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ANALYSIS OF EFFECT OF WAKE VORTICES ON AIR DATA SENSORS

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ABSTRACT

An essential part of operational clearance of a new aircraft is to certify that it is safe to fly even when wake vortices generated by another aircraft are encountered. For certification, it must be demonstrated that the aircraft can safely cross the wake at a certain separation distance. When the wake is encountered, airdata sensors that measure the directional of airflowin terms of Angle of Attack (AOA / Alpha), Angle of Side Slip (AOSS / Beta / SSA) and pressures (Total and Static Pressure) show large fluctuations. The ADS measurements are used for feedback to the fly-by-wire control laws.These large and rapid fluctuations in the ADS signals affects the commands generated by the fly-by-wire control laws and in turn results in generating high loads on the aircraft. Therefore, before actual flight tests it is necessary to predict the behaviour of Air Data System (ADS) onencountering wake. Appropriate wake encounter protection features should be incorporated in the ADS algorithms. The behaviour of the ADS including the wake protection features during wake must be cleared in a simulation environment. The simulation environment for wake analysis, behaviour of the ADS measurements with and without wake protection features are presented in this paper for a generic high performance aircraft. Paper also briefly presents the wake protection featuresincorporated in the ADS algorithms.

Keywords: Air Data System, wake, fighter aircraft, wake clearance

1. INTRODUCTION

A fighter aircraft can inadvertently encounter the wake of another aircraft during formation flying, air-to-air combat or aerial refuelling tasks. Therefore, wake encounters are an occupational hazard for fighter aircraft. There have been serious accidents involving fighter aircraft which have been later traced to a wake encounter. For example, a Gripen fighter aircraft had an accident due to wake encounter in 1999 resulting in loss of the aircraft [3]. This aspect is further exacerbated if the aircraftis designed with a complete authority fly-by-wire flight control system which employs air data sensors for feedback.

Typical Air Data System (ADS) sensors like pressure probes and vanes are exposed to the air stream and therefore pick up the disturbances in the flow due to the wake. In presence of wake, vanes and pressure probes undergo large transients. This in turn propagates into the control law resulting in control surface deflections which may cause departure of the aircraft from controlled flight.

It is well known that the wake of the leading aircraft can be modeled as a pair of counter rotating vortices. The air flow between the two vortex cores adds up resulting in a downwash. Consider an encounter scenario wherethe following aircraft enters the wake in between the two vortex cores. Vanes and pressure sensor will see the downwash and therefore, the AoA feedback path within the control laws will cause a momentary pitch up command to restore the aircraft AoA. As soon as the aircraft exits the wake theADS sensors will recover. At this point the additionalpitch up command from the control laws can lead to aircraft departure into high AoA regime. The influence of wake vortex behind another fighter aircraft is restricted to an area of about 10m by 10m in the plane perpendicular to the vortex core axis. Therefore, even at low speeds, such an encounter will last less than one second. Therefore, the ADS must be designed to reject short term large transients due to wake encounters.

On the other hand, during an air-to-air refueling task behind a large tanker aircraft, the following aircraft will remain immersed within the wake of the former for a significant amount of time. At the same time, the ADS must also have a mode to cater for the cases where the aircraft is immersed in the wake of the tanker for long periods of time.In case of the latter, the ADS switches to AoA derived from inertial measurements.

This paper presents the analysis of effect of wake on fly-by-wire system whose control is based on Angle of Attack (AOA/Alpha), Angle of Side Slip (AOSS/SSA), static pressure (Ps) and total pressure (Pt) feedback. The Ps and AoA signals generated within the ADS for use as feedback signals within the control law are synthesized by a suitable fusion of measured sensor values with signals derived from inertial measurements (accelerometers and gyros).

The organization of the paper is as follows: Section 2 describes the elements of the ADS and Wake model considered for the simulation and analysis. Section 3 presents the wake protection features that are incorporated

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in the ADS algorithms and the relevant results. Section 4 gives details on the simulation environment. Section 5 presents results of the wake situations considered through various tests on ADS measurements.Section 6 concludes the paper.

2 AIR DATA SYSTEM AND WAKE MODEL

This section describe element of ADS and wake model. Figure 1 shows the block schematic of the integrated Wake generation, Inverse ADS and Forward ADS model that have been consideredin this paper for simulation and analysis.

2.1 Air Data System

Air data system model as implemented in simulation environment consists of two modules:

2.1.1 Inverse ADS Module

The inverse ADS module generates the local measurements of AOA, AOSS, Pt and Ps as sensed by the sensor at a given flight condition in terms of free stream parameters, like Mach No., Altitude, AOA, and AOSS. It takes into account the position error correction for different sensors due to the presence of aircraft body in the flow field. As the wake is different at each point, so each sensor will see different flow parameters. This inverse model is primarily used for: (i) generate the test cases for analysis of the ADS forward model, and (ii) for evaluation of fly-by-wire flight control systems at various on ground test platforms.

