Abstract

The present project aims to the analyses and model the auto-navigation process. Putting mathematical model for aeroplane auto-navigation requires an understanding to the aeroplane mobility in atmosphere, and its relationship with the control surfaces in the aeroplane. After the accomplishment of the analysis and modeling, an auto-navigation control system was designed by using the devloped model. The developed control system is oriented to be a navigator to auto-control the aeroplane. In addition to the establishment of navigation programs, a graphical user interface was developed to display the state of the aeroplane. Hence the observer would have a chance to monitor the events which the aeroplane may undergo during its flight. In order to test the accuracy of the navigator's performance, a simulation model was developed and impelemented with the aid of the actual input data the simulator was utilized to investigate the navigator's response.

The encourageing results of the simulator paved the way to start improving the conducted work. An adaptive method had been suggested, so that the response of the aeroplane become dynamic instead of the static response which is already present. The dynamic response was curried out by two automatic methods which both depend on the amount of deviation between the actual airplane response and the planned response. The results of the dynamic response were more accebtable. The next accomplished step was modeling and simulating the external effects that would influence the performance of the aeroplane in the atmosphere. By studying the wind effect on the aeroplane's performance during its flight, and analyzing the aerodynamic forces that would deviate the aeroplane

from the planned routes, an algorithm was added to the simulator to determine the actual speed of the aeroplane after considering the wind speed and the wind force on the aeroplane's head. The efficiency of the aeroplane's engine has been modeled and added to the simulator taking into account the type of the used fuel. The last effective factor included in the simulation is the effect of the elevation and the amount of the oxygen concentration on the performance efficiency of the engine, which in turn will affect the aeroplane fly efficiency. This was curried out by proposing a simple mathematical model by which the simulator could adopt its performance at various elevations.

In general, the simulation results were encouraging, as described by the behavior test results under different conditions related to atmosphere and engine efficiency at various elevations. In addition, some unpredicted and interesting situations were investigated (such as the sudden motion of the aileron or change in its directions). The test indicates that the promoted model and the proposed methods are quite efficient.

Appendix (A)

Because of the Earth is oblate, its radius at northern pole is 6362 *km*, while it is 6391 *km* at equatorial line. So, its radius at the area surround at point have the altitude θ (i.e., area defined by $\theta_1 \le \theta \le \theta_2$ as shown in Figure A_1) could be approximately determined as follows:

$$\theta = \frac{\theta_1 + \theta_2}{2}$$

$$X = 6391 \cos\theta, \text{and } Y = 6362 \sin\theta$$

$$R = \sqrt{X^2 + Y^2}$$

note that at region such Iraq, θ_1 is 28°, θ_2 is 37°, and consequently the value of R will be 6372 *km*.

The approximate perimeter of the Earth is equal to $2\pi R$, by which one can determine the average distance (d_1) between each successive two latitude circles as follows:

$$d_1 = \frac{\pi \times 6372}{180} \approx 112\, km$$

The planatic perimeter is $\pi R \cos\theta$ as shown in Figure (A_2), which help to determine the average distance (d_2) between two successive longitude circles as follows:

$$d_2 = \frac{2\pi \times 6372 \cos(33)}{360} \approx 91\, km$$

Therefore, the matrix that transforms the (*Lat*, *Lon*) system to Cartesian could be approximated as follows:

$$\begin{bmatrix} X \\ Y \\ Z \end{bmatrix} = \begin{bmatrix} 91 & 0 & 0 \\ 0 & 112 & 0 \\ 0 & 0 & 1 \end{bmatrix} \begin{bmatrix} Lon \\ Lat \\ Height \end{bmatrix}$$

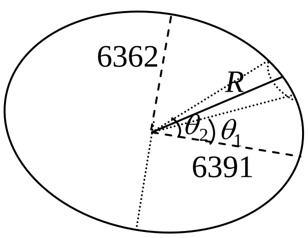


Fig (A_1) Vertical cross-section.

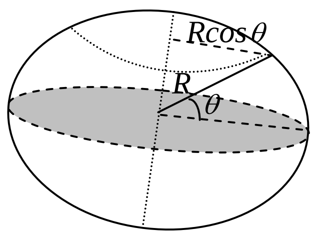


Fig (A_2) Planatic cross-section.

جمهورية العراق
وزارة التعليم العالي والبحث العلمي
جامعة النهرين
كلية العلوم

المنظومة الملاحية ألديناميكية

للطائرة المُسَيرة

أطروحة

مقدمة إلى كلية العلوم في جامعة النهرين كجزء من
متطلبات نيل شهادة دكتوراه فلسفة
في
الفيزياء

من قبل

محمد صاحب مهدي الطائي

(بكالوريوس 1995)
(ماجستير 2002)

محرم 1428 هـ
شباط 2007 م

المستخلص

يهدف المشروع الحالي إلى تحليل ونمذجة عملية الملاحة الآلية (Auto-navigation). إن وضع النموذج الرياضي للملاحة الآلية للطائرة يستلزم فهم حركية الطائرة في الجو وعلاقتها بأسطح القيادة (Control surfaces) في الطائرة. بعد إكمال التحليل والنمذجة، تم تصميم منظومة سيطرة ملاحية آلية باستعمال نموذج مطور، إن منظومة السيطرة الآلية تعمل كملاح (Navigator) يسيطر آليا على الطائرة. بالإضافة إلى بناء برنامج الملاحة، تم تصميم واجهة مستخدم رسومية (Graphical user interfaces) لعرض أحوال الطائرة، وبذلك يتسنى للمراقب مشاهدة الأحداث التي تصيب الطائرة أثناء رحلتها. ولأجل اختبار دقة أداء الملاح، تم تطوير وتنفيذ نموذج محاكاة ببيانات إدخال واقعية ليكون المقلد (Simulator) الذي يلعب دور الطائرة للاستفادة منه للتحقيق في استجابة الملاح.

إن النتائج المشجعة للمقلد هيأت للبدء بتطوير العمل وذلك باقتراح طرق لجعل استجابة الطائرة متغيرة (Dynamic response) بدلا من استجابتها الثابتة (Static response) الموجودة أصلا. الاستجابة المتغيرة تمت بالطريقتين اليدوية والآلية (Manual and Automatic)، وكلاهما يعتمدان على مقدار الانحراف الحاصل بين استجابة الطائرة الحقيقية والاستجابة المخطط لها. إن نتائج الاستجابة المتغيرة كانت مقبولة جدا، ولذلك فأن الخطوة التالية كانت نمذجة ومحاكاة المؤثرات الخارجية التي ممكن أن تؤثر على أداء الطائرة في الجو.

بدراسة تأثير الريح على أداء الطائرة أثناء الطيران وتحليل القوى الايروديناميكية التي ممكن أن تحرف الطائرة عن أوضاعها المطلوبة، تم إضافة خوارزمية للمقلد لتحديد السرعة الحقيقية للطائرة بعد تصفيتها من سرعة الريح ومدى تأثر موقع وسرعة الطائرة بقوة الريح. كذلك فأن كفاءة محرك الطائرة ونوعية الوقود المستخدم قد نمذجت واضيفت إلى المقلد. التأثير الأخير الذي شمل في المحاكاة هو تأثير الارتفاع ومقدار تركيز الأوكسجين على كفاءة أداء المحرك، والذي بدوره يؤثر على كفاءة طيران الطائرة، وذلك عن طريق معالجة رياضية بسيطة يعكس من خلالها المقلد تصرفه عند ارتفاعات مختلفة.

بشكل عـام،فأن نتائج المحاكاة كانت مشجعة وهذا ما تصفه نتائج اختبار المقلد عندما يواجه ظروف مختلفة من حالات الجو وكفاءة المحرك وبارتفاعات مختلفة. بالإضافة إلى ظهور بعض الحالات غير المتوقعة والملفتة للانتباه (مثل مناورة الطائرة ضد الريح). إن نتائج الاختبار تدل على الكفاءة الجيدة للنموذج الرياضي المطور والطرق المعتمدة.

CHAPTER FIVE

CONCLUSIONS

AND SUGGESTIONS FOR

FUTURE WORK

Chapter Contents

CHPTER FIVE
CONCLUSIONS AND SUGGESTIONS

5.1 Conclusions

Flight simulator is the most important tool to achieve high performance requirements of the auto-navigation system for UAV. From the discussion remarks, mentioned in previous chapters, the following conclusions were derived:

1. The mathmatical treatment used in the suggested model was sufficient to describele the behavior of the airplane mobility.
2. The established navigator model was efficient, since its issued commands to the simulator let him behaves smoothly.
3. The simulator passed the tests of playing the role of the airplane during the flight, and this conclusion was attained due its systematic response to different navigation cases, and to its successful interaction with the navigator.
4. The dynamic response shows better performance since it saves time by reaching the intended attitude in shortest time.
5. The adopted two methods of the dynamic response are successful, because they give good stability beside reducing the time of flight.
6. The wind always push the airplane, so it shifts the position of the airplane by an amount proportional to the wind speed.
7. The wind rotates the airplane about its longitudinal axis by an amount proportional with the roll angle and the direction of the wind with respect to the direction of the airplane.

8. The established navigator deals with the effective wind successfully, because it is enabled to compensate the fallback in the speed and repair the positioning deflection.
9. The engine efficiency, fuel quality, and altitude effects are factors affect the power or thrust of the engine that needed to make the airplane reaches the intended speed.
10. The dynamic response makes the coalition work of the combined effect approaches to zero, unless the actual speed fallback from the intended one.

5.2 Suggestions for Future Work

The following topics are suggested for future work in the field of the auto-navigation and flight simulation:

1. The present model can be reformulated to applied on any robotic or technological industrial mobile in the three-dimensions (such as astronautic ship or rocket), two dimension (such as car or boat), or one dimension (such as train or metro). The degree of freedom of the platform that intended to be guided refers to the number of allowed transitional and rotational motions, which aid to determine the control parameters and the median parameters that inter-relate the control parameters by the situations of the mobile.
2. More numerical analyses can be conducted on the developed navigator when the pitch and roll angles are considered to be changed within the range 360^o. In this case, there are more associated situations should be considered, one of such cases is when the pitch is changed by 180^o, the head will be gained 180^o.

CHAPTER FOUR

RESULTS AND ANALYSES

Chapter Contents

CHAPTER FOUR
RESULTS AND ANALYSES

4.1 Preamble

The implementation of the simulation had not depended on the logical results only, but it was dealt with an actual input data, which had simplified the excess to the next step (i.e., practical application). Also, the viewer watch the simulation does not sense the naked numbers, but he can watch the behavior of the simulator in a proper correct way of the simulation.

The navigator can sense the simulator during the flight test, it reflects the flight behavior to the viewer through a set of graphical gauges that included in navigator graphical interface. The possibility of viewing the flight replay will help the viewer to get an indication about the navigator (controller) efficiency.

This chapter includes a qualitative documentation for the behavior of the simulator; starting from the designing phase of the navigator and reaching to the finishing or completion phase. It is concerned with analyzing the effect of the surrounding conditions on the simulator performance. Therefore, the results of the simulator behavior were presented, and the amount of obeying the simulator to the navigator commands was noticed and analyzed. Also, the behavior of the simulator under different wind direction and speed was presented, in addition to display the behavior of the simulator with regarding different values of engine efficiency due to low relative power or high altitude.

4.2 Results of Simulated Static Navigator

The aim of implementing simulation software part of the auto-navigation is to be used as a subjective testing tool. And each software part is installed on an individual computer. The two computers of both navigator and simulator were connected by a network cable to exchange the data. Figure (4.1) shows the performance of both the navigator and simulator schematically.

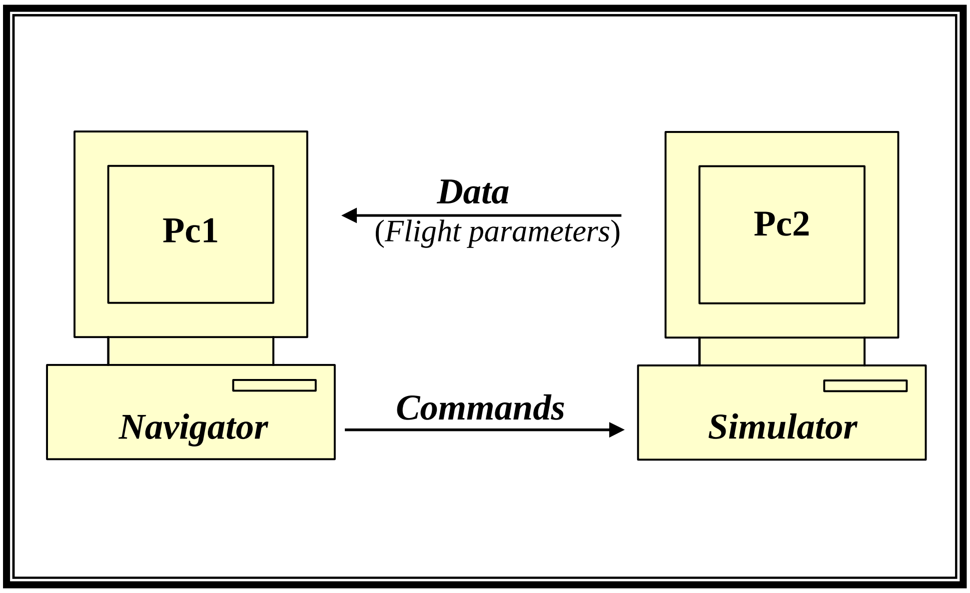


Fig (4.1) Data exchange between the navigator and simulator computers.

The data table of the flight parameters, shown in table (4.1), is fed to the auto-navigation system as navigation situations. While the data listed in table (4.2) presents the flight specifications that could be setup for calibrating and to ensure smooth behavior in the change of the flight parameters. Usually, these specifications are determined depending on the airplane capabilities.

Table (4.1) Data table of the flight trip

Way point	**Position**			**Speed (*km/h*)**	**Radius of arrival (*m*)**
	Longitude (*km*)	**Latitude (*km*)**	**Height (*km*)**		
1	3×91	8×112	2	100	5
2	9×91	4×112	8	800	5
3	5×91	5×112	4	800	5
4	7.5×91	1.5×112	7	400	5
5	5×91	2.5×112	6	600	5

Table (4.2) Specifications of the considered parameters

Parameter	**Min. value**	**Max. value**	**Max. change**
Speed (*km/h*)	50	800	
Height (*km*)	10	-2	
Head (*Degree*)	-180	180	
RPM (*rpm*)	8000	1000	70
Pitch (*Degree*)	-90	90	1.8
Roll (*Degree*)	-90	90	1.8
Yaw (*Degree*)	-15	15	0.3
Throttle	1000	8000	
Elevator (*Degree*)	-90	90	
Aileron (*Degree*)	-90	90	
Rudder (*Degree*)	-15	15	

In the simulation, the response coefficients was set one (static response), the radius of arrival was set 5 *m*, and the maximum change of each parameter was equal to one percent (1%) of the range of any parameter, this percentage is called ***Fraction*** factor since it may be useful to control the dynamic response, it was noticed that the setting of the *Fraction* factor at 1% provides least overshot and more stability for the navigation and flight parameters. Therefore, the maximum change is calculated according to the following suggested relation;

$$MaxChgF_i = Fraction \times \left[MaxF_i \text{ - } MinF_i\right] \qquad \ldots\ldots (4.1)$$

where $MaxF_i$ and $MinF_i$ are the maximum and minimum values of the i^{th} flight parameter.

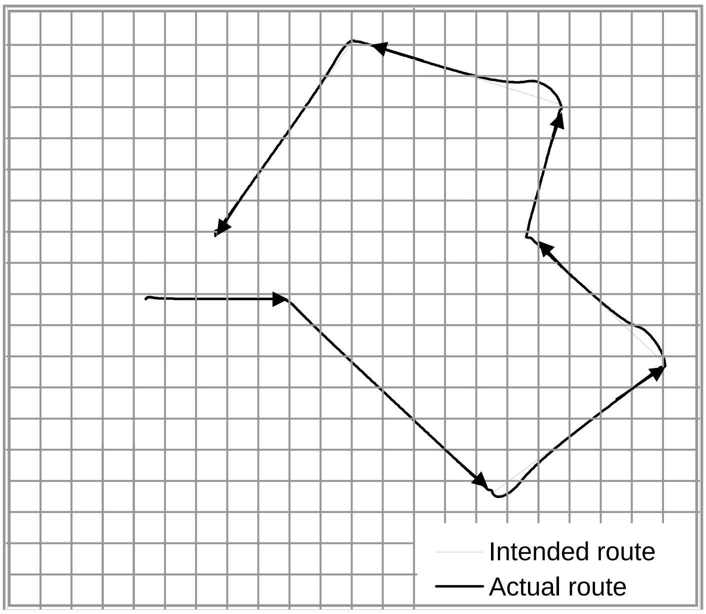
For example: the maximum change in the pitch or roll is

$$MaxF_{2,or3} = 0.01 \times \left[90 - (-90)\right] = 1.8^o$$

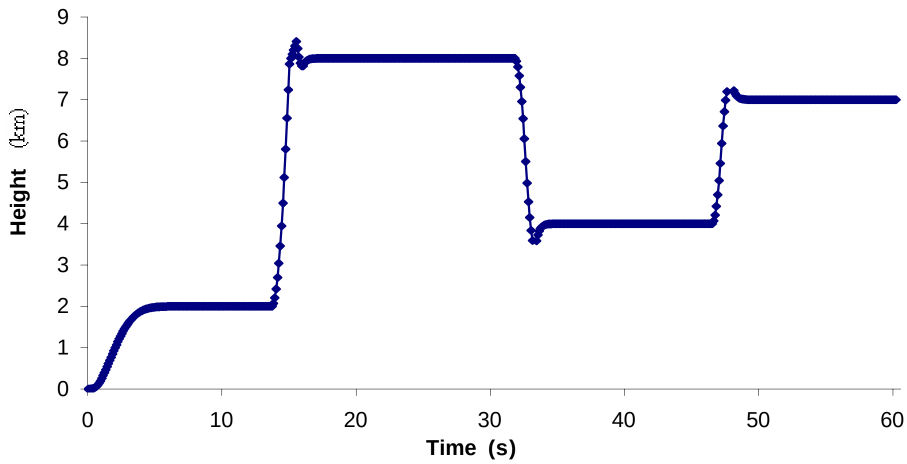
Since the correction interval is $\Delta t = 100\, ms$, the simulator must spend a time *t* to change its roll from 90^o to –90^o, this time is

$$t = \frac{90 - \left(-90^o\right)}{1.8^o} \times 100\, ms = 10\, s$$

In the consideration of UAV, this time is regarded long for changing the aileron status (for example), because of the maximum change (1.8^o) at each correction interval (100 *ms*) make the simulator influence slow, but this time may be proper for changing the pitch. The difference between pitch and roll fitting with *t* comes from the difference between the ranges of their corresponding navigation parameters (i.e., height and head) that used to determine the intended values of the pitch and roll. As a result, this time is useful for the purpose of analysis and study. Figure (4.2-a) shows the actual and intended route of the simulator, while Figure (4.2-b) shows the actual and the intended height behavior when the navigator directed the flight mission toward the five way points list in table (4.1). Figure (4.3) displays the interface of the navigator that presents the current flight situations synthesized by simulator. The captions of the interface contents refer to their functionality.



a. Route (*Lon-Lat*).



b. Height (*Hgh*).

Fig (4.2) The route and height behavior of the simulator.

