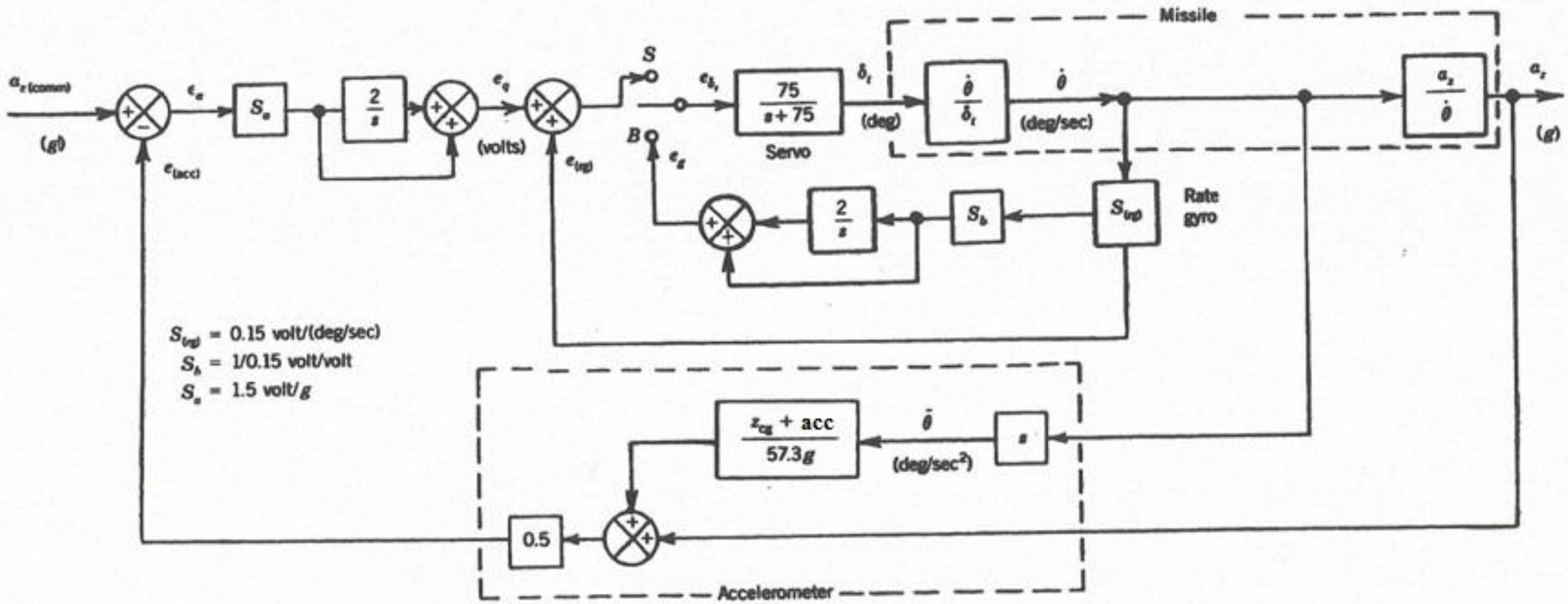
- Table shows that, the acceleration position is moved forward, the NPM zeros moves out towards infinity. (as a result static loop sensitivity gets decreases).

- At a position of near 6.1ft forward of the cg the non minimum phase effect disappears and the point correspond to an" instantaneous center of rotation"

- In modern fighter aircraft the accelerometer are placed close to the pilot desk. It is not exactly at instantaneous center of rotation, so mathematical correction must be applied depends upon the position of the accelerometer w.r.t to CG.



Accelerations sensed by accelerometer located forward of aircraft.



Missile
$a_{z\,(comm)}$
(g)
ϵ_a
S_a
$\frac{2}{s}$
e_q
(volts)
$e_{(acc)}$
$e_{(rg)}$
S
B
e_g
e_{δ_c}
$\frac{75}{s+75}$
Servo
δ_c
(deg)
$\frac{\dot{\theta}}{\delta_c}$
$\dot{\theta}$
(deg/sec)
$\frac{a_z}{\dot{\theta}}$
a_z
(g)
$S_{(rg)}$
Rate gyro
S_b
$\frac{2}{s}$
$S_{(rg)}$ = 0.15 volt/(deg/sec)
S_b = 1/0.15 volt/volt
S_a = 1.5 volt/g
$\frac{z_{cg} + acc}{57.3g}$
$\ddot{\theta}$
(deg/sec²)
s
0.5
Accelerometer

- Fuselage of aircraft and body of missile as subject to bending due to structural vibration and aerodynamics phenomenon.

- It is also important to place the accelerometer close to a node of the most important fuselage bending mode. If this is not done, structural oscillation will be coupled into rigid body control system and degrade the flying qualities or even cause “aero servo-elastic” limit cycle acceleration.

- A disadvantage of normal acceleration feedback is the gain of the transfer function varies widely with dynamic pressure. Accelerometer noise may occurs at very low dynamic speed. So gain has to be boosted heavily in order to achieve the desired handling qualities.

Thank you