2.1.2 Forward ADS Module

The forward ADS algorithms that are implemented in the on board computer include:

(i) correction to the local measurements received from sensors to obtain free stream signals, and

(ii) Redundancy management logic which include detection of sensor failure and selection of healthy signal for feedback to control laws and displays for navigation. The sensor failure is identified in two ways: (i) due to data path failure (due to cable cut or electrical failure of the computer etc.) and (ii) due to mistrack between similar types of signal from multiple sensors based on a priori set thresholds.

(iii) The forward ADS algorithms also include the following features for wake protection: (i) Rate limiting on the signal to avoid sudden jumps or spurious spikes, (ii) Inertial filtering scheme wherein alternate source of the reference signal is used. In case of AOA, equivalent AOA computed from either pitch rate or inertial velocities is used as reference during wake encounter. In case of Ps inertial filtering scheme, the vertical velocity is used to derive the equivalent reference signal.

Therefore, by analysis of ADS, one can know the value of parameter that will be used in feedback and different sensors failure, under different types of wake encounter.

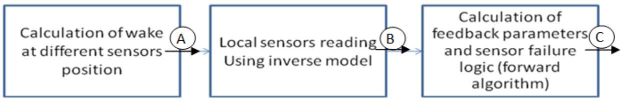
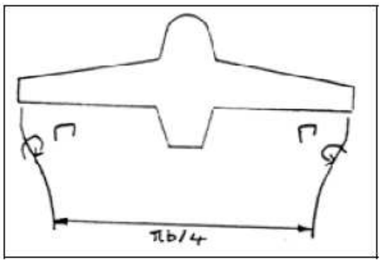


Figure 1. Integrated Wake + Inverse ADS + Forward ADS Model

2.2 Wake Model

Induced wake velocity is modeled using Burnhan-Hallock model [1]. In this model it is assumed that two vortex systems with distance between their cores (b_v) is equal to (πb/4) where b is the wing span of the wake generating aircraft.



Vortex circulation (Γ) is the measure of the strength of a vortex. The strength of a fully developed single vortex is given by the following first order approximation:

$$\Gamma = \frac{W}{\rho . V . b_v} \qquad \dots (1)$$

where

W = n_Z.m.g,

n_Z = load factor (in ‘g’ unit), m = mass (in kg) of the wake generating aircraft

ρ = air density (kg/m³)

V = true airspeed of the wake generating aircraft (meter/sec)

b_v = πb/4 (meter) is the distance between cores of the two vortices

2.2.1 Burnham-Hallock Model

According to the model velocity induced by a single vortex is given by

$$V_t(r) = \frac{\Gamma}{2\pi}\frac{r}{{r_c}^2+r^2} \qquad \dots (2)$$

Where$r_c$is core radius. The core radius can be approximated with the following equation:

$$r_c = 0.0125\sqrt{\Gamma\frac{d}{V}}$$

Where d is separation distance between the two aircrafts.
So the effect of both vortices can be considered by vector addition of induced velocity by both vortices.

The circulation causes a radial velocity distribution within the vortex; it features sharp peaks and rapid changes in sign. Effect of different load factor of leading aircraft on induced velocity is shown in figure 2. Total vortex velocity distribution at different distances from the leading aircraft is shown in Figure 3.

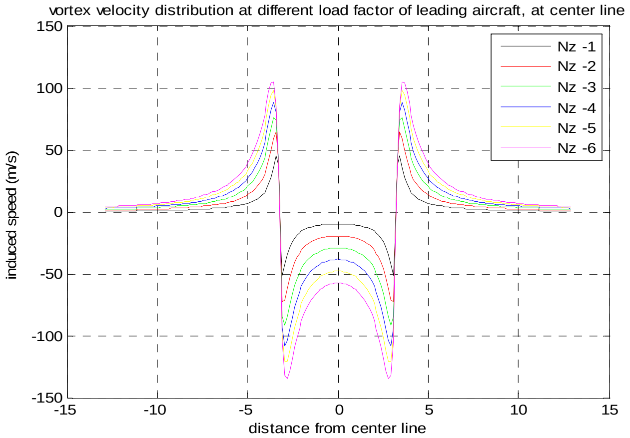


Figure2. Vortex velocity distribution at different Load factors of leading aircraft