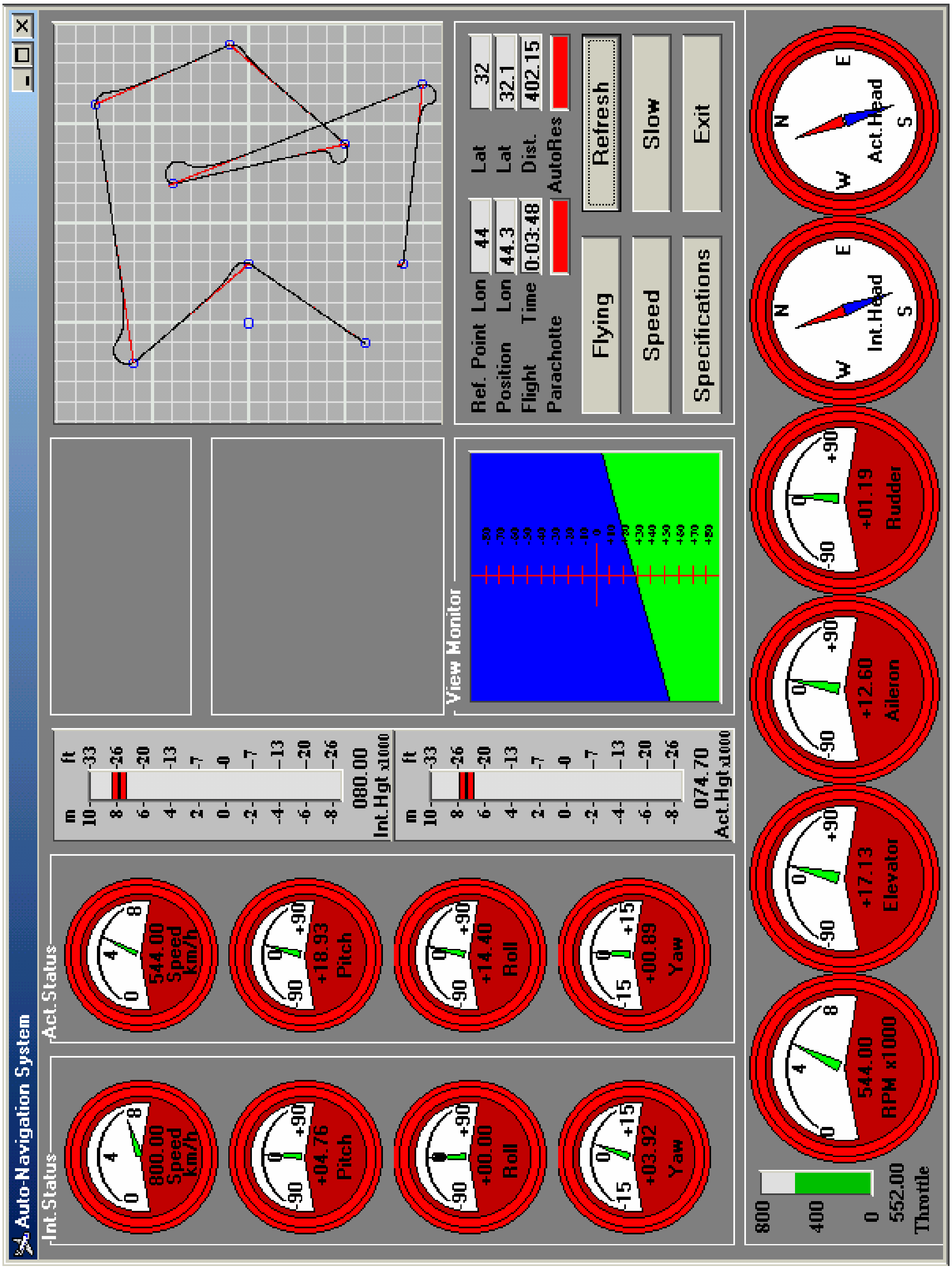


Fig (4.3) Navigator interface that picture the airplane situations during the flight

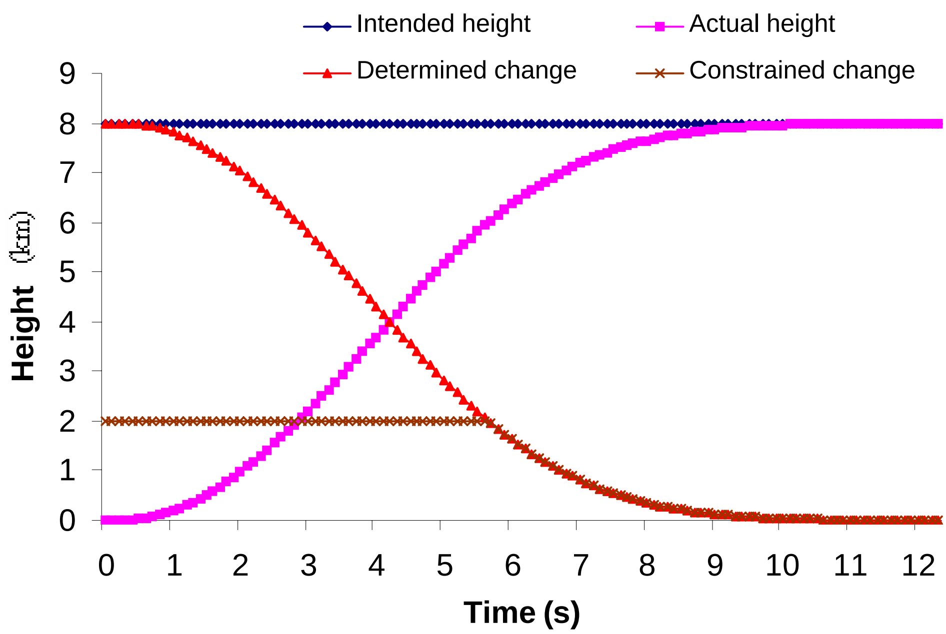
Experimentally, the behavior of the flight parameters against intended changes of the navigation parameters (such as pitch and height) was analyzed by adopting the route segment toward the second way point in table (4.1). After switching to the second way point (i.e., this happen after passing through the first way point), according to the flight plane the value of the intended height will be 8 *km* and the actual height became 2 *km* (2 *km* was the intended height of last route segment). Thus, the deflection in the height is

$$\Delta N_2 = IntN_2 - ActN_2 = 8 - 2 = 6\, km$$

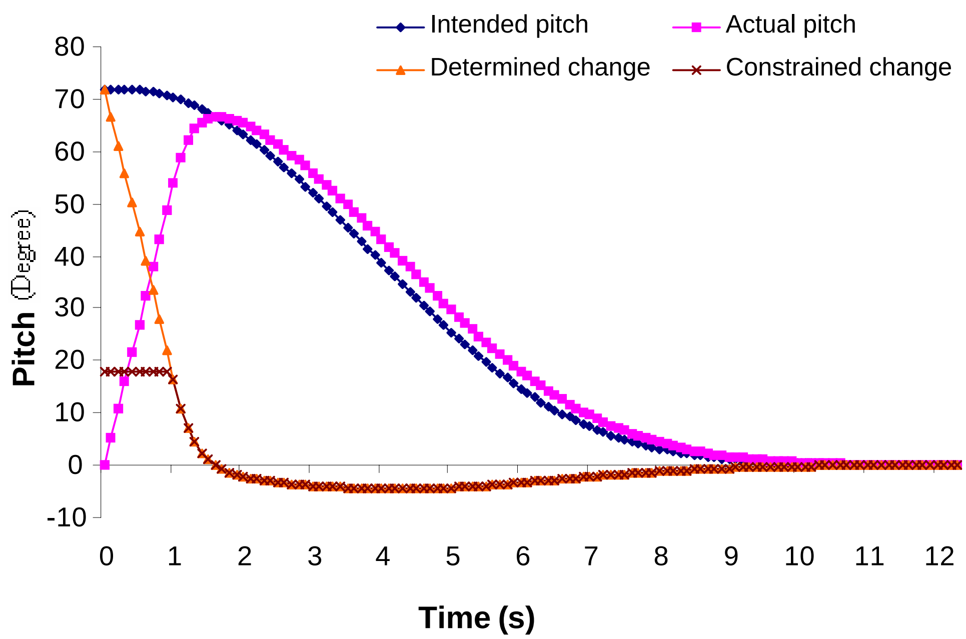
By the way, the determined pitch was allowed to reach large positive value (maximum, $IntPitch=90^o$) according to equation (3.9), the positive sign came from the positive desired change of the height (i.e., the simulator was climbing). While the actual pitch was zero ($ActPitch=0^o$) since the flight (at the way point switching instance) was still level. Therefore, the deflection of the pitch was large positive value given as

$$\Delta F_2 = IntF_2 - ActF_2 = 90^o - 0^o = 90^o$$

The deflection in the pitch is restricted by the maximum amount of change in the pitch ($MaxChgF_2 = 1.8^o$), therefore, the current value of the actual change of the pitch was $\Delta Pitch = 1.8^o$. This small change in the pitch requires the elevator changes smoothly by an amount proportional with the amount of the attended deflection in the pitch according to equation (3.3). Thus, the elevator that had a previous value equal to zero should change its status to 1.8^o, this led to an increase in the height and the simulator showed climbing behavior. As a result, the status of the elevator was changing with an associative increase in the height. Figure (4.4) illustrates the smooth behaviors of both the height and the pitch along the given route segment.



a. Height behavior.



b. Pitch behavior.

Fig (4.4) Simulated height and pitch at climbing status.

At intervals follow the start of climbing phase; the intended height remained constant along all the correction intervals, while the actual height changed periodically toward the desired value due to the continuous correction. The deflection of the height had reached the maximum value at first correction intervals (after the switching instance) and it was decreased gradually.

Since the intended pitch is proportional to the deflection of the height, its same behavior of change was similar to that for the deflection of the height (i.e., it was gradually decreased). The actual pitch that was in normal status (i.e., its angle equal to zero) was continually raised toward the intended. The deflection of the pitch earned large value at the first correction intervals and then it decreased smoothly due to the decrease in the deflection of the height. The positive pitch deflection had caused the elevator to move continuously toward the positive direction. These changes in elevator status caused gradual increase in the height that computed by the simulator. The predicted deflection of the pitch was straight line as well as the determined deflection of the pitch was greater than the amount of the maximum allowed change in the pitch, for such cases it was clipped to be equal to the maximum allowable value. This led to make the simulator actual pitch increased by a rate proportional with the maximum change of the pitch. The deflection of the determined pitch remained straight until it reaches the case where the deflection of the pitch became less than the maximum allowable change in the pitch, and then it was gradually decayed at each correction interval.

As a result, the deflection in the height was decreased, the intended pitch was decreased, and the actual pitch was increased. At the same time, both the deflections in the height and pitch have been decreased sequentially due to continual correction in the height. Also, the determined actual height (by the simulator) was increased following the intended height, till reaching the ***transient state***.

The transient state occurs when the actual pitch value reach a maximum allowable value, in such cases the deflection of the pitch became zero. This state remain until the intended pitch become equal or close to the actual pitch, in this state the actual pitch shows an intendment to reverse its behavior.

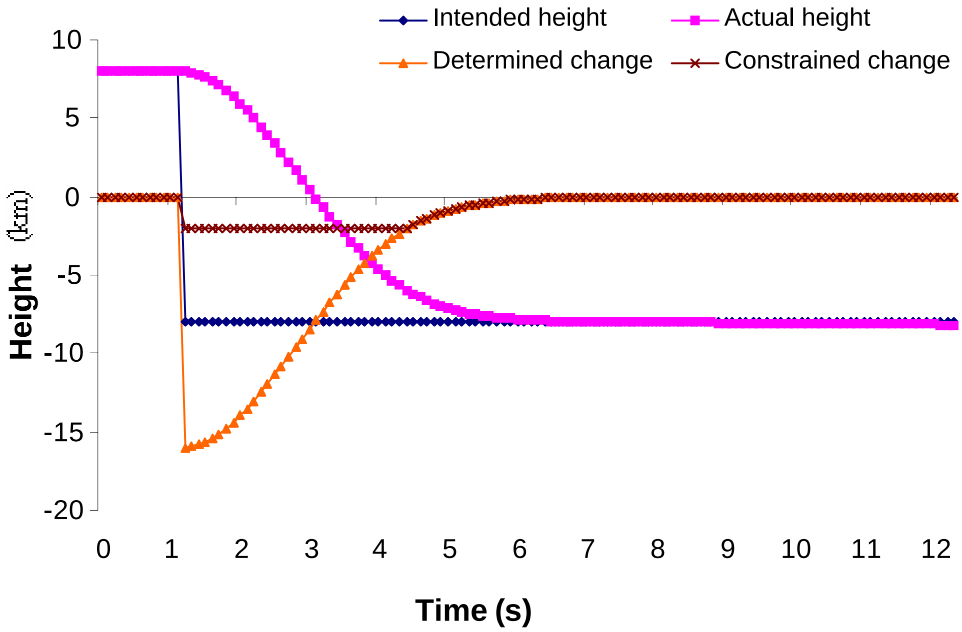
After the transient point, the correction of the height still continuing, the intended height remains fixed as it was, the actual height remains continually increase, and the deflection of the height is decreased till reaching the value zero. On the other hand, the decrease in the pitch deflection continue and became a negative value because of retreat the intended pitch from the actual pitch, i.e. the intended pitch became smaller than the actual one. The intervals characterized by the negative change of the pitch led the elevator decrease back to its normal state ($Elevator = 0$), such state occur when both the actual pitch and the intended pitch become zero. Also, one can notice that the change in the predicted deflection of the pitch was little and it takes a long time to become zero.

Figures (4.4-4.11) illustrate the corresponding behaviors of the flight parameters due to the deflections happened in the navigation parameters for different cases. The navigation parameters are also shown in these figures. An interesting behavior was noticed, it is the uniform increase in the speed behavior, which is due to the uniform acceleration imposed by the simulator (airplane) during the simulation (flight).

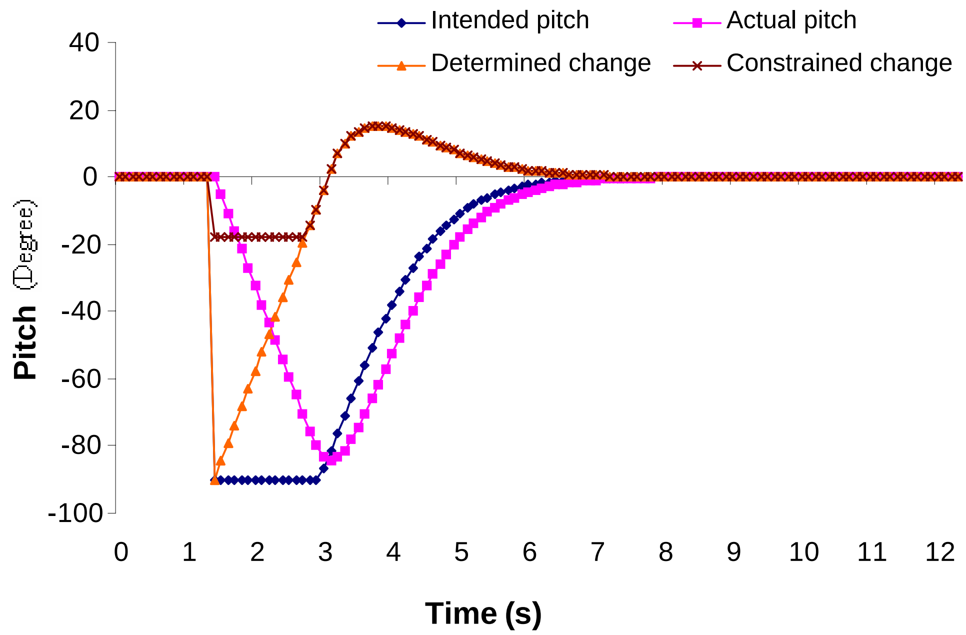
In the following paragraphs the behaviors of both the navigation and flight parameters (except the speed and *RPM*) are described for the two cases; before and after the reversion state.

Before the transient state, the intended navigation parameter remained as it is along the time of the corrections, the actual navigation parameter was gradually corrected toward the intended value. The changes in the navigation parameters behaved in inverse manner with the behavior of the actual navigation parameter. While the behaviors of the intended flight parameters were similar to the behavior of the change of the corresponding navigation parameters. The actual flight parameters follow the intended, and the behavior of deflection of the flight parameters (before the transient state) is linear as long as the determined deflection of the flight parameters is greater than the amount of the maximum changes in the flight parameters. The deflection in the flight parameters decreases, and begins to decay slowly until it reaches zero at the transient state.

After the transient point, the change in flight parameter continues in same way. The actual flight parameter changes its course of change from increasing/decreasing or decreasing/increasing (i.e., reverse its behavior). The deflection of the flight parameter became negative, because the actual flight parameter will, always, follow the intended flight parameter. As the actual flight parameter decreases/increases due to the negative/positive deflection, the intended flight parameter decreases/increases due to continual correction in the navigation parameter, and they are never meet unless reaching zero when both reach the normal state.

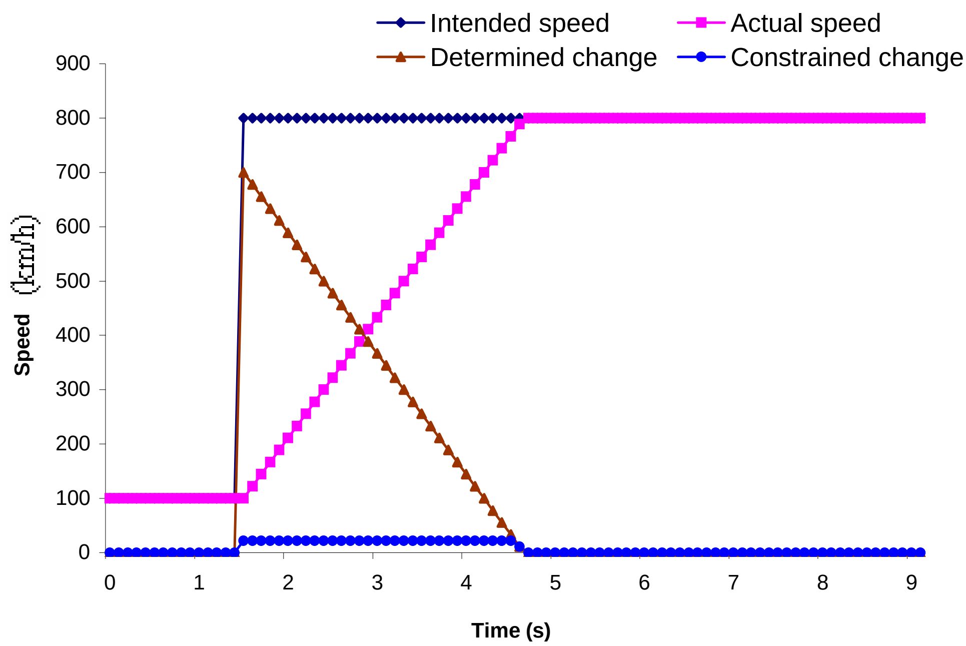


a. Height behavior.

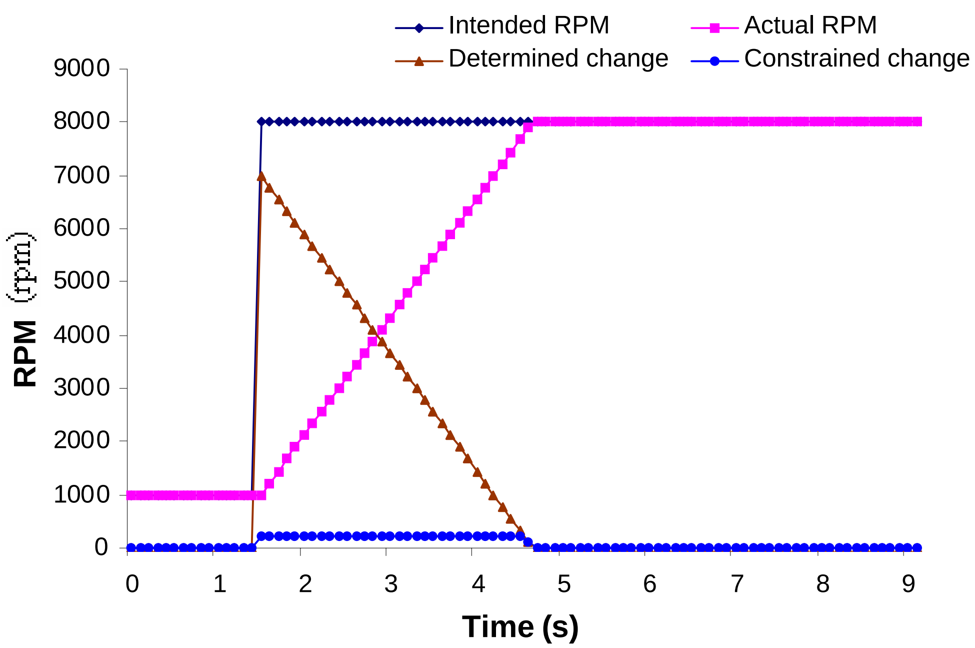


b. Pitch behavior.

Fig (4.5) Simulated height and pitch at descending status.

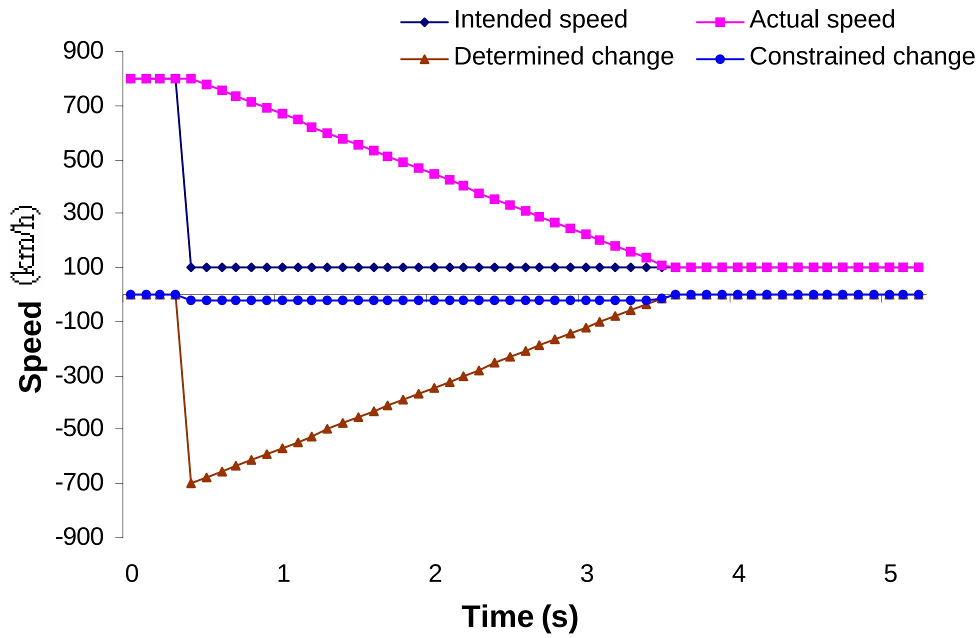


a. Speed behavior.