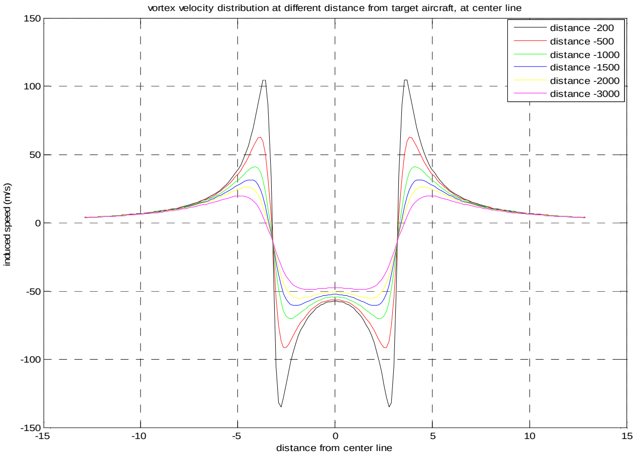
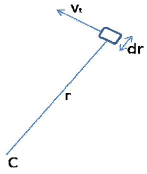


Figure 3. Vortex velocity distribution at different distances of wake encounter for Nz=6

2.2.2 Pressure Distribution Due to Wake Considering Induced Velocity as Burnham Hallock Model

Due to centripetal force of circulating wake, pressure distribution in atmosphere changes. Pressure distribution due to centripetal force is modeled using assumption that air is to be incompressible (density constant).



From the adjacent figure, C is the center of core of vortex. At a distance r from C, pressure gradient with respect to distance r can be given by

$$\frac{dp}{dr} = \varrho\frac{{V_t}^2}{r} \qquad \dots (4)$$

Where V_t can be obtained by equation 2 and ρ is density of air. By integrating for distance r to infinity

$$\int_{P_{r^*}}^{P_\infty} dp = \varrho\left(\frac{\Gamma}{2\pi r_c}\right)^2 \int_{r_c^*}^{\infty} \frac{r^*}{(1+{r^*}^2)^2}\,dr^* \qquad \dots (4)$$

Where

$$r^* = \frac{r}{r_c}$$

Integrating equation 4:

$$P_\infty - P(r^*) = \varrho\left(\frac{\Gamma}{2\pi r_c}\right)^2 (I_\infty - I_{r^*}) \qquad \dots (5)$$

Where

$$I_x = \int_0^x \frac{r^*}{(1+{r^*}^2)^2}\,dr$$

3. WAKE PROTECTION FEATURES IN ADS ALGORITHM AND RELEVANT RESULTS

Various wake protection features incorporated in the forward ADS algorithms are listed in Section 2.1.2. In this section, the details on the AOA rate limiter and the AOA inertial filtering scheme including relevant results are presented. Due to space constraint, the Ps rate limiting and Ps inertial filtering schemes are not presented in this paper.

3.1. AOA Rate Limiter

Identification of wake encounter situation is done based on the exceedance in the rate of change of AOA signal beyond the normal range. A fixed value has been used for AOA rate limiting. Whenever the rate of the AOA signal exceeds the specified value, it gets rate limited. This avoids sudden jumps or spurious spikes in the signal. Upon rate limit, a discrete signal is generated which is later used in AOA inertial filtering scheme to take appropriate action.

3.2. AOA Inertial Filtering Scheme

Since the AOA signal from the ADS is corrupted in the presence of wake, an alternate equivalent reference signal is generated using data from the inertial sensors mounted on the aircraft whose measurements are not significantly affected due to wake disturbance. This alternate signal is then used for feedback and display. AOA inertial filtering scheme is shown in Figure 4 wherein equivalent AOA is obtained from the pitch rate signal. The signal obtained after washout filter of the pitch rate referred to AOA_dot (designated as Q_dot in Figure 4) is passed through a low pass filter that generates the high frequency content of AOA signal and AOA from vanes is passed through a low pass filter. Thus, the blend of the two signals, i.e., AOA from ADS and AOA from Q_dot is used to generate the equivalent AOA during wake encounter.

After a fixed amount of time that exceeds the normal wake encounter time, the corrected signal from the ADS sensor is selected back for feedback and navigation displays. To avoid abrupt changes, the transition between two signals is made smoothly over a specified time by using fader logic. Figure 5 shows plots of AoA with and without wake perturbance in AoA inertial Filtering scheme. In the selected flight data segments an external wake input was injected for a specified time window. The following legends are used in the figures showing the results related to AOA inertial filtering scheme:

- ‘AOA_in_no_wake’ is the signal corresponding to AOA obtained from flight data. It does not include wake effects, since no wake was actually encountered in flight.
- ‘AOA_in_with_wake’ is the input signal to the ‘AOA inertial filtering scheme’ with simulated wake/large turbulence effects added to ‘AOA_in_no_wake’. Wake is simulated as a pulse input of 15 deg local AOA magnitude and time duration of 1.5 sec. This is a typical variation in the ADS signals during wake encounter as can be seen in the results presented in Section 5.
- ‘AOA_from_qdot’ is the signal obtained after washout filtering of the pitch rate signal generated (Q_dot).
- ‘AOA_after_Fader’ is the selected AOA signal obtained after fader. This is the output of rate limiter block.
- ‘AOA_wake_hit_flag’ is a discrete signal. When this flag goes from 0 to 1, it indicates that at least one vane AOA signal has hit the rate limit, which in turn indicates a wake encounter situation. This flag is forced to stay at 1 for 2 seconds and then brought back. This ensures that the wake encounter would be over when direct ADS sensor inputs are used again.