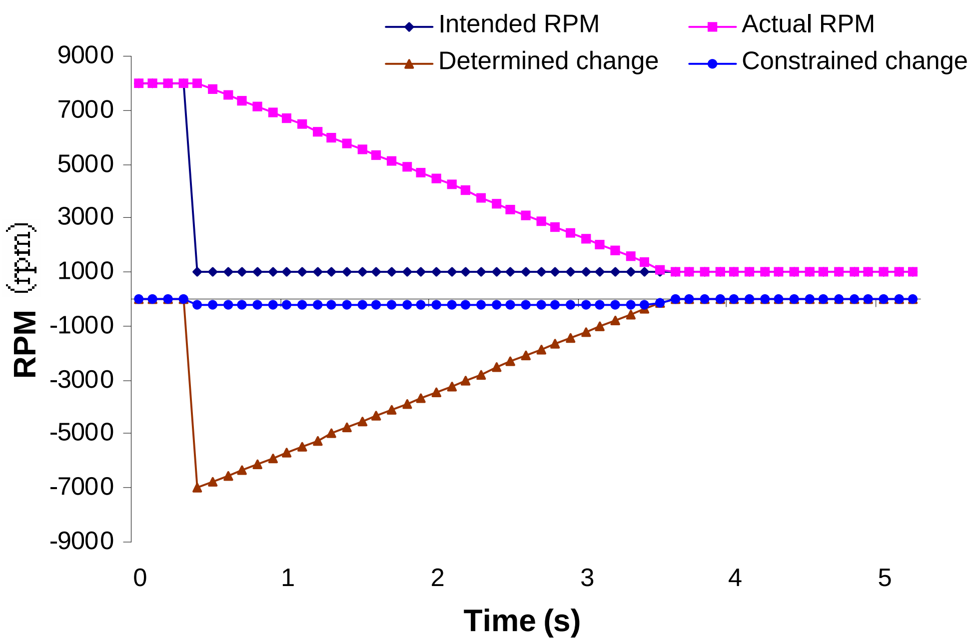


b. RPM behavior.

Fig (4.6) Simulated speed and RPM at speedup status.

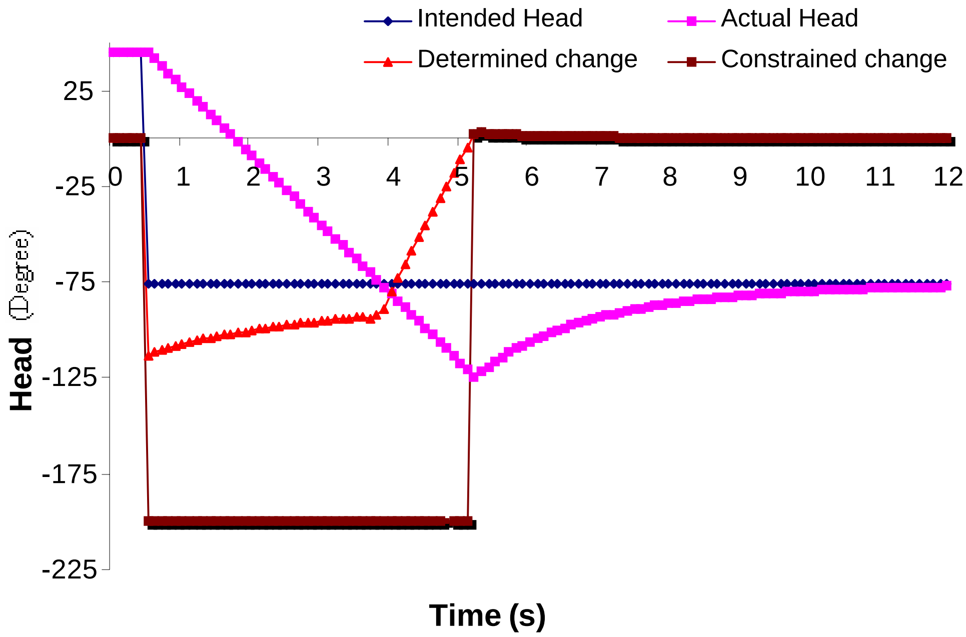


a. Speed behavior.

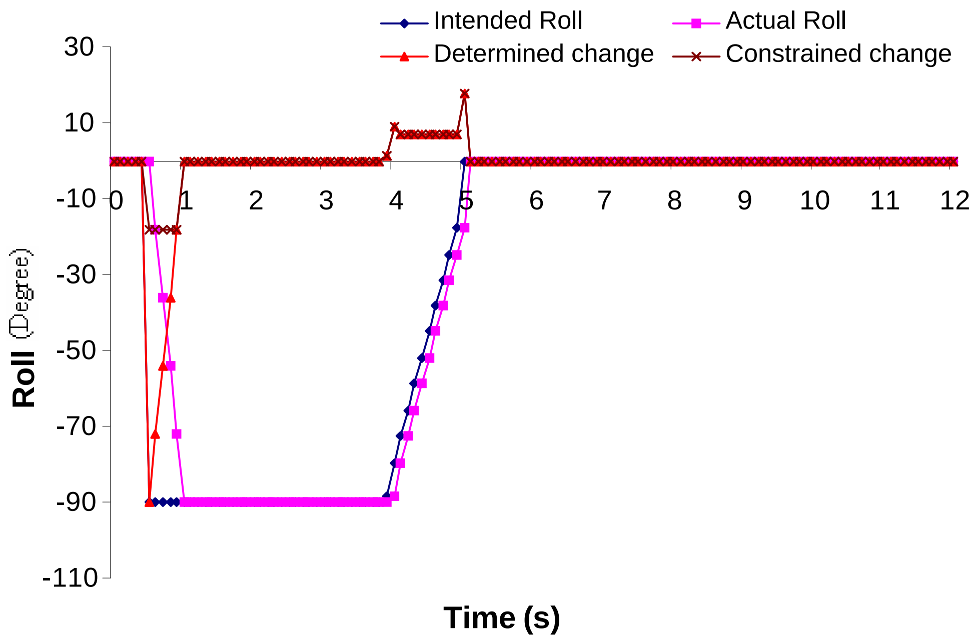


b. RPM behavior.

Fig (4.7) Simulated speed and RPM at slowdown status.

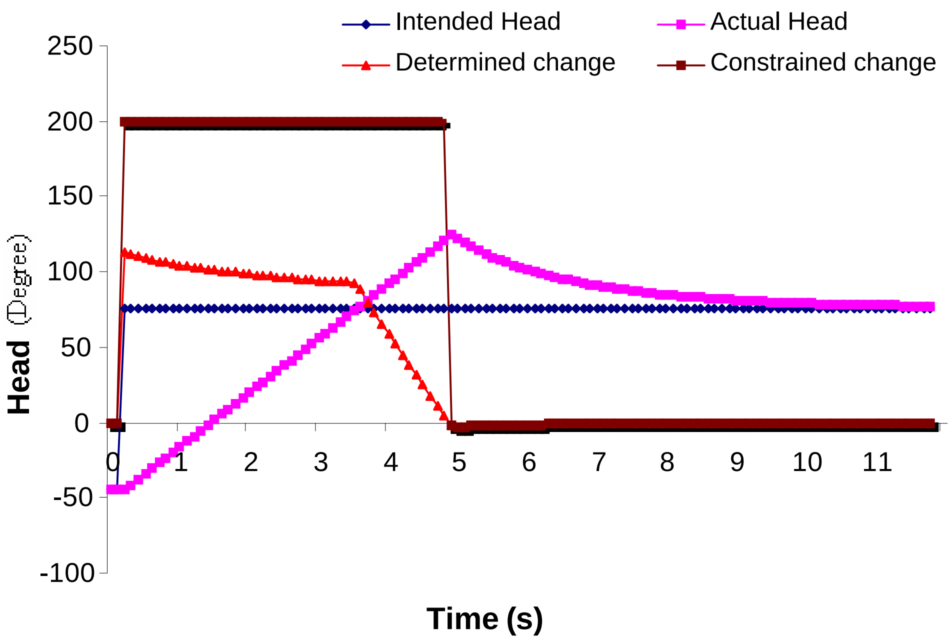


a. Head behavior.

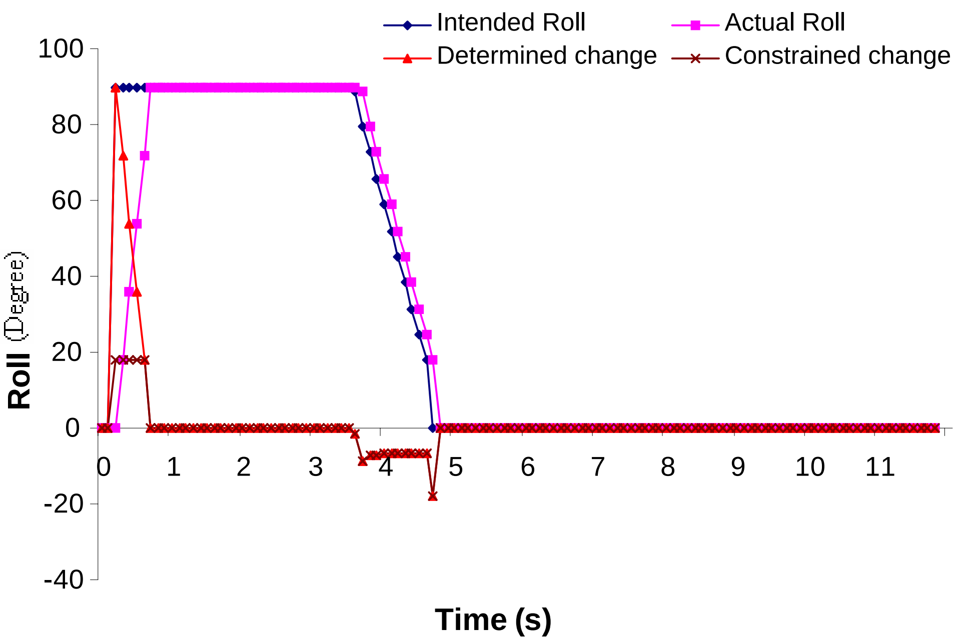


b. Roll behavior.

Fig (4.8) Simulated head and roll at left turn status.

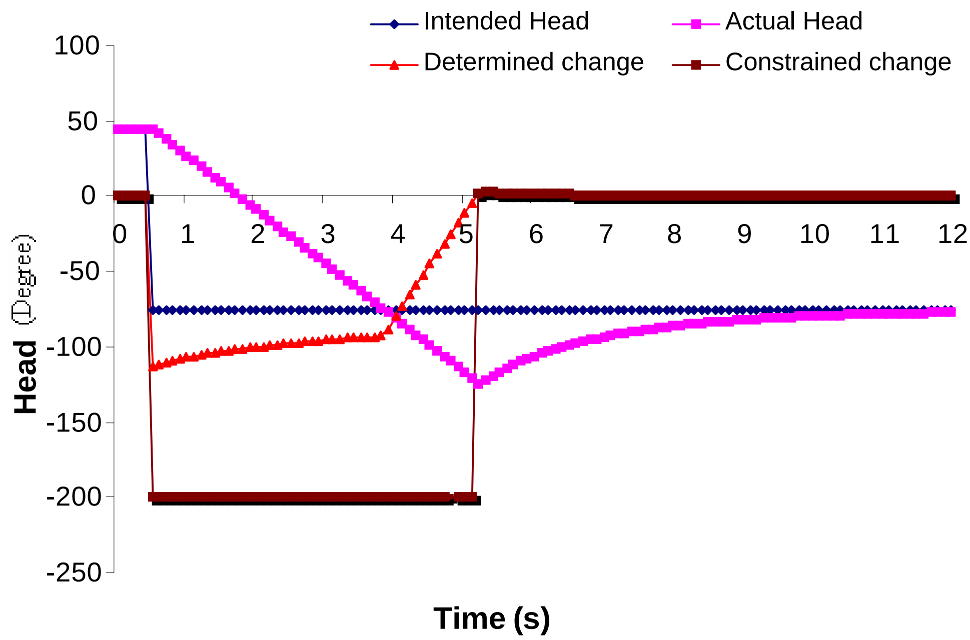


a. Head behavior.

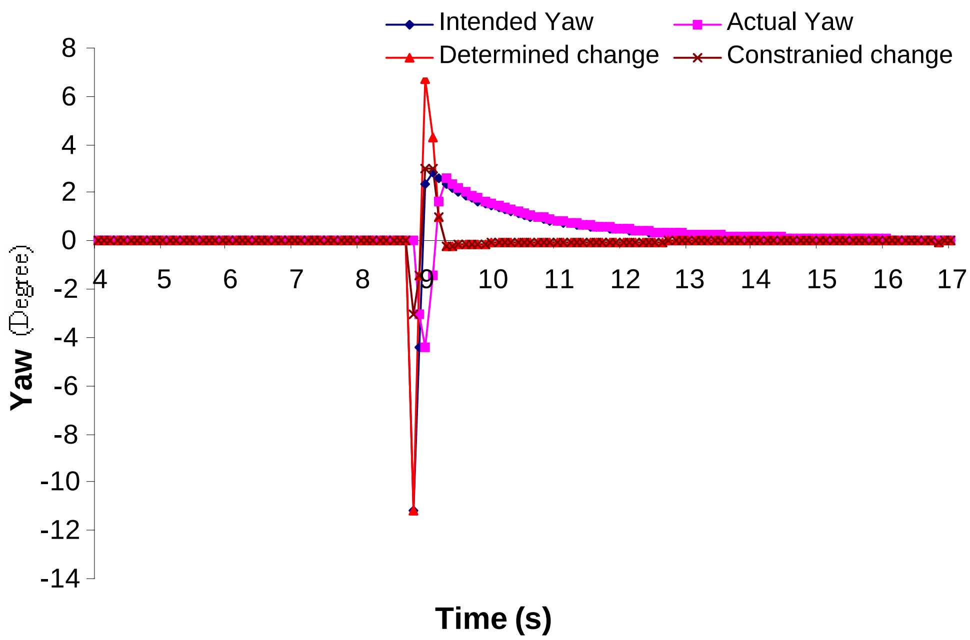


b. Roll behavior.

Fig (4.9) Simulated head and roll at right turn status.

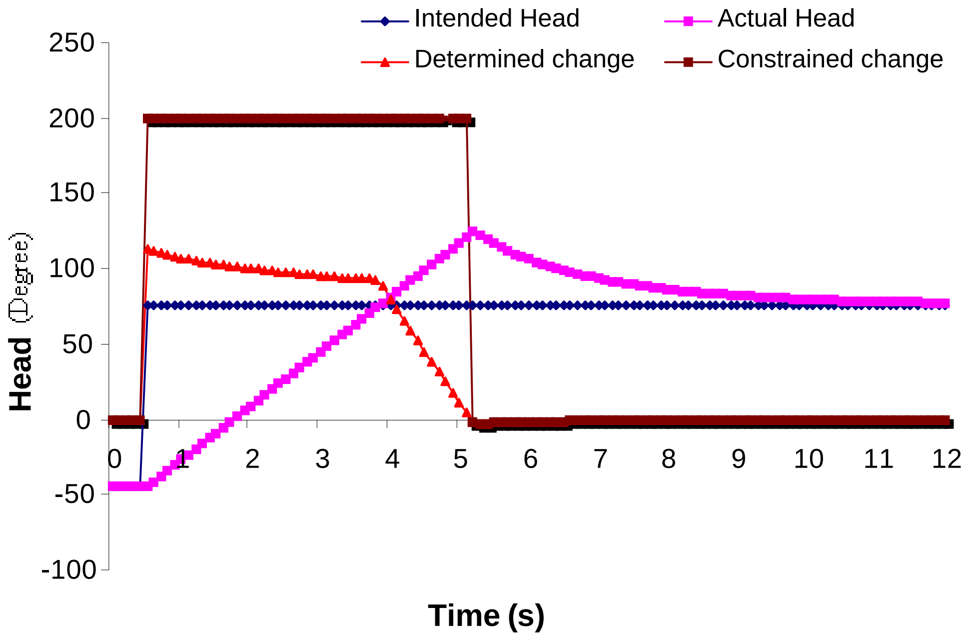


a. Head behavior.

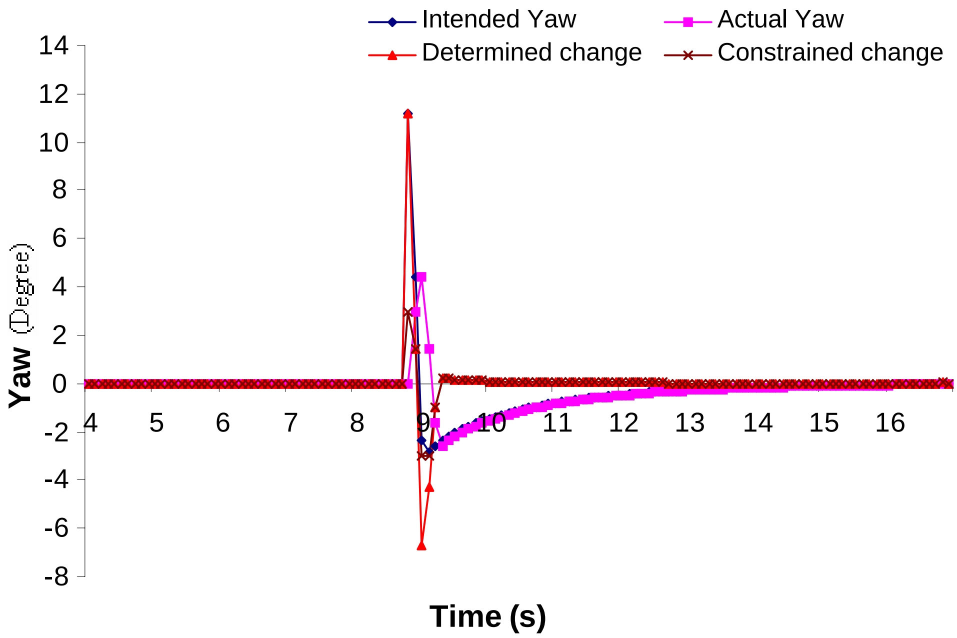


b. Yaw behavior.

Fig (4.10) Simulated head and yaw at left turn status.



a. Head behavior.



b. Yaw behavior.

Fig (4.11) Simulated head and yaw at right turn status.

Figures (4.4-4.11) describe the behaviors of the flight parameters and its effect on the navigation parameters when the response coefficients taken equal to one. The behaviors become faster when the corresponding response coefficients are taken greater than one, and they are slower when the response coefficient is taken less than one.

In order to analyze the effect of the response coefficient on the simulated flight behavior, the effect of changing response coefficients which corresponds to pitch angle and consequently to height is taken, where different response coefficients were used. Figure (4.12) shows the behavior of the simulated height when the response coefficient (k_{2i}) was 0.5, 1, or 2, which corresponds to the behavior shown in Figure (4.4). While Figure (4.13) presents the behavior of the simulated height when the response coefficient k_{2i} is 0.5, 1, or 2.

Large k_{2i}, shown in Figure (4.12), enable the flight (simulator) to spent less time to reach the intended height (in comparison with behavior shown in Figure 4.4), where the slope of the actual height curve was higher. Also, the intended pitch decayed quickly toward zero and the actual pitch raised by larger rate till reaching the transient state, then it began to descend quickly toward zero. To reach the transient state smaller time was needed, in other words the occurrence of the transient state was shifted toward less time in the time domain.

Generally, the diagrams in Figures (4.12) and (4.13) show shrink behaviors for both the navigation and flight parameters in comparison with those shown in Figure (4.4). The slope of change of all parameters seems sharper, and the position of the critical points (transient state and the state at which the pitch deflection reaches the maximum allowed change) are budged toward less time in the time domain.

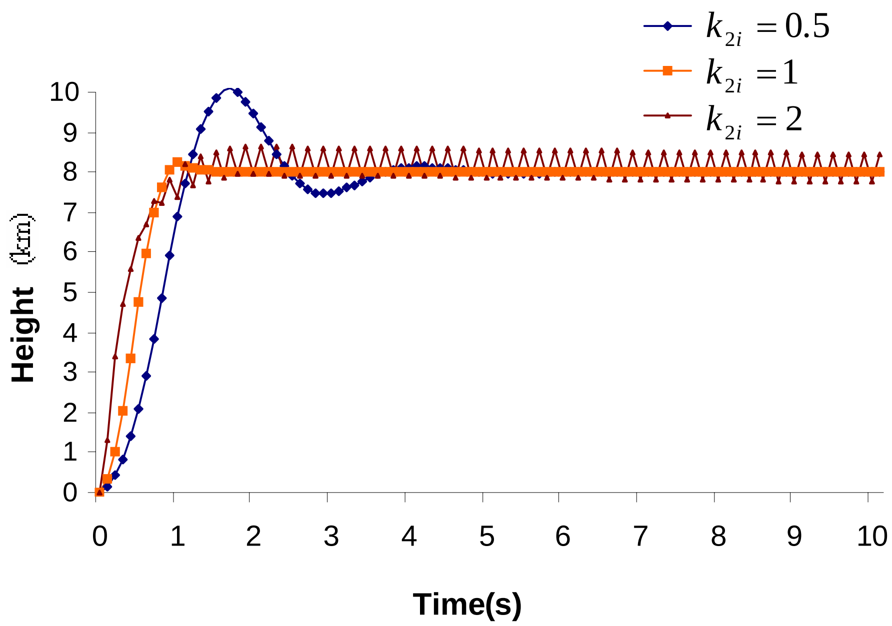


Fig (4.12) Simulated actual height behavior for different values of k_{2i}.