Before ‘AOA_wake_hit_flag’ is set (i.e. before wake encounter), and after the flag is reset, the ‘AOA_after_Fader’ becomes same as ‘AOA_in_no_wake’. As can be seen in Figure 5, the ‘AOA_after_Fader’ gradually changes (over 0.5 second) from AOA_in_with_wake’ to ‘AOA_from_qdot’ when ‘AOA_wake_hit_flag’ is set. The ‘AOA_after_Fader’ signal gradually changes (over 0.5 second) from ‘AOA_from_qdot’ to AOA_in_no_wake’ when ‘AOA_wake_hit_flag’ transits back to zero after 2 seconds.

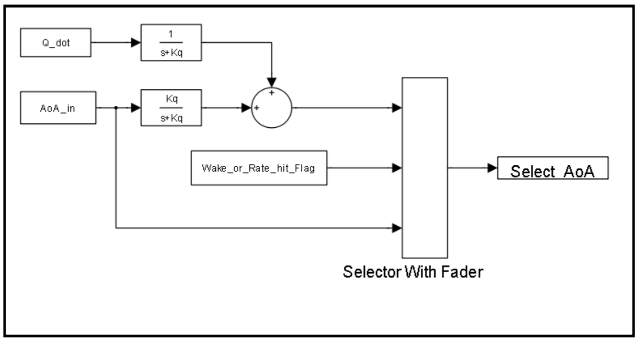


Figure 4. AOA inertial filtering scheme

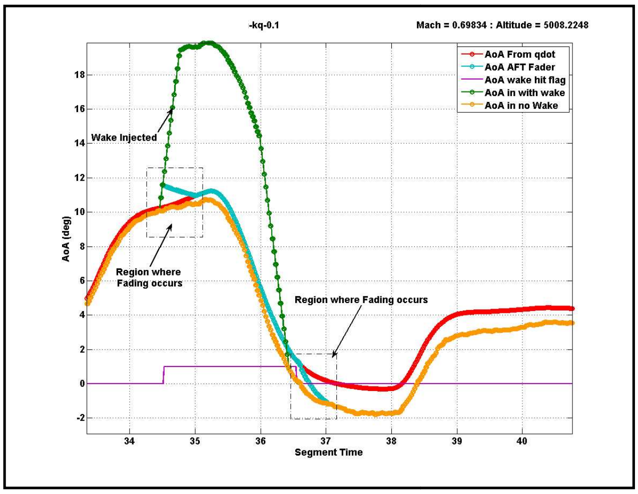


Figure 5.Comparison of AoA with and without wake perturbance in AoA inertial Filtering scheme

3.2.1 Tuning of the Filter Constant

It is desirable to have a good match between 'AOA_from_qdot' and AOA_in_no_wake. However, depending upon the different orientation of the follower aircraft in terms of body rates and angles, the 'AOA_from_qdot' deviates from the AOA_in_no_wake. The filter constant 'Kq' plays an important role in shaping of inertial filtered AOA signal. Therefore, tuning of the filter constant Kq is crucial.

4. SIMULATION ENVIRONMENT FOR WAKE ENCOUNTERS

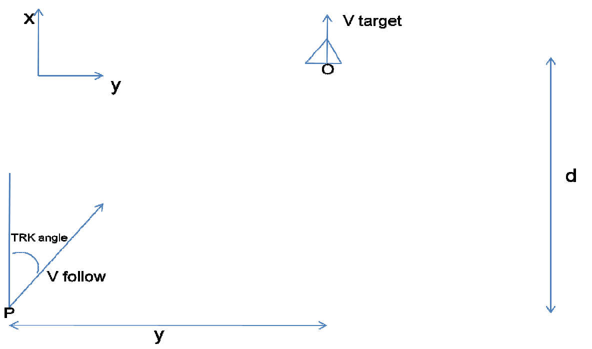


Figure 6. Modelling of following aircraft crossing the wake at different track angle

The scenario considered, is that of a point mass (follower aircraft, point P in Fig. 6) crossing the wake of the leading aircraft (point O in Fig 6, currently at distance d ahead and y to the right of P) for different track and bank angles at vertical separation.

The sensors are at some fixed relative distance from the c.g. of the point mass model of following aircraft in Figure 6. Analysis has been performed for the planar case only (both aircraft are moving in X-Y plane only). Following aircraft crosses the wake at different track angles (TRK angle in Fig. 6). The leading aircraft moves with speed V and following aircraft with V_f. It is also assumed that the follower aircraft is flying at fixed AoA and SSA and that are not changing with respect to the free stream condition.