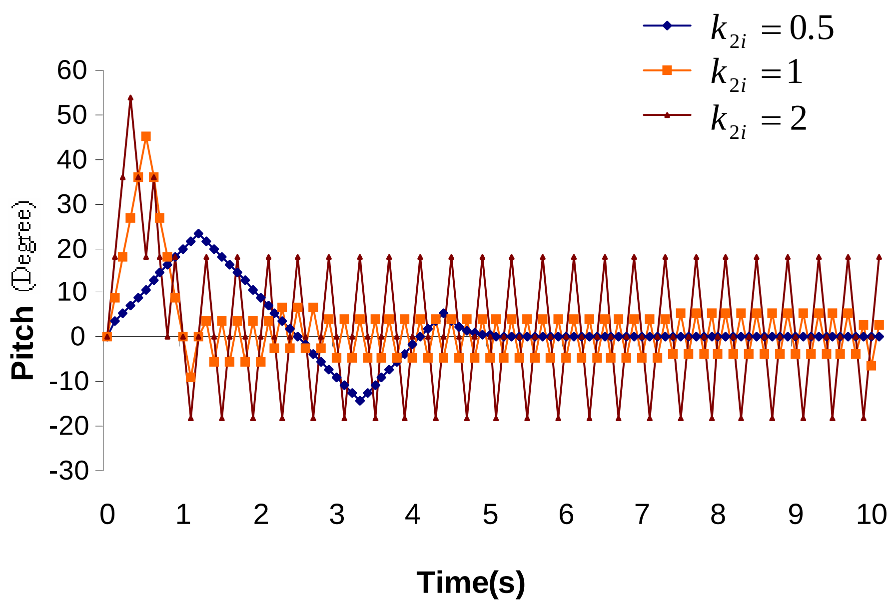


Fig (4.13) Simulated actual pitch behavior for different values of k_{2i}.

Also, in the middle case ($k_{2i} = 1$), shown in Figure (4.4), the simulator was stable and behaved smoothly, while unstability happened in the navigation and flight parameters for the case $k_{2i} = 2$, this instability occur when the actual height become close to the intended height, also both the intended and actual pitch become zero. This is due to the fact that the deflection of the pitch is doubled, and due to the airplane moment inertia the pitch continue in its incrementation which make the height exceeded the intended value, when this overshot happen then the correcting pitch angle jump to the negative position around the zero position. Whereas the deflection of the pitch was oscillated between the negative and positive around zero, it was never stabled at zero.

When k_{2i} was set 0.5, the behavior of both the navigation and flight parameters was slowed down and the time spent to reach the attended as shown in Figure (4.13). The slope of the curve was decreased and the positions of critical points have been shifted toward longer time on the time domain. The behavior of the simulated flight was slow, and consequently longer time was needed to achieve the intended states. In this case an overshot may also happen, which causes damping oscillation in the navigation and flight parameters.

The analysis indicates the response coefficient has the ability of speeding up or slowing the behavior of the simulated navigation parameters if it is processed well. Therefore, the response coefficient can be used to shorten or dilate the time required to reach the desired situation as needed under the condition of stability. In order to satisfy the stability condition, it is suitable to make $k_{2i} = 1$ at small deflections, especially those less than the maximum change.

Furthermore, the speed affects the actual navigation parameters, where the rate of change of the actual height depends directly upon the actual speed according to equation (3.3). Figure (4.14) shows the behavior of the actual height at three different speeds (200, 500, and 800 *km*/*h*) the values of all other parameters were fixed. It is obvious that the actual speed has a large effect on the behavior (rate of change) of the actual height, it behaves as a scaling factor affect the actual height, where the high speed increase the rate of change of the actual height at the middle intervals, while at the start and end intervals of climbing phase it is kept with small changes. But, some effect may associate the high speeds; it is the overshot. The appearance of an overshot proves the occurrence of large deflection in the head when a high speed airplane needs to be turned. This large deflection is due to the continuity of motion.

Also, the maximum change in the pitch parameter ($MaxChgF_2$) will affect the behavior of the actual height. The considered values of *Fraction* (i.e., 0.1, 0.01, and 0.001) lead to different values of maximum change in pitch (i.e., 18^o, 1.8^o, and 0.18^o respectively) as shown in Figure (4.15). It shown the behavior of *Fraction*=0.01 is the best. When Fraction=0.1 the height showed sharp climbing, overshot, and less time to reach the intended height in comparison with behavior of height at *Fraction*=0.01, this ascribe to the great response handle due to the large change in pitch. Whereas, when the Fraction=0.001 the associated height showed very smooth behavior, damping oscillation, and longer time to reach the intended height, this ascribe to late correction due to the least rate of change in pitch. Therefore, it is very important for the control to set the value of *Fraction* at proper value (i.e., rate of change in flight parameters is proper).

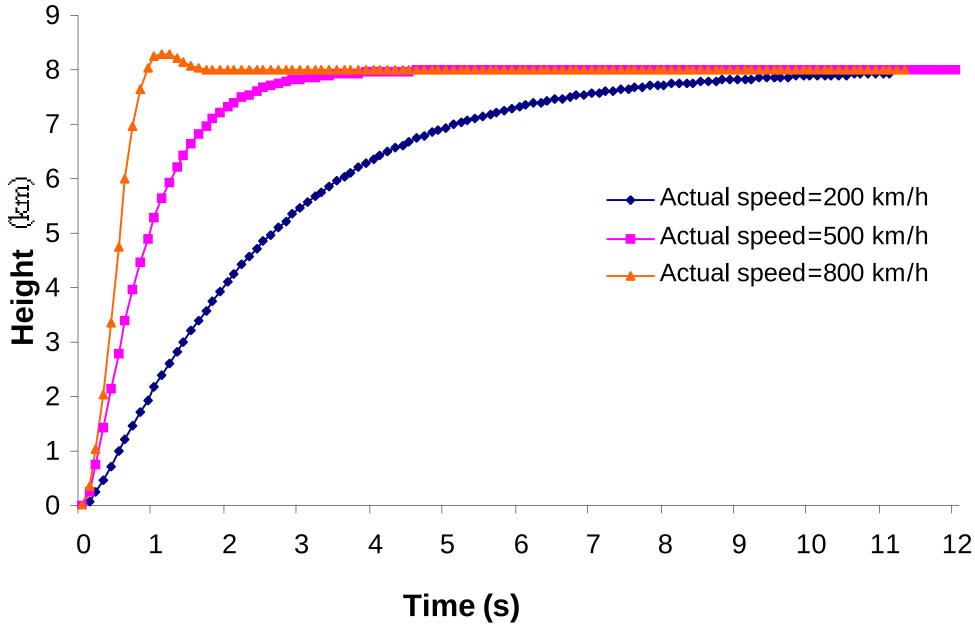


Fig (4.14) The effect of speed on the behavior of simulated height.

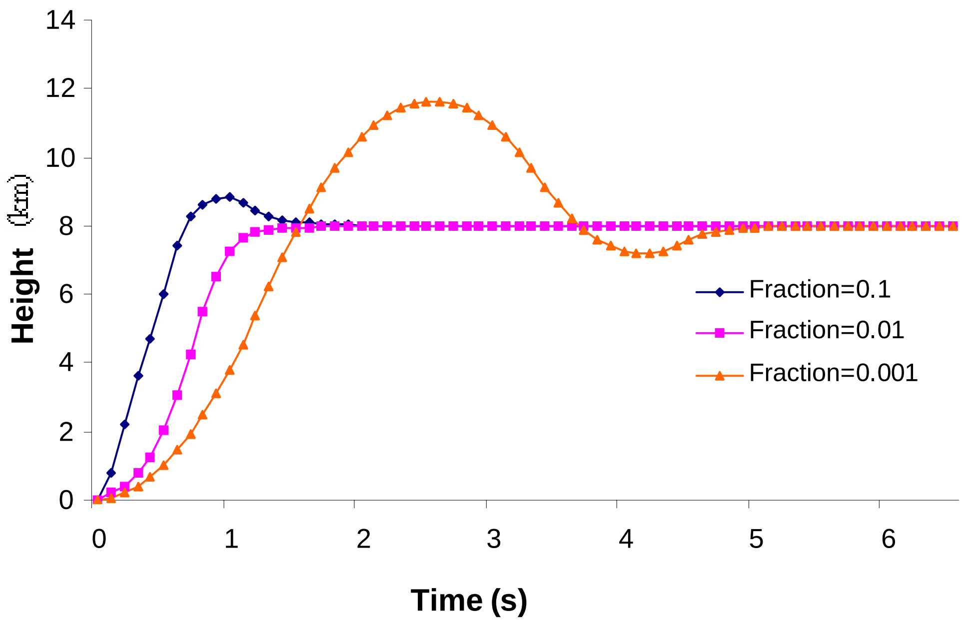


Fig (4.15) The effect of $MaxChgF_2$ on the behavior of simulated height.

4.3 Results of Simulated Dynamic Navigator

Analytically, it is found that the range of variation in the value of k_{2i} has an inverse dependency on the amount of maximum change in a specific flight parameter, i.e. it is depend on the factor *Fraction* as follows.

$$Range_k \propto \frac{1}{Fraction}$$

$$Range_k = c \times \frac{1}{Fraction} \qquad \ldots (4.2)$$

Practically, it is useful to set the proportional factor *c* equal to 0.1.

$$\therefore\ Range_k = 0.1 \times \frac{1}{0.01} = 10$$

Since the deflection of the flight parameters may be negative or positive value, the response coefficient must be distributed along the positive range to preserve the direction of the correction. Also, in order to overcome the unstability problem, the minimum value of k_{2i} was taken equal to 1 at small deflections (i.e., deflections close to zero). Thus, the allowed range of k_{2i} is

$$1 \le k_{2i} \le 10$$

under the condition; ***if*** $\Delta F_i \le MaxChgF_i$ ***then*** $k_{2i} = 1$

where, $k_{2i} = 1$ specify the lowest response and $k_{2i} = 10$ specify the highest response.

The variation of k_{2i} within the range from1 to 10 leads to speed up the simulator behaviors by 10 times at high deflections without leading to instability and overshot status. In the following, the results of both the manual and automatic methods are presented in addition some detailed description and analyses are given about the parameters.

1. Manual Method Results

In the static responded navigator (i.e., when k_{2i} is constant) the overshot occurs due to the small deflections of the flight parameters, such overshot cannot be exceeded unless using dynamic responded navigator (i.e., making k_{2i} is variable) such that it can fit the deflections to smoothing follow the intended.

Figure (4.16) displays the established graphical interface of the navigator, it employs a manual method to control the dynamic response. The four moveable scroll bars under names speed, pitch, roll, and yaw in the navigator interface refer to the influence of the simulator. They govern the four values of the response coefficient that belong to the four flight parameters. The value given by each scroll bar is limited between 1 (as minimum) and 10 (as maximum). The behavior of the simulator is more controllable when the user treats the navigator with more care and accuracy, which is earned by more practicing.

It is found that high speed is useful in the case of fallback the actual values from the intended ones. The reach to high speed makes the deflection in the navigation parameters higher, which require to raise the influence of the simulator. Furthermore, the rigging of the simulator influence near the desired situations was sensitive and may harm the stability or lead to occur great overshot. Therefore, the user should be advised to decrease the values of scroll bars (i.e., k_{2i}) with rates proportional with the occurred corrections, such that the lowest value is taken when the actual value of the flight parameters is near the desired (intended) situations.

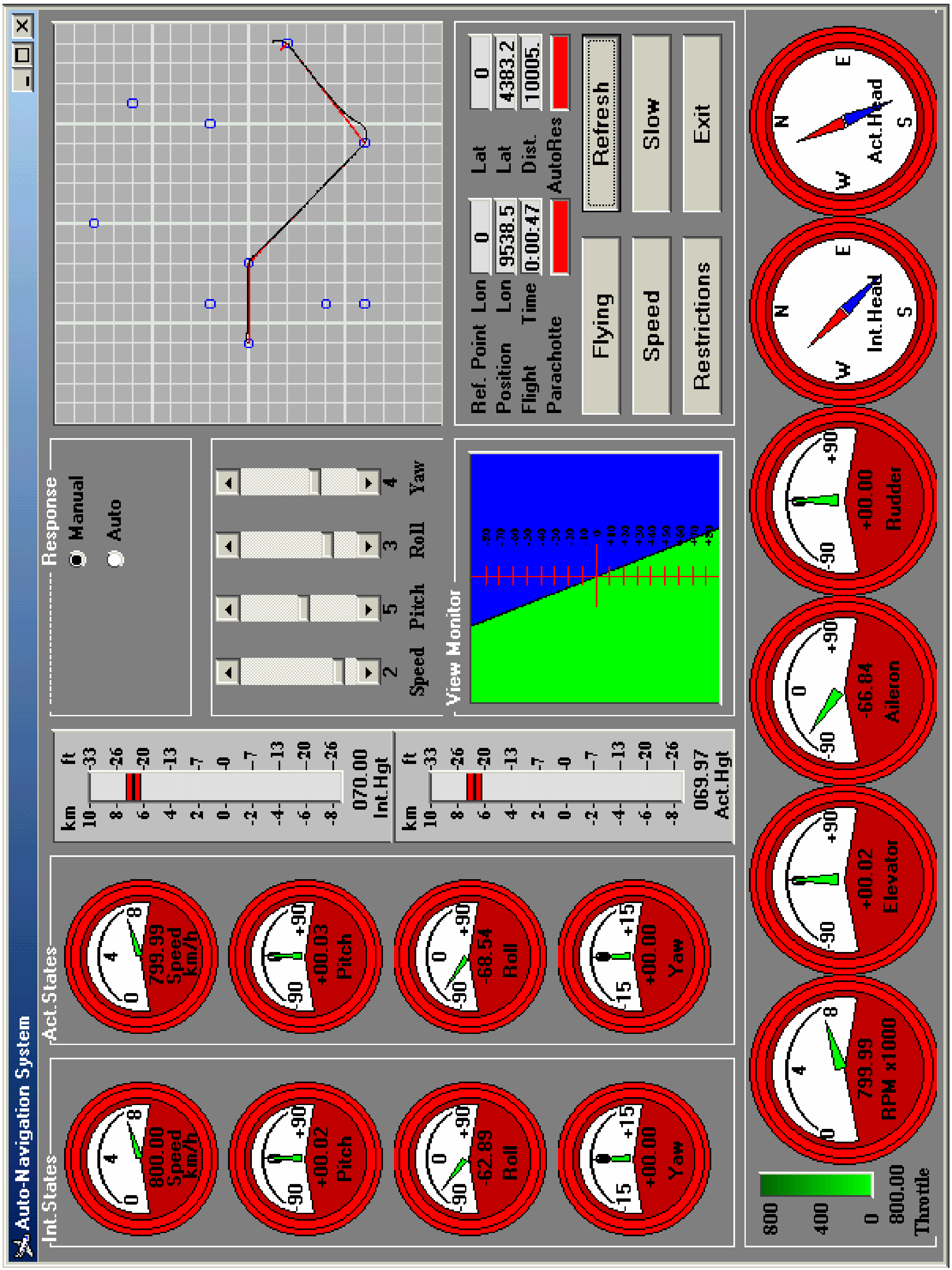


Fig (4.16) Navigator interface that imploys the manual method to determine the influence of the navigator

2. Automatic Method Results

Experimentally, it is found that both of the two automatic suggested methods are efficient in computing the dynamic response of the simulator. The scroll bars used in the manual method were replaced by a graphical board to display the instantaneous behaviors of the four planned and actual corrections. This board enables the viewer to observe how the actual parameters go toward the planned values.

Figure (4.17) shows the interface of the navigator that use the deflection based method for determining the response coefficients, while Figures (4.18-4.21) shows the behaviors of the flight parameters corresponding to their resulting navigation parameters. Figure (4.22) shows the interface of the navigator that uses the time averaging method and Figures (4.23-4.26) show the behaviors of navigation and flight parameters when the navigator employs the time averaging method. The behaviors of the two method's parameters present how the simulator goes to the desired situation with less time and more stability comparing with the case of static response ($k_{2i} = 1$) presented previously in Figures (4.4-4.11)

It should be mentioned that the deflection based method gives k_{2i} is originally calibrated between the range of 1-10, while the time averaging method gives an instantaneous values of k_{2i} whose lowest value is unity (when $\Delta N_i \leq MaxChgN_i$) and its highest value sometimes exceed 10 (at higher deflections). Thus, the following condition should be taken into account to clip the values of k_{2i} that greater than 10.

If k_{2i}>10 ***then*** k_{2i}=10

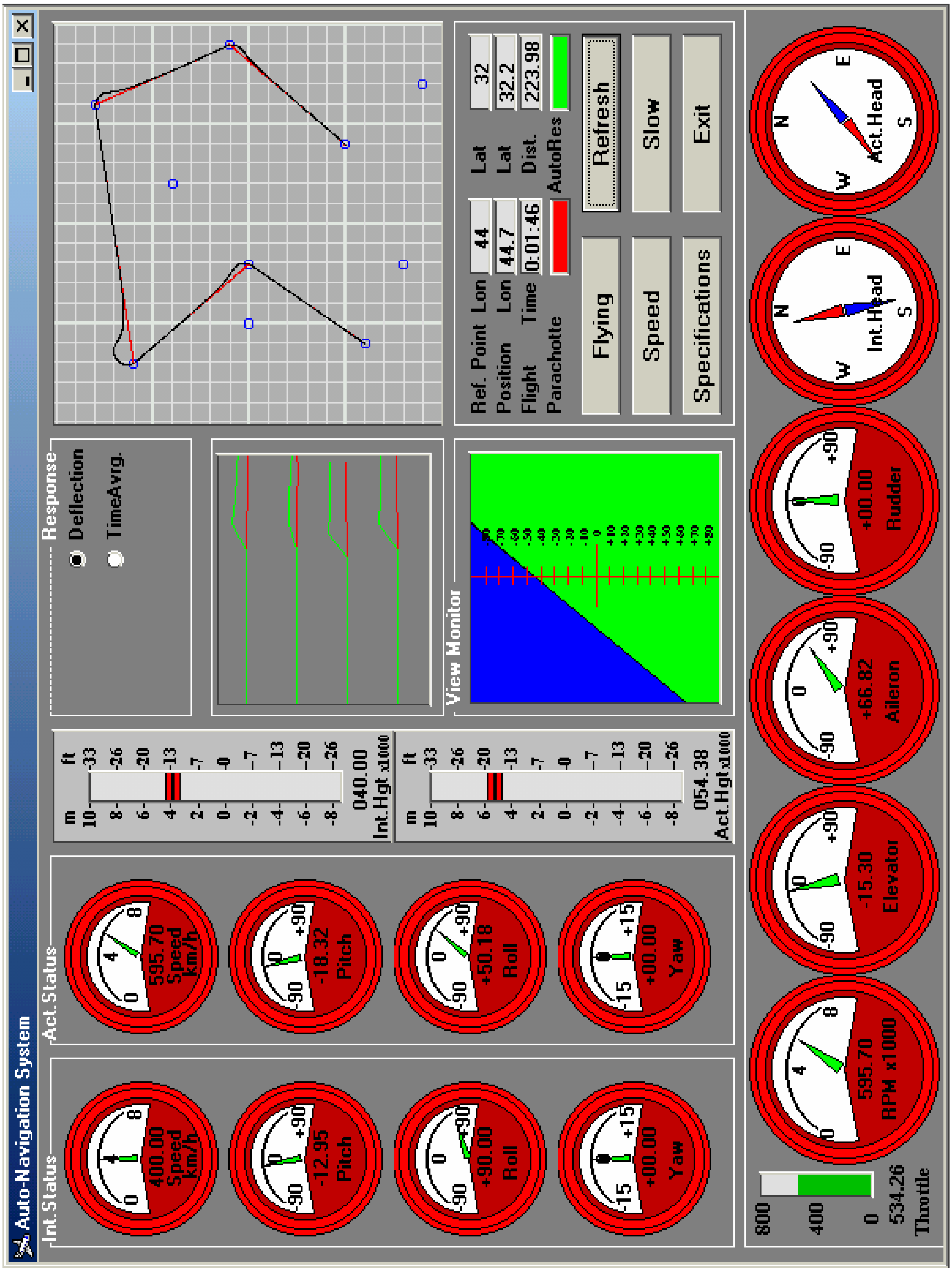
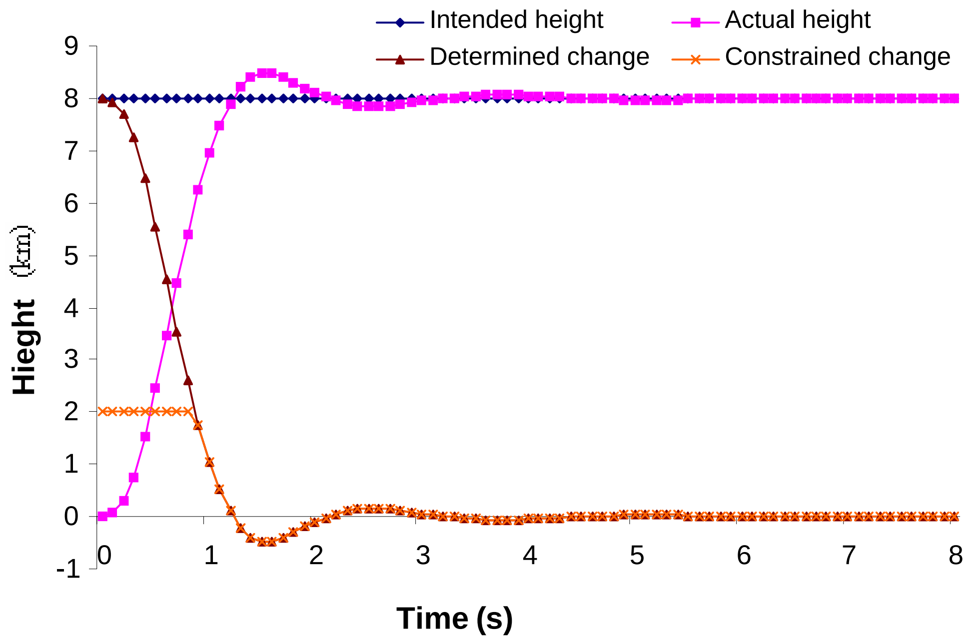
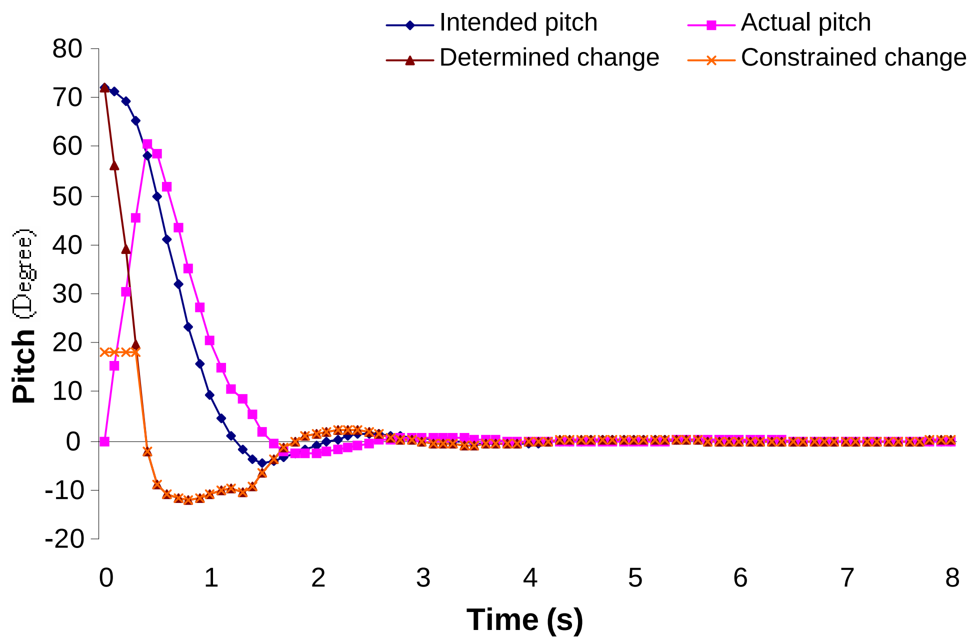