For different cases, initial distance y was selected such that the following aircraft should enter the left wake at the same time (approximately at 0.8 sec). This was done to ease the analysis of all cases with a common time of wake entry. Initial distance d was selected such that aircraft meet the first vortex at the desired X distance.

4.1 Sensors Considered

Aircraft is assumed to have 6 different sensors for reading flow angles (AOA, AOSS), static and dynamic pressures. Each sensor will face different wake depending upon their location on the aircraft. These sensors are:

1. Left Vane (LV): for flow angle
2. Right Vane (RV): for flow angle
3. Beta Vane (BV): for flow angle
4. Left Probe (LP): static, dynamic pressure and angle
5. Right Probe(RP): static, dynamic pressure, angle
6. Nose Probe (NP) : static, dynamic pressure, angle

4.2 Test Case

Test case has been chosento capture the worst case wake encounterand for this the following parameters are chosen such that the circulation produced by lead aircraft is maximum:

1. For lead aircraft Nz =6
2. Lead aircraft speed V =120m/s
3. Phi: for extreme case 0 and 90deg
4. Density = 1.225 kg/m^3 (Sea level)
5. Simulation for different Mach number and angle of attack
6. AOSS =0 deg

7. Height of follower aircraft from center line =0
8. A default Mach number of 0.25, track angle 15 deg and wake encounter distance of 400m has been chosen to display the results unless the results for variation of track angle, distance or Mach number are needed.

5. RESULTS OF THE DIFFERENT WAKE SITUATIONS CONSIDERED THROUGH VARIOUS TESTS ON ADS MEASUREMENTS

Simulation test cases are divided into two parts:

1) Follower aircraft crosses the wake at 90 deg. bank angle at different track angles ranging from 15 deg to 90 deg in steps of 15 deg.
2) Follower aircraft crosses the wake at 0 deg bank angle at different track angles ranging from 15 deg to 90 deg in steps of 15 deg.

The results presented in this section are the direct effects of the wake on ADS measurements, i.e., without wake protection features.

5.1 Aircraft is Crossing the Wake at 90deg Bank Angle at Different Track Angles

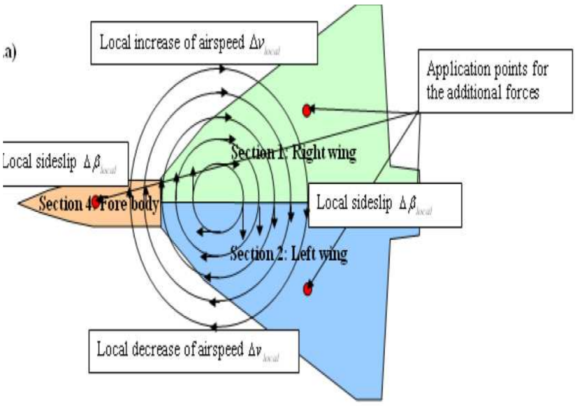


Figure 7.Following aircraft crossing the wake at 90deg bang angle at different track angle.

This case is shown in Figure 7. Lead aircraft is assumed to be going into north direction. For follower aircraft, pitch angle is assumed to be equal to zero, yaw angle is equal to track angle and bank angle to be 90 deg (for Euler angles).

Figure 8, shows the AOA reading for LV and RV sensor as a function of time for different distances of wake encounter. It is clear from Figure 8that as distance increases, there is huge drop in maximum peak error.

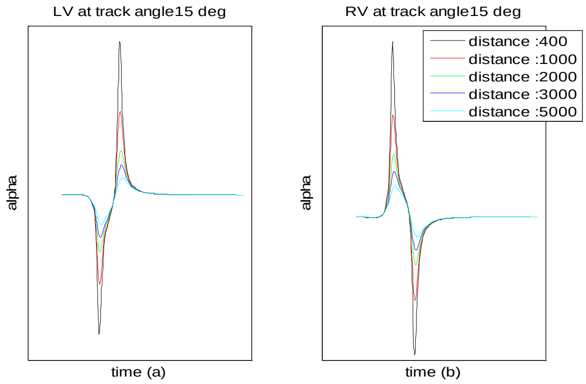


Figure 8. Different sensors reading of alpha at track angle 15deg for free stream AOA and AOSS=0

Figure 9 shows the left vane AOA readings for different track angles. It can be seen that increasing the track angle, error in AOA reading decreases. Also as track angle increase, time to cross the wake also decreases.

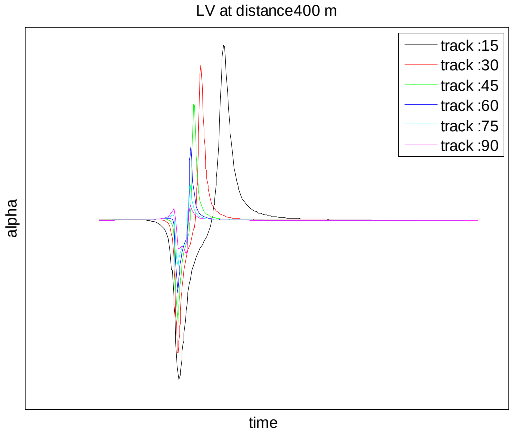


Figure 9. Left vane reading for free stream AOSS=0, AOA=0 for different track angles and wake encounter at 400m distance

Figure 10 shows the AOSS reading of beta vane as aircraft crosses the wake for different track angle for free stream AOA and AOSS=0. It can be seen that maximum AOSS deviation is limited to a threshold value after which it freezes the value in ADS.