Fig (4.17) Navigator interface that employs the deflection based method to determine the influence of the navigator.

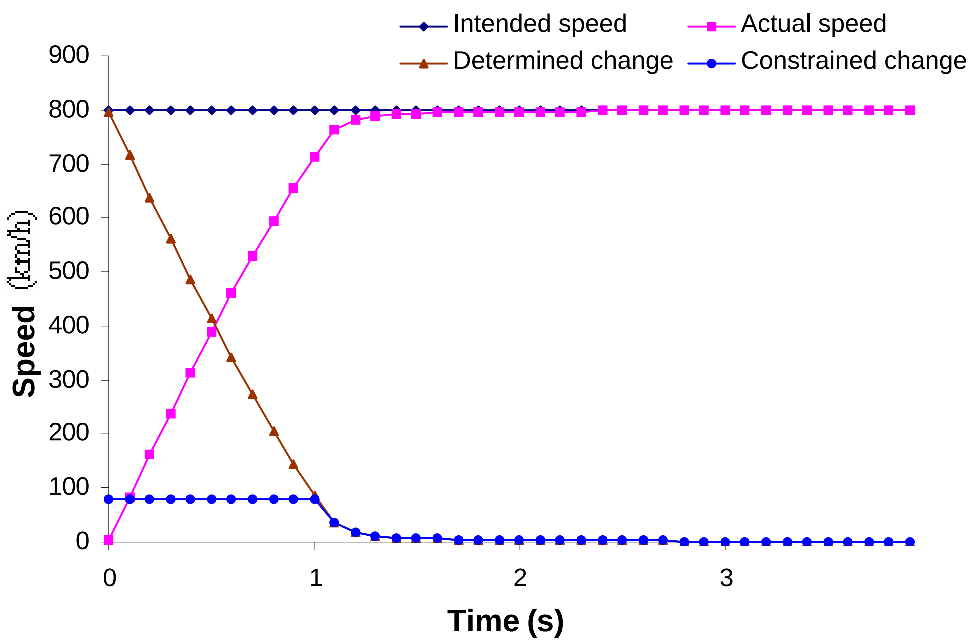


a. Height behavior.

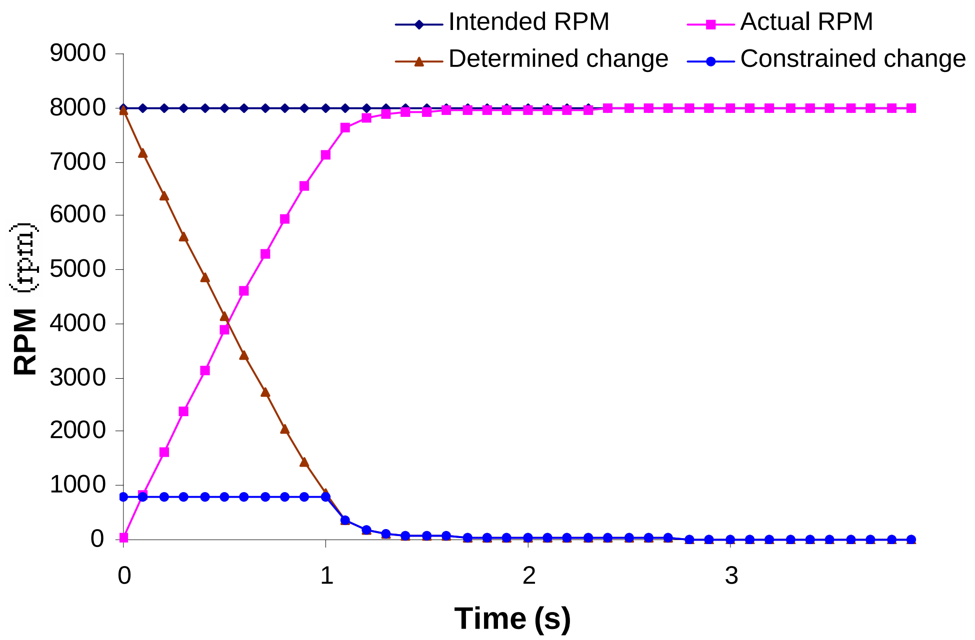


b. Pitch behavior.

Fig (4.18) Height and pitch of a simulator use the deflection based method at climbing status.

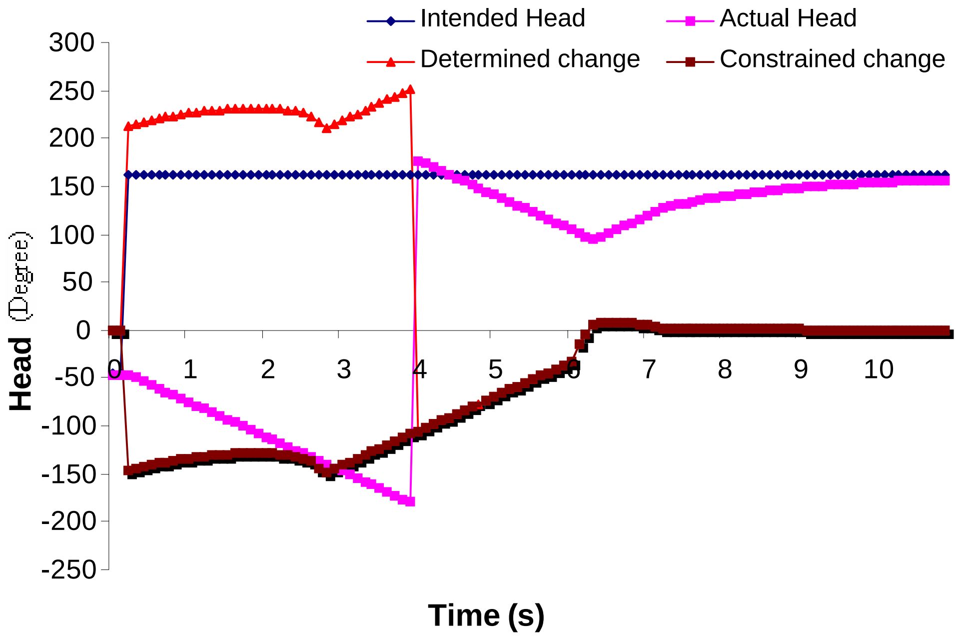


a. Speed behavior.

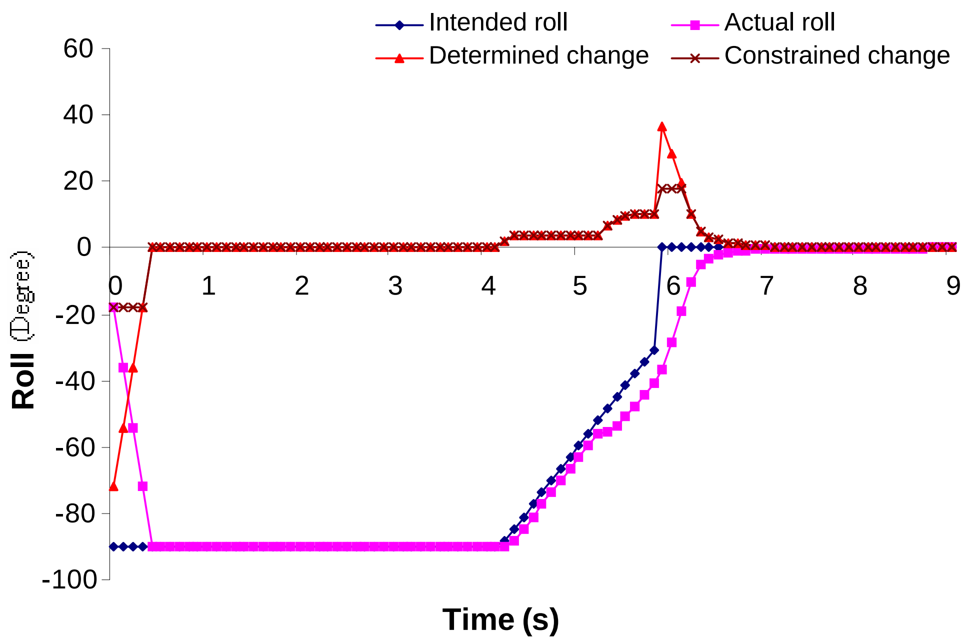


b. RPM behavior.

Fig (4.19) Speed and RPM of a simulator use the deflection based method at speedup status.

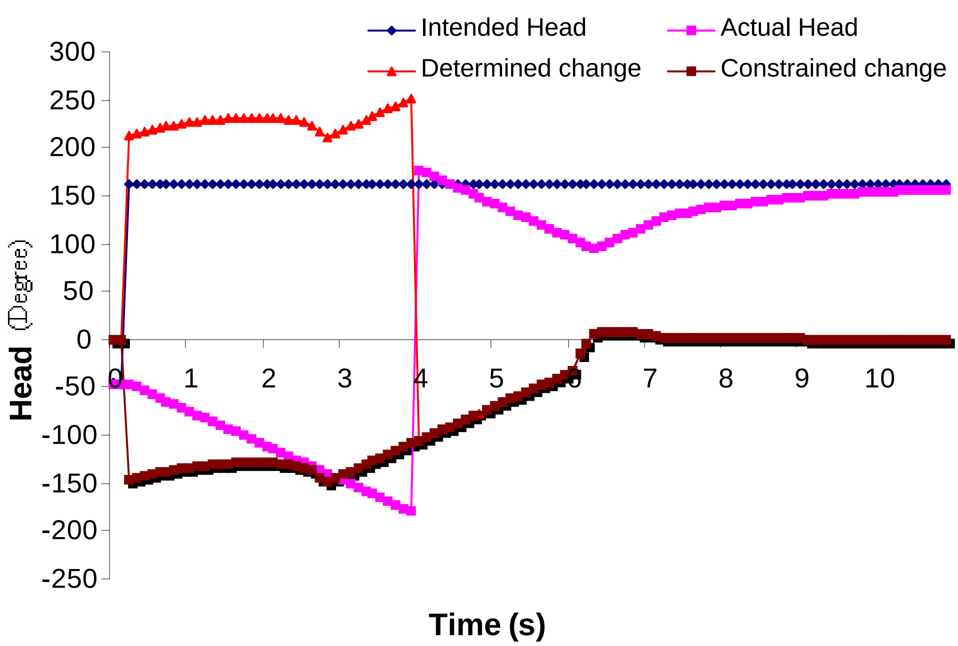


a. Head behavior.

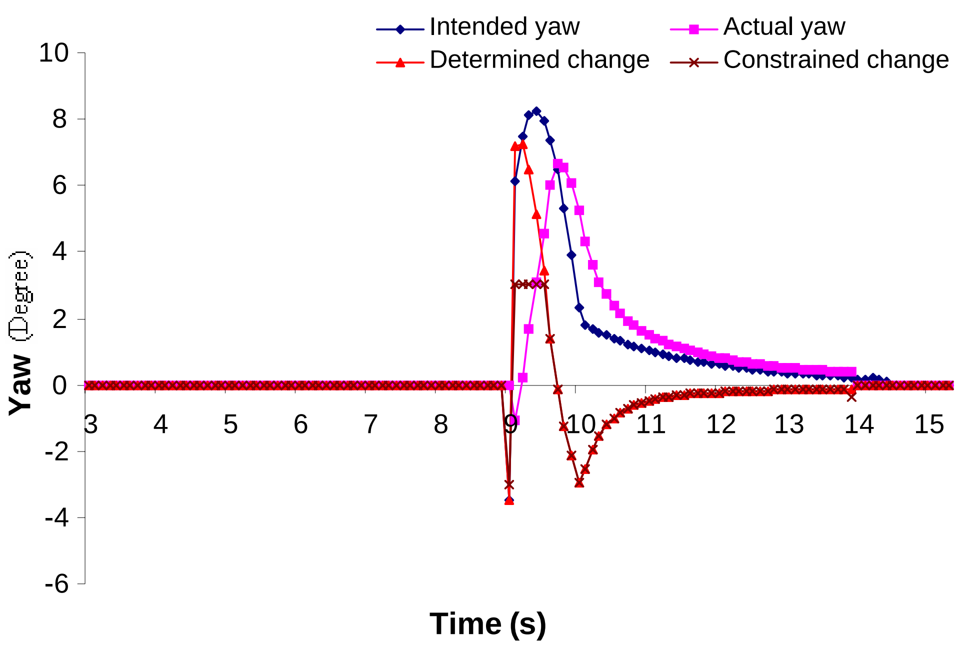


b. Roll behavior.

Fig (4.20) Head and roll of a simulator use the deflection based method at left turn status.



a. Head behavior.



b. Yaw behavior.

Fig (4.21) Head and yaw of a simulator use the deflection based method at left turn status.

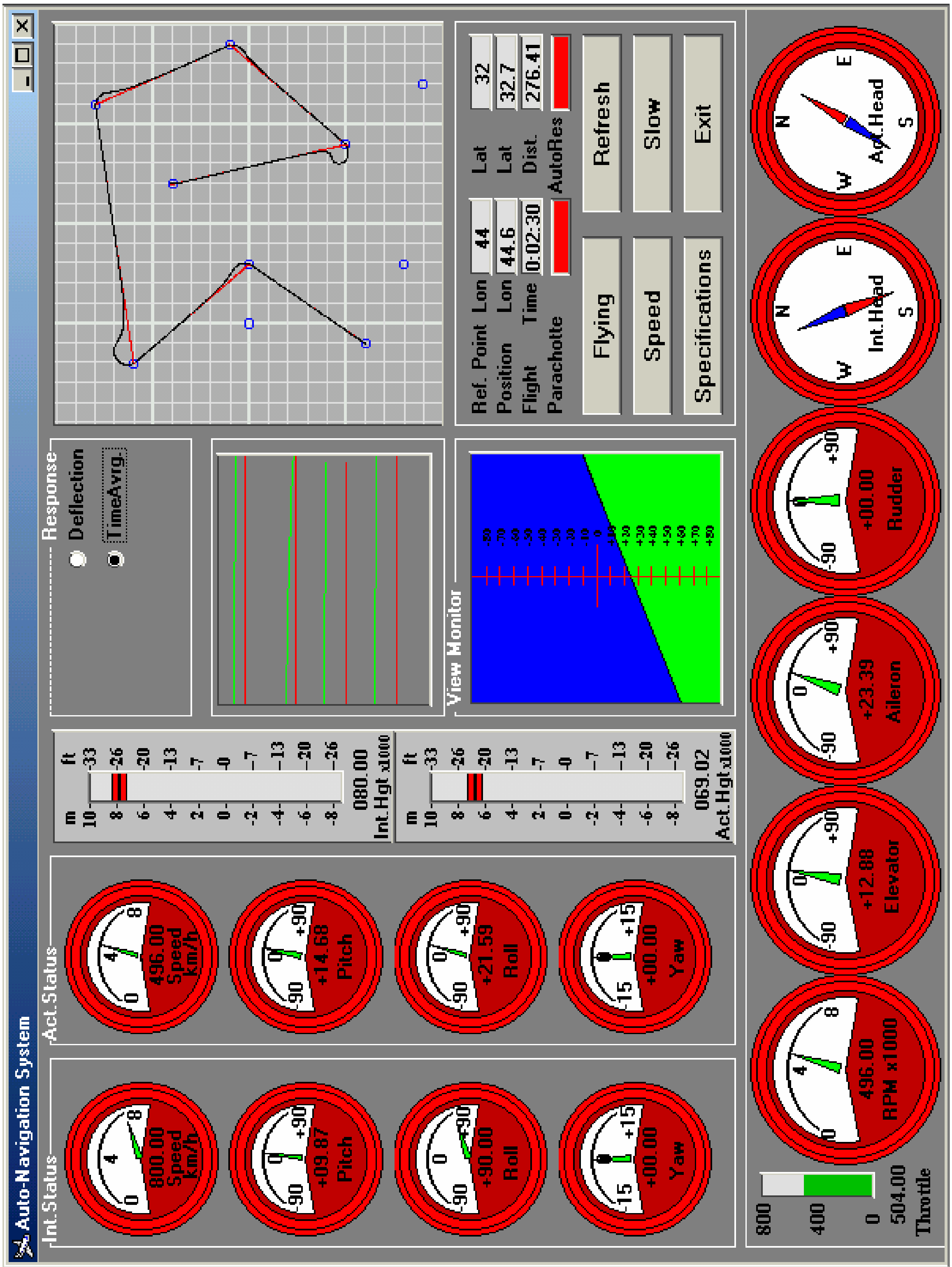
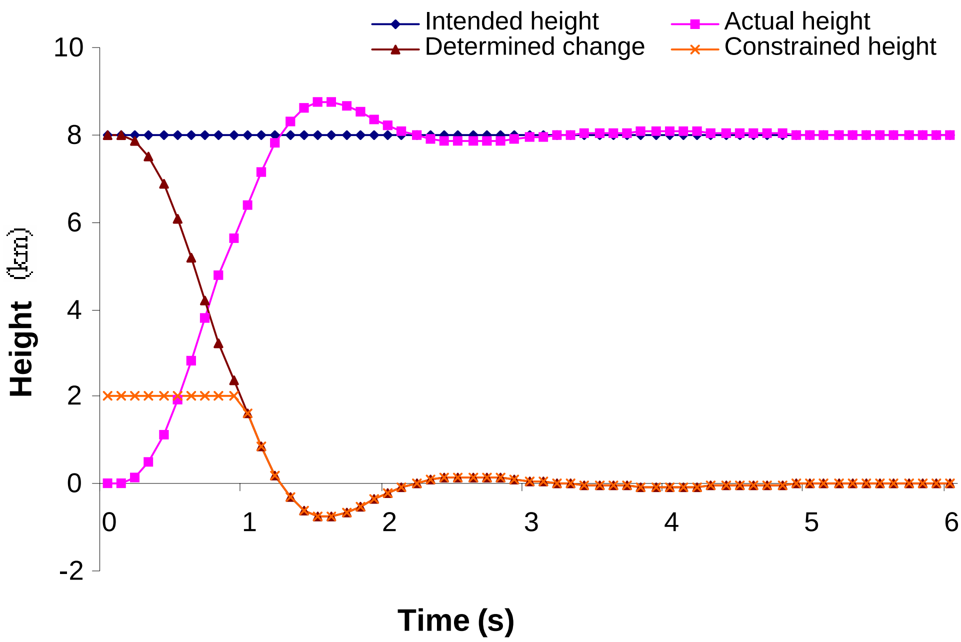
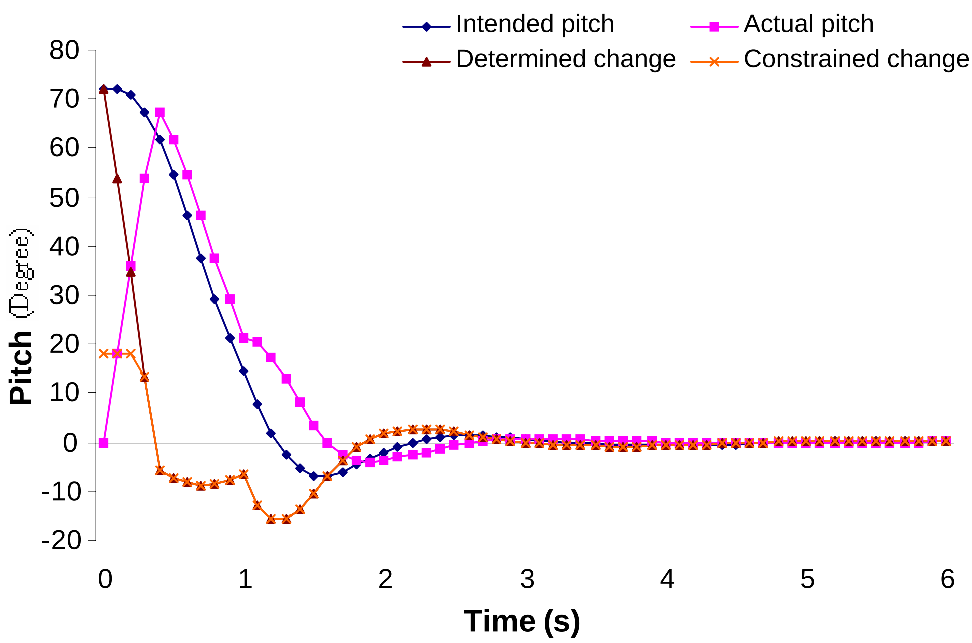


Fig (4.22) Navigator interface that imploys the time averaging method to determine the influence of the navigator.