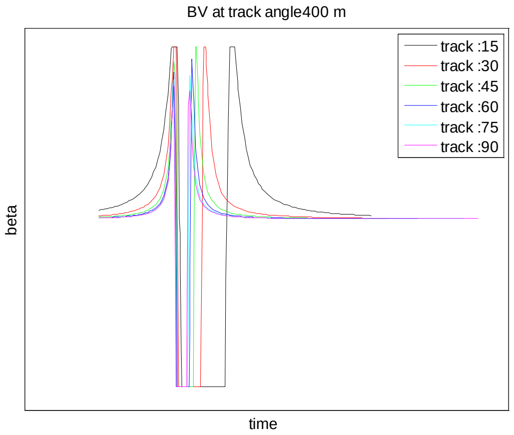


Figure 10. AOSS vane reading for free stream AOSS=0, alpha=0 for different track angles and left wake encounter at 400m distance

Figure 11 shows the beta readings of different sensors for encounters at different distances.

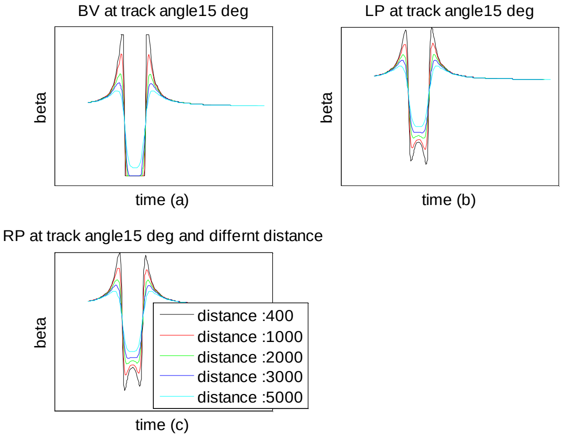


Figure 11. Different sensors reading of beta at track angle 15deg for free stream AOA and AOSS=0

Figure 12 and 13 shows the feedback value of AOA and side slip angle respectively after forward algorithm for encounter at different distances.

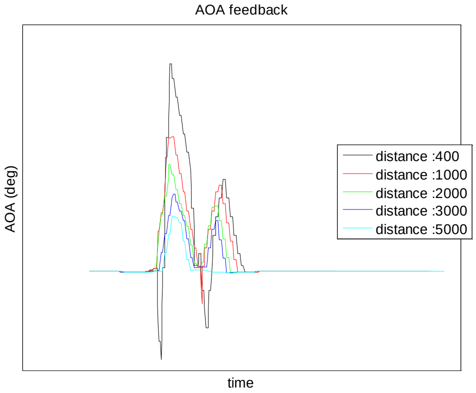


Figure 12. Value of AOA feedback after forward model

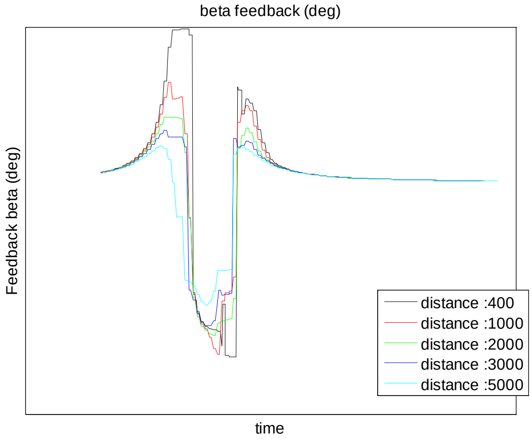


Figure 13. Value of side slip angle feedback after forward model

Figure 14 and 15 shows the feedback value of total pressure and static pressure respectively after forward algorithm. Results are for encounter at different distances.