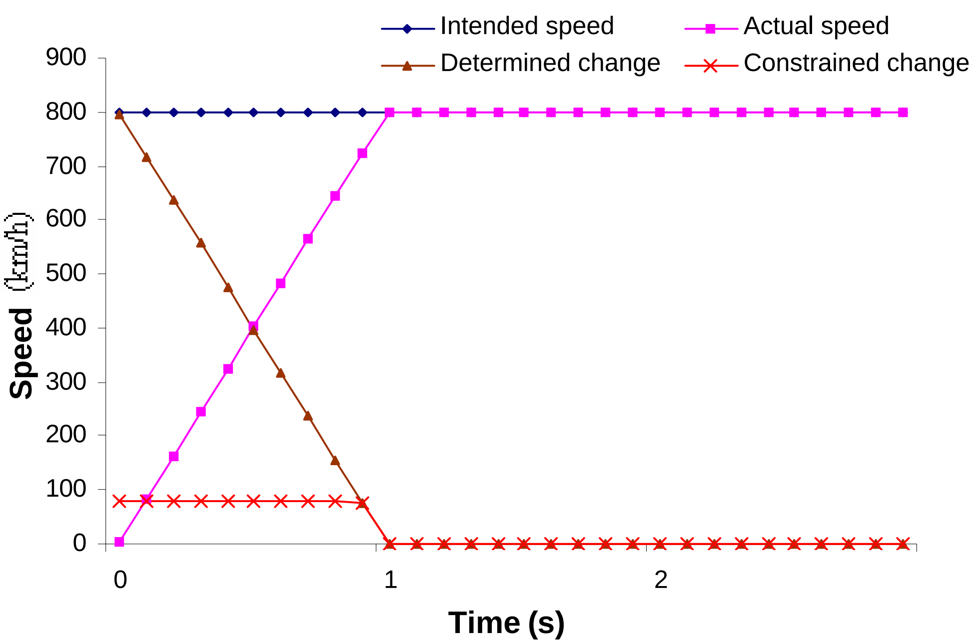


a. Height behavior.

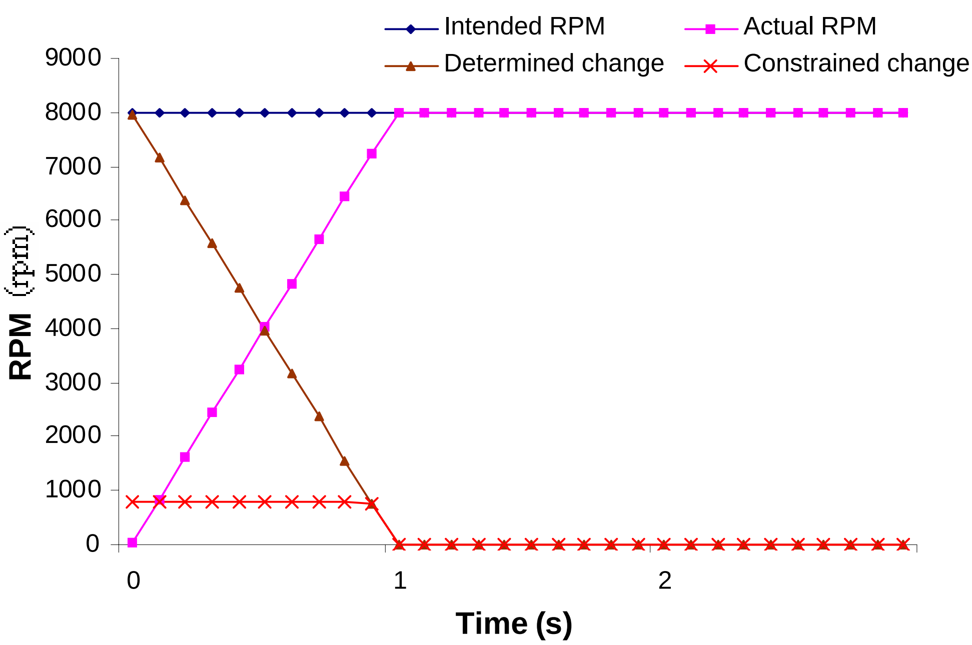


b. Pitch behavior.

Fig (4.23) Height and pitch of a simulator use the time averaging method at climbing status.

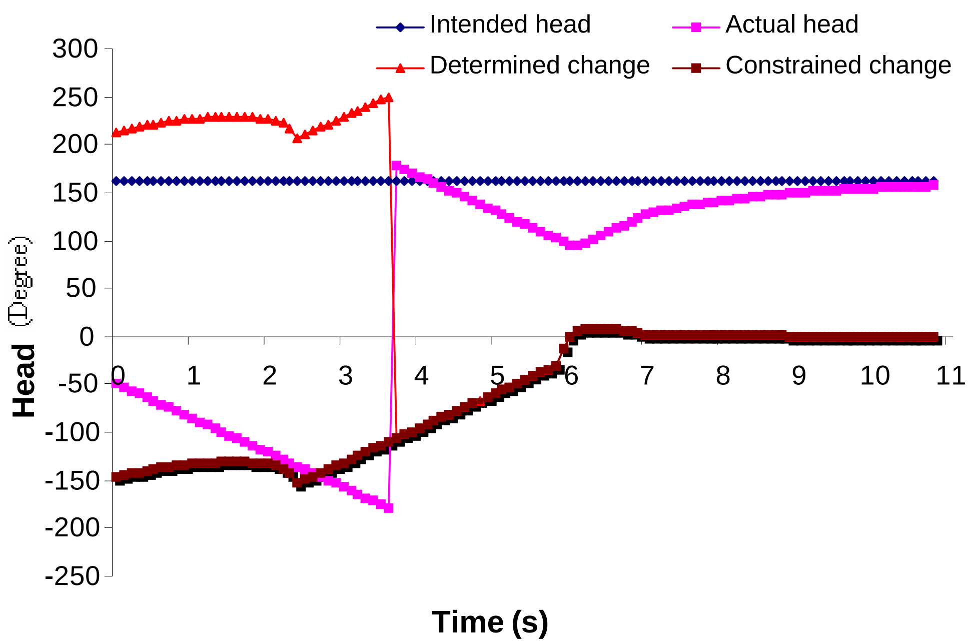


a. Speed behavior.

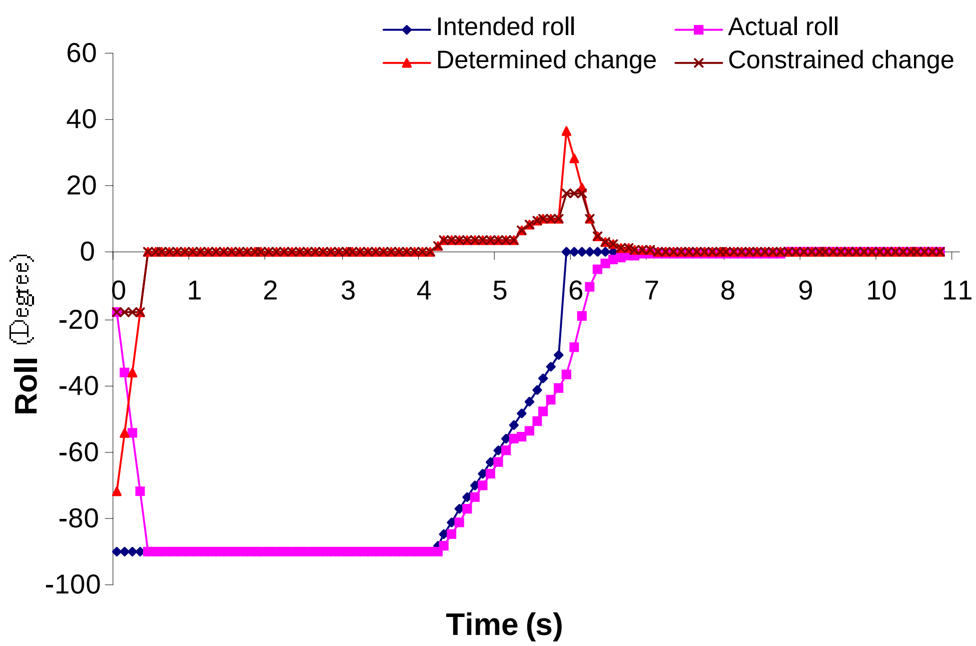


b. RPM behavior.

Fig (4.24) Speed and RPM of a simulator use the time averaging method at speedup status.

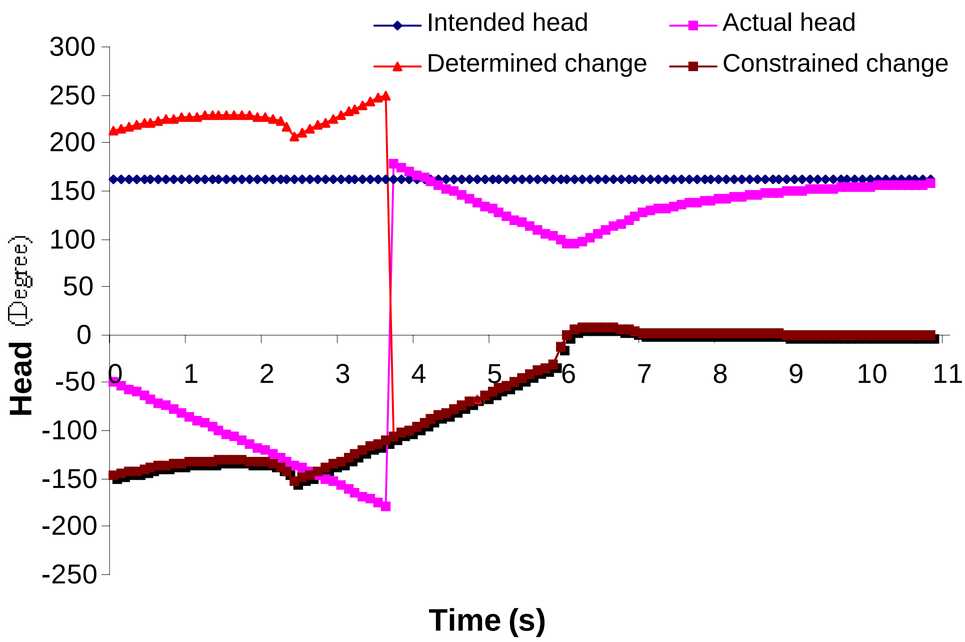


a. Head behavior.

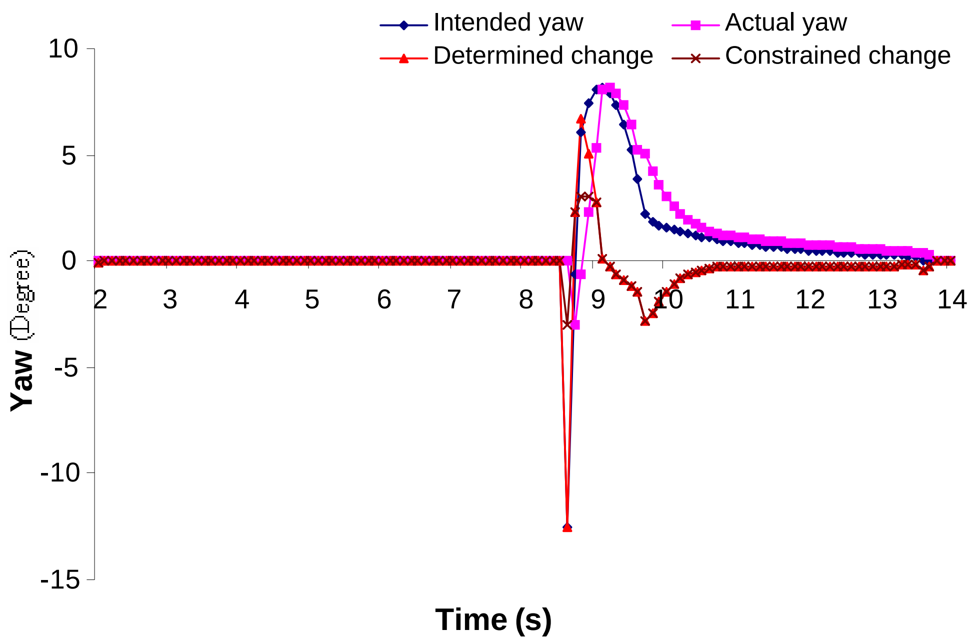


b. Roll behavior.

Fig (4.25) Head and roll of a simulator use the time averaging method at left turn status.



a. Head behavior.



b. Yaw behavior.

Fig (4.26) Head and yaw of a simulator use the time averaging method at left turn status.

The test issues for different models of the response coefficients show that there is one shortcome affect the simulator during its mission, which is the overshot. The analyses of the two adopted methods used to determine the simulator's influence show that both possess the ability of reaching the desired situations by the same time; this time was 2.1 *s* in Figure (4.18) and 2.2 *s* in Figure (4.23). Moreover, the comparison between them and the case of the static response that shown in Figures (4.4-4.11) the simulation results shows they are faster than the static method about 6 times fore the same stability. The rapid response of the two adopted methods comes from dynamic treatment of k_{2i}, where it was given higher value at high detected deflections, and then it decreased with the gradual decrease of the deflection.

The comparison between the two proposed dynamic method shows the deflection based method had some irregularities that might appear in the behaviors of flight parameters, it shows less overshot since it deals with the actual values of the deflections which make the deflections and the behaviors of the flight parameters were sharper. While the time averaging method provided smooth behavior than the deflection based method, but relatively larger overshot. The consideration of the response coefficient in the time averaging method along $N = 25$ intervals make the behavior of k_{2i} smooth and, consequently, the actual deflection appear smoother. The overshot of the time averaging method depends on the behavior of the response coefficient with respect to time; this made the change of k_{2i} has a delay with respect to time of change of the simulator's situations. Less value of N make the response coefficient appears sharper, the overshot being smaller, and the behavior of the flight parameters being sharper. Larger value of N make k_{2i} seems to be smoother, the overshot became larger, and the behavior of the flight parameters became smoother, which causes some late in the change of k_{2i}. Therefore, it is necessary to choose a proper value of N at which the influence will be the best in terms of both overshot and smooth behavior.

It should be mentioned that the values of the actual changes (A) and planned changes (P) were determined depending on the navigation parameters not on the flight parameters, since the behavior of the navigation parameters were smoother than the flight parameters, furthermore the use of flight parameters may cause passage through a confusion at the transient state which reverse the sign of the deflection and affect the simulator stability.

Figures (4.27-4.30) show the behavior of A, P, and k_{22} for both the navigation and flight parameters (height and pitch), which correspond to their behaviors in case of climbing, see Figures (4.18) and (4.23). The behaviors shown in Figures (4.27) and (4.28) do not give good indication about the determination of k_{22}, because the behavior of k_{22} was not monotonic, it was negative, positive, and have clipped values along the time of determinations. While the behavior of k_{22} shown in Figure (4.29) and (4.30) was following and behaved as wanted to be, at first stage it was greater than 1 (when $\Delta N_2 > MaxChgN_2$), then it was gradually decreased until reaching $k_{22} = 1$ at ($\Delta N_2 = MaxChgN_2$), and finally it was remained equal to one at ($\Delta N_2 < MaxChgN_2$). Therefore, it is suitable to determine k_{22} by using of the navigation parameters.

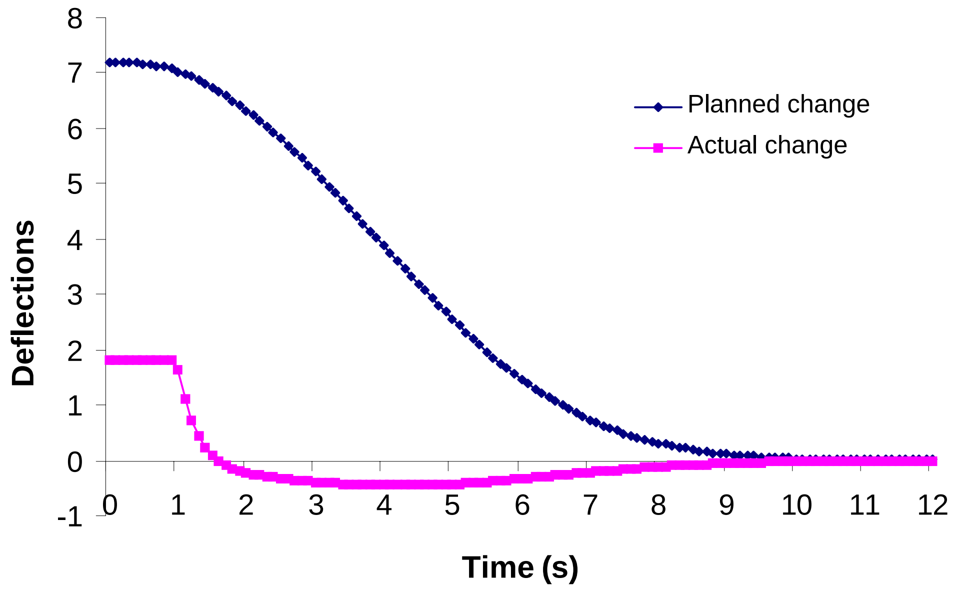


Fig (4.27) The actual and planned changes of the simulated flight parameters.

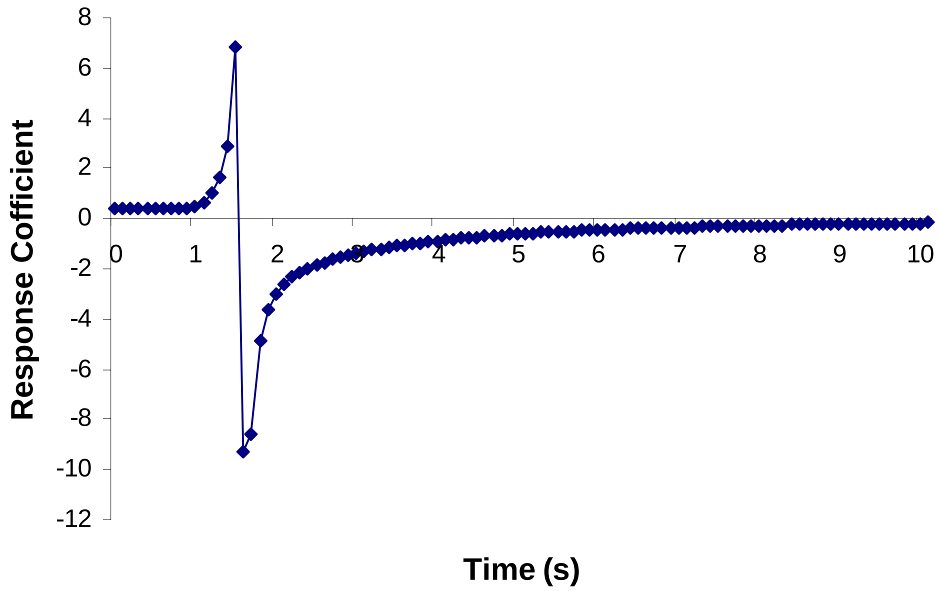


Fig (4.28) The response coefficient k_{22} computed by the simulated flight parameters.