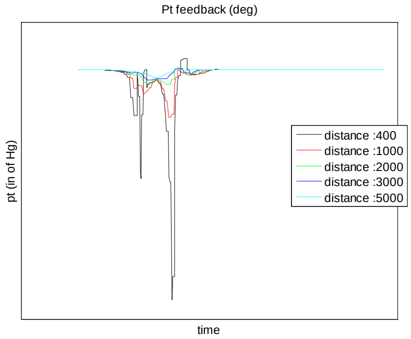


Figure 14. Value of total pressure feedback after forward model

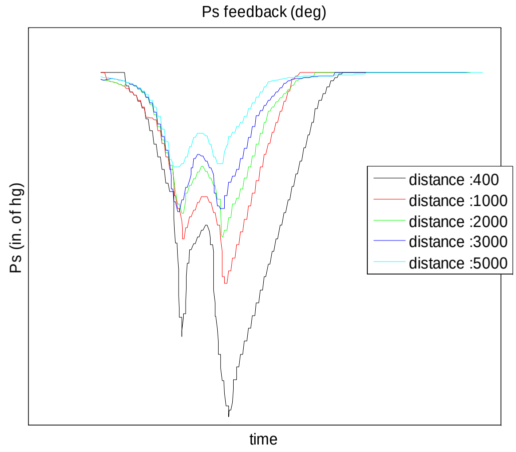


Figure 15. Value of static pressure feedback after forward model

Figure 16 shows the effect of wake protection on AOA readings. Figures 16 (a) and (b) show the effect of rate limit on LV and RV readings (blue plot: before rate limit, red plot: after rate limit). Figures 16 (c) and (d) show the effect of rate limit on side probe reading and final selected AOA. Rate limiter has helped in removing the spurious spikes in AOA reading that otherwise affect the commands generated from control laws.

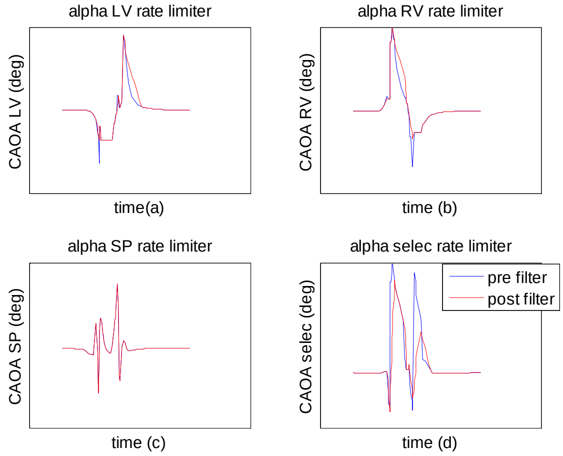


Figure 16. Effect of rate limiter in reducing the AOA spikes in sensors value due to wake

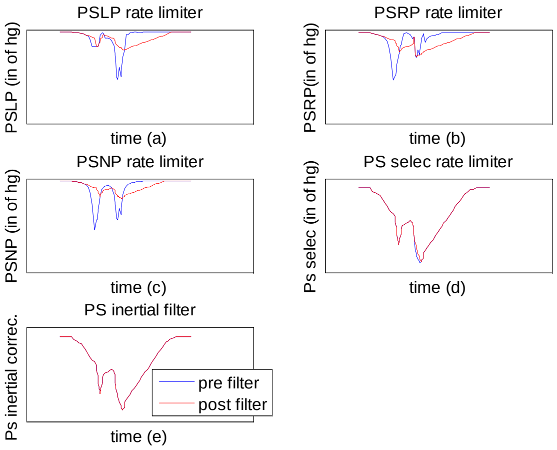


Figure 17. Effect of rate limiter in reducing the pressure spikes in sensors value due to wake

Figure 17 shows the effect of wake protection on static pressure readings. Figures 17 (a) and (b) show the effect of rate limit on LP and RP readings (blue plot: before rate limit, red plot: after rate limit). Plot in Figures 17 (c) and (d) show the effect of rate limit on NP and selected pressure (selected pressure from LP, RP, and NP). Figure 17 (e) shows the inertially filtered Ps. Rate limiter has helped in removing the spurious spikes in Ps reading that otherwise affect the commands generated from control laws.

Figure 18 shows the contour plot of maximum error in AOA reading for left vane, right vane and nose probe during the wake encounter at 400m distance. Against each plot, the colored scale is also shown to identify the magnitude of the parameter in the two dimensional contour plot. It can be seen that as Mach number increases, the error in sensors reading decreases. This can be explained by the fact that aircraft speed becomes high as compared to wake velocities. As expected nose probe reading error is zero in entire envelop (Figure 18(c) has a scale of 10^{-15}).

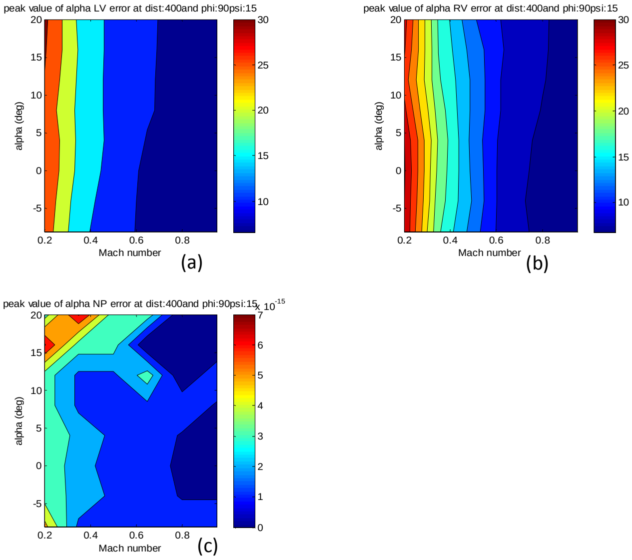


Figure 18. Contour of maximum possible error in different sensors during wake encounter as a function of free stream Mach number and AOA of following aircraft

Figure 19. shows the result error contour for AOSS reading across multiple sensors. Nose probe, left probe and right probe AOSS reading are independent of angle of attack for this case.