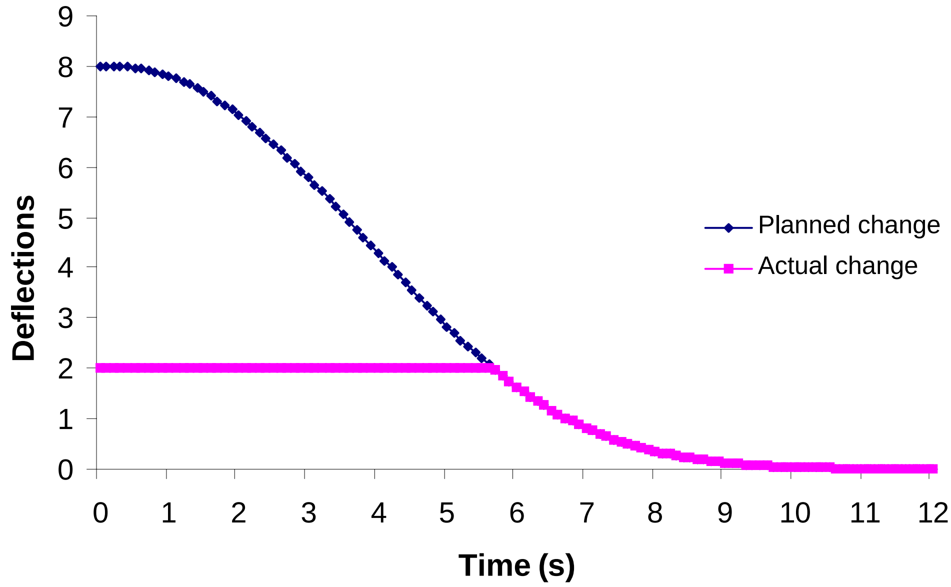


Fig (4.29) The actual and planned changes of the simulated navigation parameters.

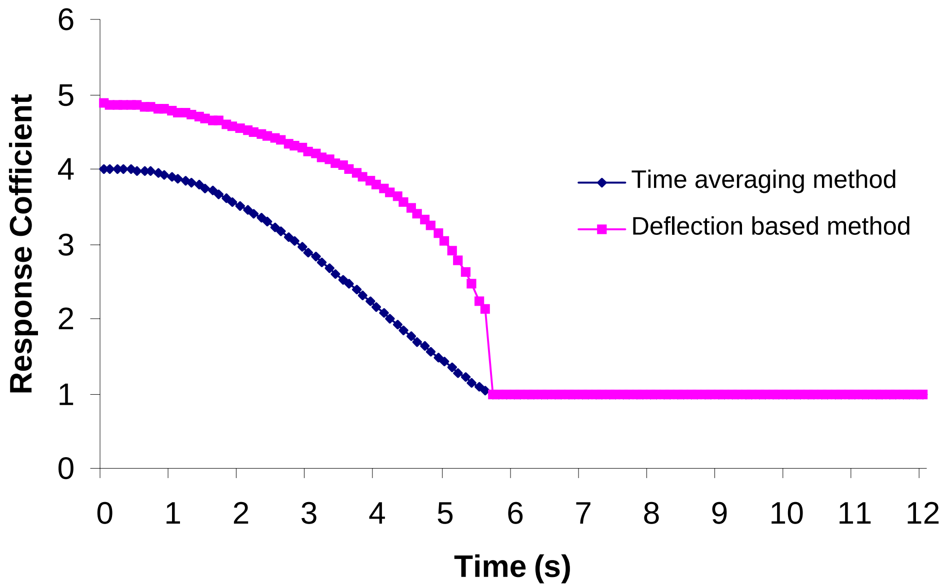


Fig (4.30) The response coefficient k_{22} computed by the simulated navigation parameters.

It is noticeable at the first intervals; the planned changes (*P*) are large in comparison with the actual changes (*A*), so the response of the simulator became higher. For correction purpose, *P* was decreased by an amount greater than that of the previous time, while *A* was for the two cases; if ΔN_i is greater or equal to $MaxChgN_2$ then *A* was constant, since the same increment of previous time was added to *A* (i.e., $MaxChgN_2$), else if ΔN_2 is less than $MaxChgN_i$ then *A* decreased by the same decreasing of *P*, so *A* was being equal to *P* and $k_{2i} = 1$.

The rate of increase or decrease in both *A* and *P* is respectively, proportional to the amount of the correction, which lead to increase or a decrease in value of k_{22}. The different incremental or decremental rate at each interval made *P* and *A* are behaved in non-linear manner when $\Delta N_2 \geq MaxChgN_2$, and k_{22} behavior being non-linear increate, because of the change occur on *P* only. Continuing the correction made *P* decreased toward *A* further, this situation is terminated when *P* became equal to *A*, the same changes occurred in both *P* and *A* make their ratio tend to be unity, and k_{22} behaved linearly when $\Delta N_2 < MaxChgN_2$. In general, it was noticed that the actual change goes toward the planned changes, and the corrections goes toward the desired. The response coefficient decreases slowly by increasing, gradually, the correction until there is no deflection, where the response coefficient is unity and both *P* and *A* are equal.

The second curve in Figure (4.30) is the behavior of k_{22} when the deflection based method was applied. The linearities are found in the beginning and last intervals, while it was curved at middle intervals. Thus, the difference between them was the largest at the middle. Keeping the response values high in the first intervals only lead to the occurrence of less overshot, while the large decay at middle intervals interpret the occurrence of less oscillation in the deflection based method. Whereas the first behavior possess an inverse conditions and results.

In order to test the performance of both suggested methods, the case in which the navigator tries to navigate the simulator that suffer a positioning deflection but there is an opposite effect resists its navigation was considered. Such case occurs when there is an external effect (such as wind resistance) affects the simulator performance. It is assumed that the opposite action of the resisted effect make the determined actual change is zero at each interval, so the simulator will always show a deflection, as shown in Figure (4.31).

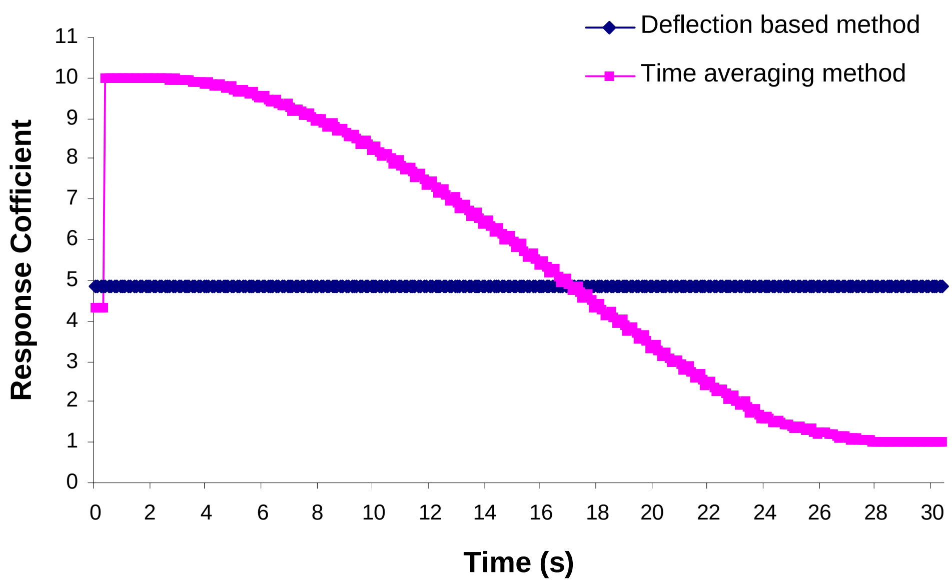


Fig (4.31) The response coefficient (k_{23}) behavior computed by the two adopted methods at abnormal situation.

It was noticed that the needed correction was very large at the first interval, so the response was high. Later, the value of *P* maintained as it was, while *A* reached zero.

The use of the deflection based method gave same values of k_{2i} at all times since the actual changes (*A*) zero, which make the calculation of k_{2i} depends on the planned changes (*P*) only. Whereas the use of time averaging method gave the highest value of k_{2i} since the ratio of *P*/*A* was infinity (*A*=0). This may lead to terminate this situation by making the correction slowly progressed to the desired situation, and consequently the simulated behavior will be slow which cost a long time.

The above interesting note proves that the time averaging method is better than the deflection based method because the latter does not offer any improvements at the abnormal situations (the simulator stay far away from the desired conditions without correction). While, the time averaging method provides the ability of terminate this situation of time delay by making k_2^i maximum. The results show that the deflection based method spent 8 *s* for reaching the desired situation, while the time averaging method does not make any correction and the deflection remain as it is (i.e., the correcting time was same). It should be mentioned that both methods have spend a time interval (5 *second*) when there was no external effect.

4.4 Results of Simulated Wind Effect

In the purpose of testing the simulator that face the effect of the wind, the wind had been considered to be moved with a speed of 15 *km*/*h* toward the direction of 315^o northern west (i.e., -45^o in the considered direction) and strikes the simulator that flying by a speed of 400 *km/h* in direction of east (i.e., 90^o in the considered direction), with head deflection of 90^o and height deflection of 5 *km*, *r* was assumed to be 1.5 *m*.

Figure (4.32) shows the navigator interface that regarded the wind effect, the thick arrow appears in the upper left corner of the graphical map of the navigator interface refers to the direction of the wind. It was noticed that the pointer of the throttle gauge had increased above its normal state in order to compensate the decreases occurred in the simulator's speed due to the opposite wind. Also, the pointer of the aileron gauge was set (not zero) to handle the deflections in the head and position.

Figures (4.33) illustrate the behaviors of the actual speed, true speed, *RPM*, and throttle in the two cases (existence and absence of the wind effect). Figures (4.34-4.36) present the behavior of the other navigation, flight, and control parameters in the two cases (existence and absence of the wind effect).

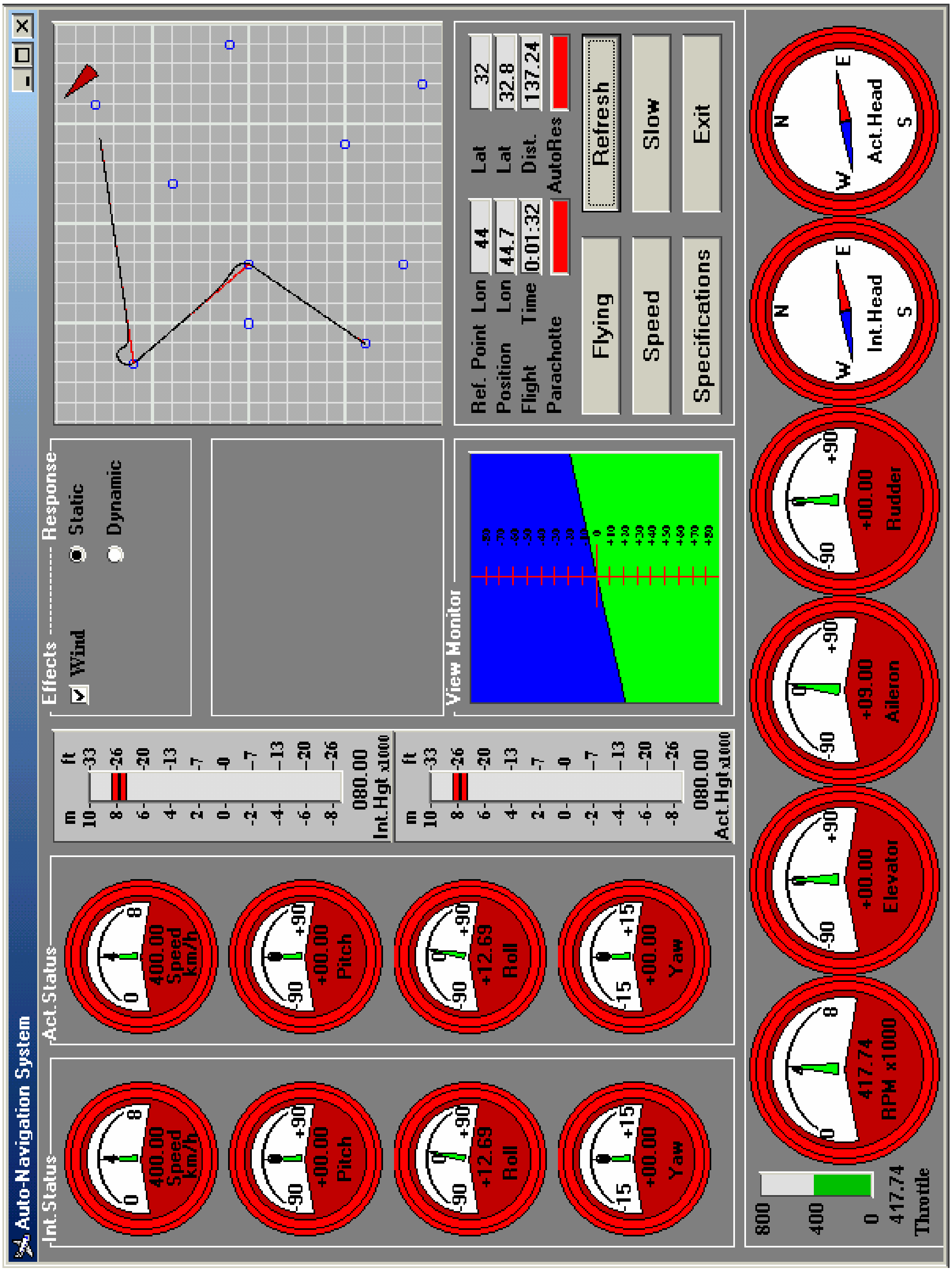
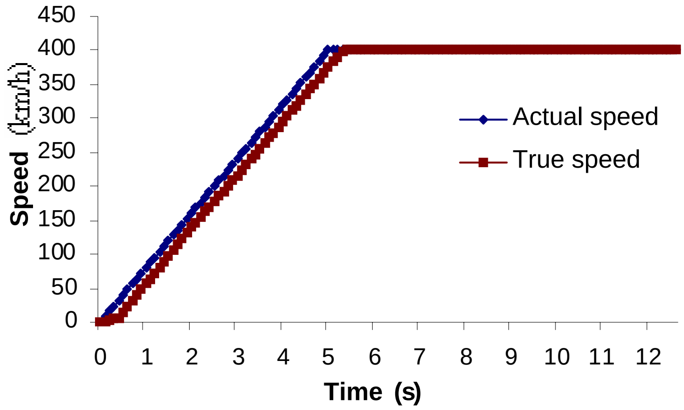
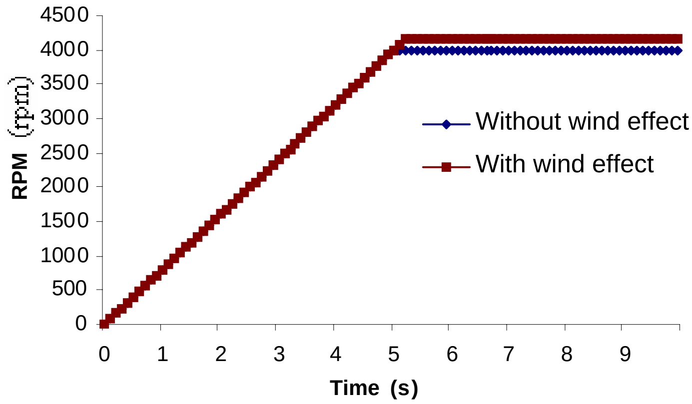


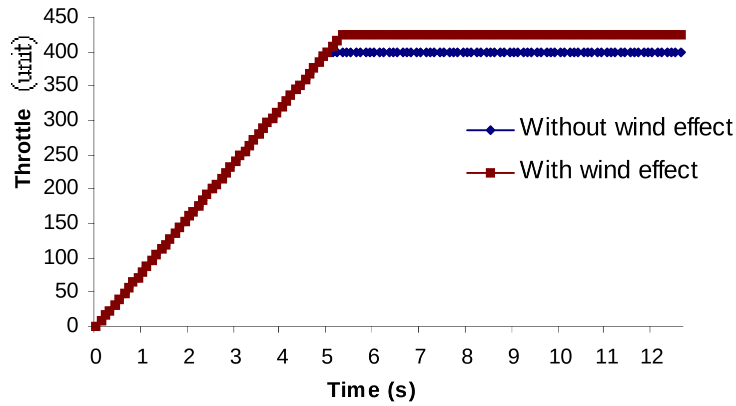
Fig (4.32) Navigator interfac that handle the wind effect.



a. Speed behavior.

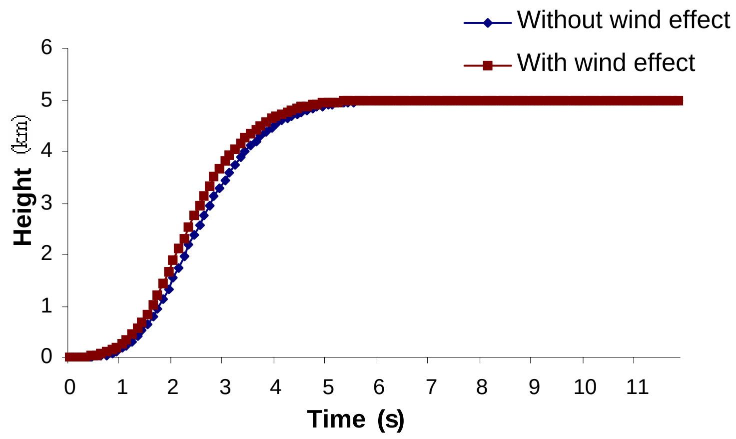


b. RPM behavior.

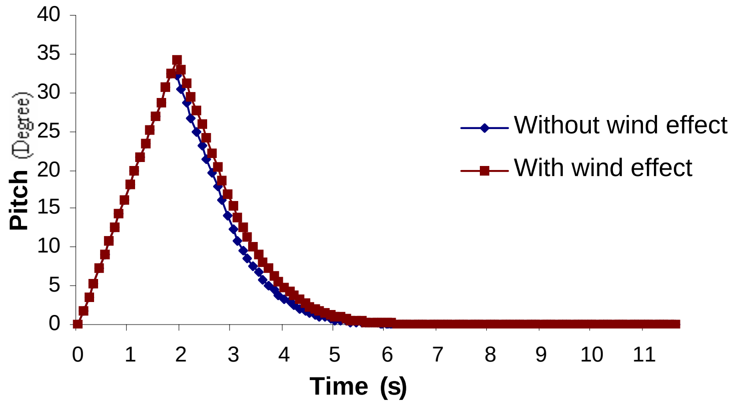


c. Throttle behavior.

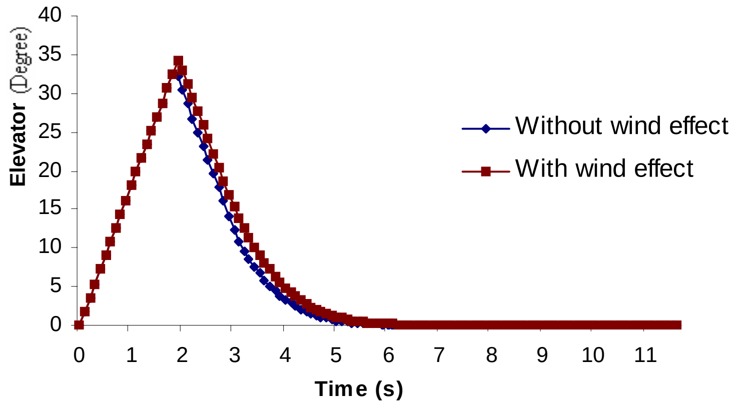
Fig (4.33) The behavior of simulated speed, RPM, and throttle at an opposite wind effect.



a. Height behavior.

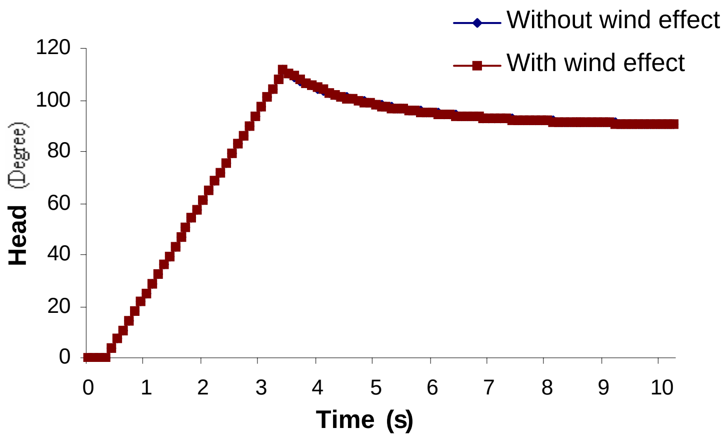


b. Pitch behavior.