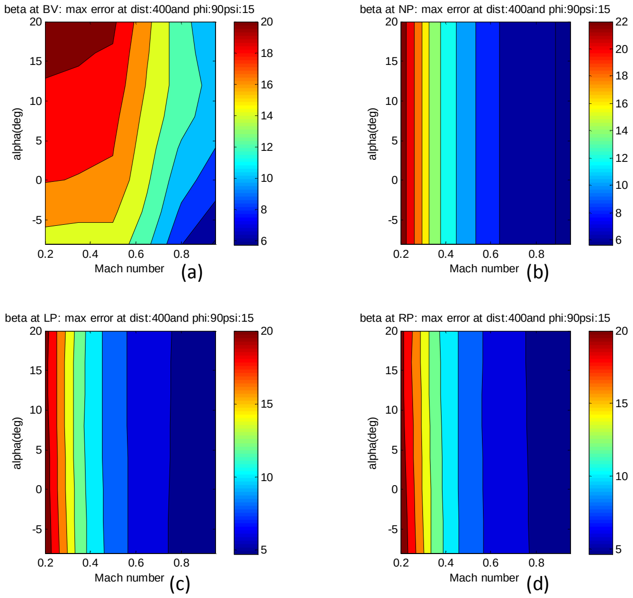


Figure 19. Contour of maximum error in reading for AOSS at different sensors

Figure 20 shows the result of error in AOA feedback. As expected, it can be seen that maximum error is at low Mach number range for this case.

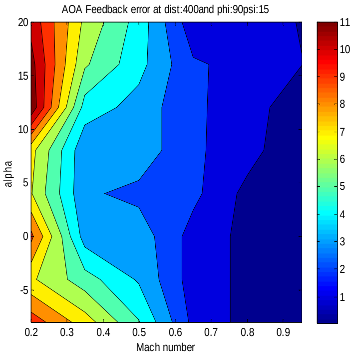


Figure 20. Contour of maximum possible error in feedback AOA as a function of Mach number and free stream AOA

Figure 21. shows the contour plot of maximum error in AOSS feedback for different AOA and Mach number. For the case considered under study, the maximum error in AOSS is of 14 degree which occurs at low Mach No. andnegative values of AOA.

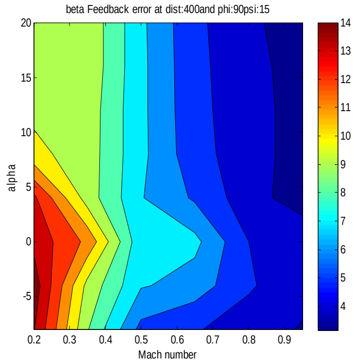


Figure 21. Contour of maximum possible error in feedback AOSS

5.2. Aircraft is crossing the wake at 0deg bank angle at different track angle

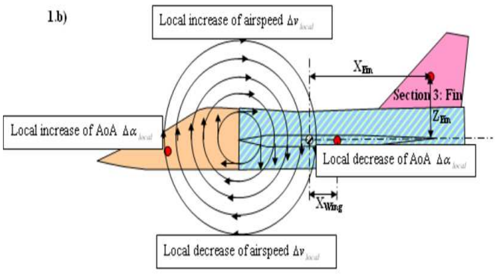


Figure 22. Following aircraft crossing the wake at 0deg bank angle at different track angle

This case is shown in Figure 22. Leading aircraft is assumed to be going into north direction. So for Euler angles of following aircraft, pitch angle is assumed to be equal to angle of attack, yaw angle is equal to track angle and bank angle to be 0. The analysis of the effects of wake on ADS measurements have been carried out in the similar manner as that has been done for the first case in this section.The results for this case are not presented here.

6. CONCLUSION

Paper presents complete analysis methodology for ADS system of a generic aircraft under wake encounter of another aircraft. Analysis has been extended to full flight envelop to give complete picture of behavior of individual sensor under wake and output feedback parameter that are used by fly-by-wire control laws. Analysis is very useful for clearance of the wake penetration related flight testing and also helps to analyze the effects of the wake protection features in ADS algorithms during wake encounter scenario. The AOA inertial filtering scheme presented is useful during wake encounter to have a good signal for feedback and navigation displays.

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