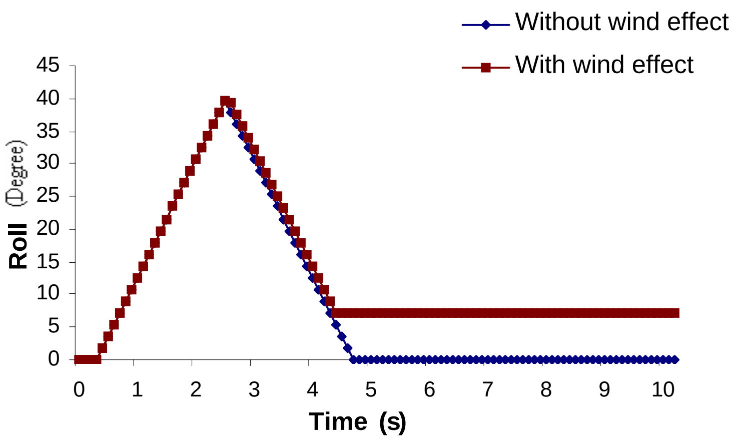


c. Elevator behavior.

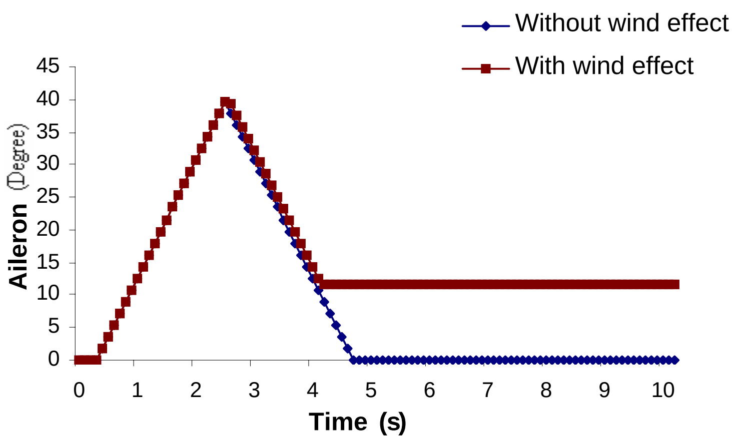
Fig (4.34) The behavior of simulated height, pitch, and elevator at an opposite wind effect.



a. Head behavior.

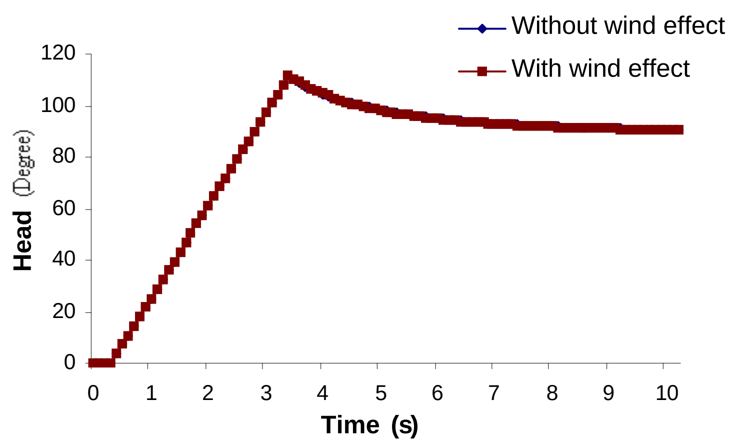


b. Roll behavior.

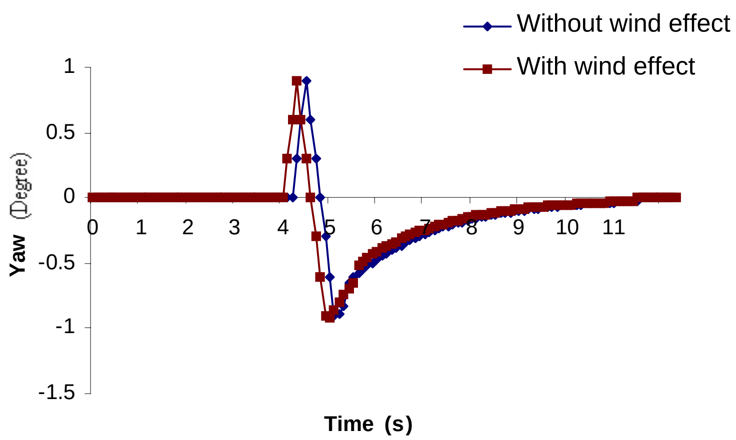


c. Aileron behavior.

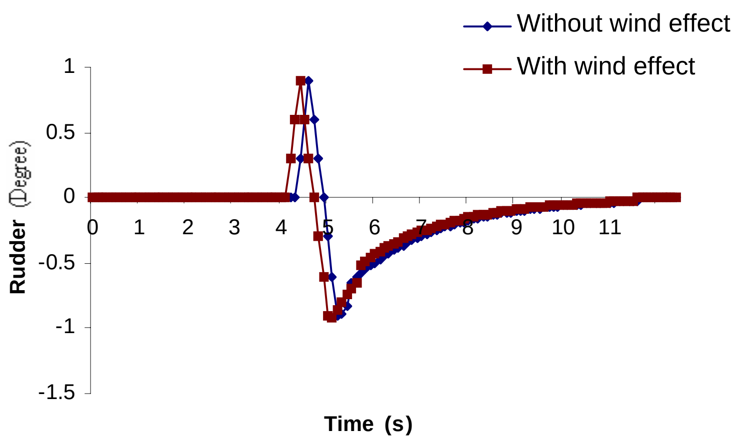
Fig (4.35) The behavior of simulated head, roll, and aileron at an opposite wind effect.



a. Head behavior.



b. Yaw behavior.



c. Rudder behavior.

Fig (4.36) The behavior of simulated head, yaw, and rudder at an opposite wind effect.

In both cases of interest, the actual speed reached to the same amount of the intended speed. In the case of absence the wind effect, the simulator results indicate normal behavior; throttle was 400 unit, *RPM* was 400 *rpm*, and both the actual and true speeds were 400 *km*/*h*. Whereas in the case of existence the wind effect, the navigator ordered throttle to increased by 35 unit, so it became 435 unit, this caused the *RPM* to be increased to 4350 *rpm*. This increase in the *RPM* led the true speed to be increased to reach 435 *km*/*h* in order to keep the actual speed at the normal state (i.e., 400 *km*/*h*). The successful basis of the adopted control model makes the increases in both throttle and *RPM* compensate the loss of speed, so the actual speed was compensated exactly. The simulated true speed obeyed the throttle behavior where it increased when throttle is increased. While the actual speed obeyed the navigator control where the navigator tries to keep the actual speed at the intended value.

Also, the time spent to reach the intended value of the height was 5.5 *s* in the case of absence of the wind effect, and it was 5.2 *s* in the case of existence the effect. The elevator returned back to its normal state in both cases, since the wind affects the height only and it has no effect on the pitch. The opposite wind had make the elevator (and then pitch) return back to the normal state with a time less than the climbing time (fast climbing). The climbing rate was increase because the vertical component W_{Ver} of the wind speed produces an upward force that strengthens the lift force and consequently decreases the climbing time. Whereas an inverse situation was found when the direction of the wind has the same direction of the simulator, i.e. the climbing time becomes larger (slow climbing) since the vertical component W_{Ver} of the wind speed produces downward force against the lift force. In the case of flight descending, the descending rate behaved similar to that of the climbing with respect to the wind direction, i.e. the descending was fast when the wind was opposite to the direction of the simulator and it was slower when the wind has the same direction.

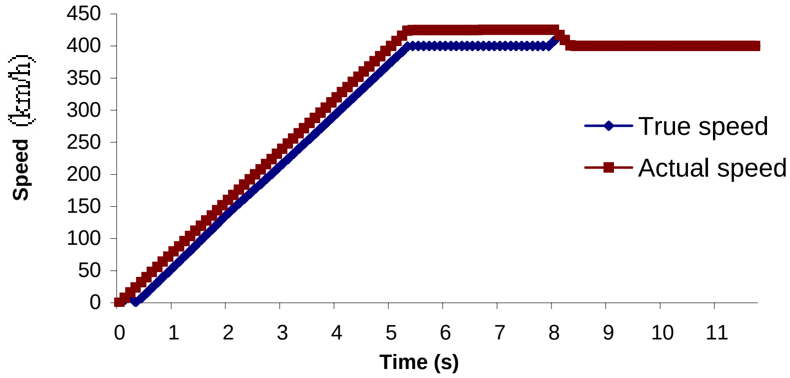
During a right turn toward the third route segment, in either cases of interest it is noticed that the aileron was used to correct the heading. After the turn, the aileron was kept deflected by an angle. It is well-known that the heading deflection is due to both the positioning deflection ΔR_P (due to the effect of side wind) and roll deflection ΔR_T (due to the effect of rotational torque). Thus, the aileron carries the responsibility of compensating the positioning error. At the same time, the navigator ordered the aileron to correct the roll by taking an additive angle $-\Delta R_T$ to keep the roll at ΔR_P; this is the reason behind disappearance of ΔR_T in the behavior of the roll. As a result, the averaging of the simultaneous deflection and correction of the aileron had adopted a value about 12^{o}, while the roll take just 8^{o}, which implies ΔR_T=4^{o}.

In addition, the rudder did not share in compensating the wind effect, because of any small yawing besides the high speed of the airplane may lead to large deflection in the head, so it is better to avoid this bad attitude by leaving the rudder as it is. The value of the observed rudder angle raised in the case of wind effect, and its value remains non-zero for a time period. The length of this period was less in the case of wind effect. This associated effect come from the relation between the head and position compensation process and the rudder surfaces control.

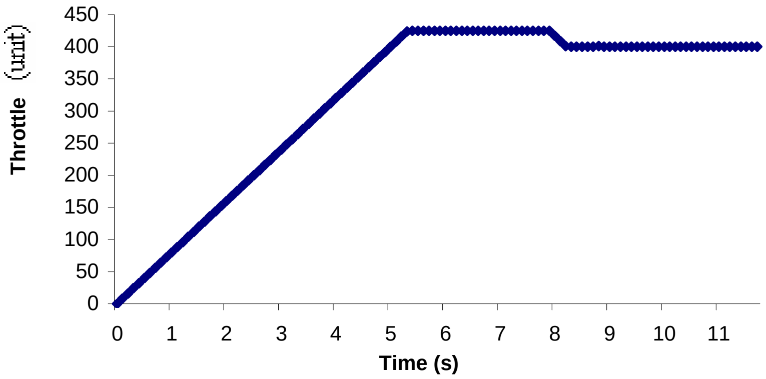
The comparison between the behaviors of the control parameters indicates that the final states of both throttle and aileron gained a nearly constant increase above their normal states to compensate the difference occurred in speed and correction angle of roll during the time of the opposite effect existence. This behavior did not match with the behavior of other control parameters (elevator and rudder) since the wind effect has indirect effect on both of them. This indirect effect appears as a side effect corrected by the navigator automatically. This is the reason behind making an increase in the throttle and aileron over their normal states at the intervals comes after reaching the intended value. The increase of the other parameters was different at each interval according to the correction scenario, a greater increase occurred during the middle intervals (before reaching the normal state).

The difference between any two behaviors, associated with the two cases of interest, was dependent on the ratio between the wind speed and flight speed. This difference was high when the ratio is high, and vice versa. Moreover, the observations showed that the behavior of the flight parameters was similar to their corresponding of control parameters, so the instantaneous events of increasing or decreasing the control parameters had the same events occurred in the flight parameters. Whereas, the observations on the navigation parameters showed that all of them have reached the intended situations in both cases (existence and absence the wind effect), which confirm the ability of the navigator to control the simulator well.

When the wind effect had suddenly removed during the flight, the navigator made the actual and true speed reach the intended speed, as shown in Figures (4.37), where one can notice that 8 *second* the wind effect was passed.



a. Simulated behavior of actual and true speeds.



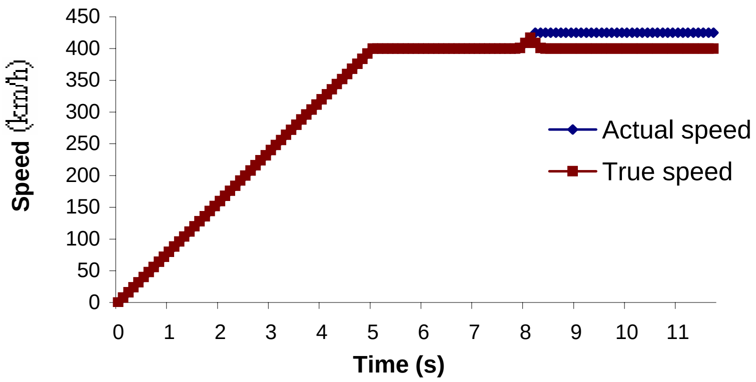
b. Simulated behavior of throttle.

Fig (4.37) The behavior of simulated actual speed, true speed, and throttle when the opposite wind effect removed suddenly at time 8 *s*.

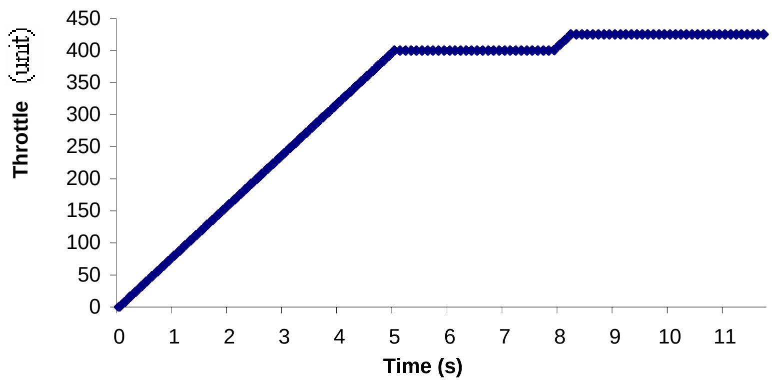
After the removal of wind effect the throttle that took a value greater than the normal, made the actual speed increases toward the true speed slowly. At specific time, the navigator detected the increase of the actual speed, so it had ordered the throttle to be decreased, which in turn caused decrease in the true speed. The increase of the actual speed and decrease of both the throttle and true speed had been continued till both the actual and true speed become equal, but their values are greater than the intended value. Therefore, the decrease of the throttle continued toward the normal state, both the actual and true speed decreased until they reach the intended speed. An inverse behavior was noticed in the case of sudden enabling an opposite wind. Throttle and *RPM* were increased, and value of the true speed increased above the actual speed, as shown in Figure (4.38), where the time of enabling the wind was 8 *s*econd.

As a summary, the established navigator is capable to control the airplane flight when there is a wind affect the simulator. The most interesting remark is the behavior of the navigation parameters is same in both cases (existence or absence the wind effect). The associated control parameters have taken specific states that are proportional with the speed and direction of the effective wind.

Also, the simulator succeeded in describing the speed, height, and head. The simulation results showed that the wind will cause increase/decrease in the speed of the airplane, and the opposite reaction of the navigator is to increase/decrease the amount of the issued throttle command to correct the deflection of the speed. The side wind deflects the airplane position, while the direct wind affects the speed only and not the position of the airplane. The navigator orders the aileron/rudder to take a specific angle proportional with the occurred defection of the position. Also, the wind affects the time of reaching the airplane to the intended height when both the wind velocity and airplane velocity are oblique with each other. This time increases when the wind is opposite to the direction of the airplane, and it decreases when the wind has the same direction of the airplane.



a. The behavior of actual and true speeds.



b. The behavior of throttle.

Fig (4.38) The behavior of the actual speed, true speed, and throttle when the wind effect enabled at time 8 *s*.

4.5 Results of Simulated Engine Efficiency

The adoption of equation (3.53) to control the flight parameters led to the results given in the following paragraphs, which describe the observed behaviors of the navigation parameters at different engine efficiencies;

Figure (4.39) presents the establish navigator interface to setup the efficiency of the engine. As a first test the engine efficiency was assumed 80%. It was noticed that the amount of the spent throttle is increased above its normal value in order to provide the desired speed. A specific scroll bar appears in the navigator interface when the engine efficiency is checked, the efficiency value was limited between 50% and 100%. During the test it was noticed that the pointer of the throttle gauge was pointing to a value greater than its normal state in order to substitute the fallback of the speed due to the lower engine efficiency. Both the pointers of the *RPM* and speed gauge have pointed to the intended values, which refers to the wise control of the navigator. Other gauges belong to the remaining parameters were stable and not much affected, this refers to less effect of the engine efficiency on navigation other parameters.

Figure (4.40-4.43) picture the behavior of the navigation, flight, and control parameters for three different values of the engine efficiency (90%, 80%, and 70%). These behaviors have been compared with their corresponding behaviors computed when the engine efficiency was taken (100%). The qualitative comparison shows how the engine efficiency affect the behaviors of the parameters and how much the effect was.

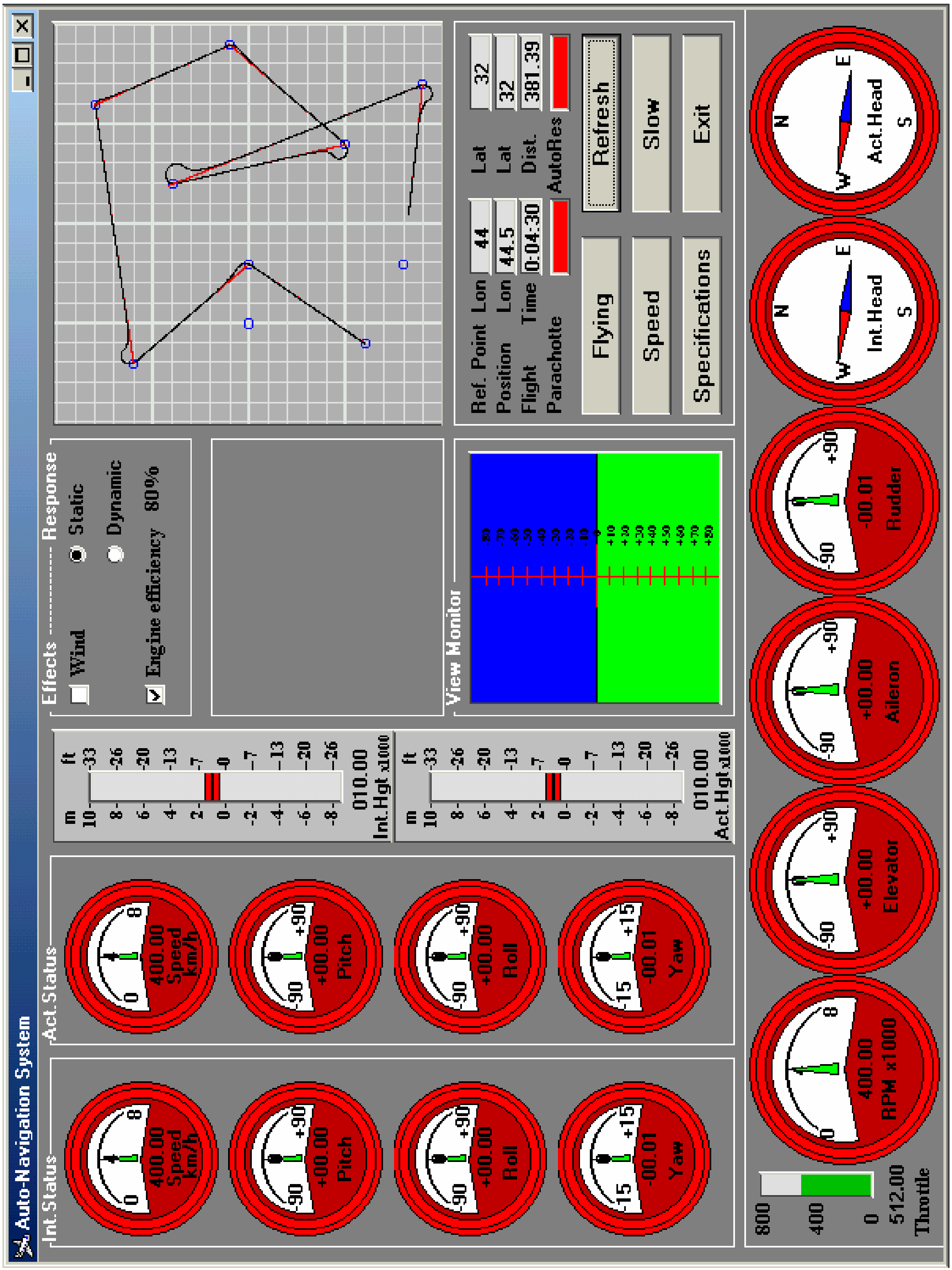
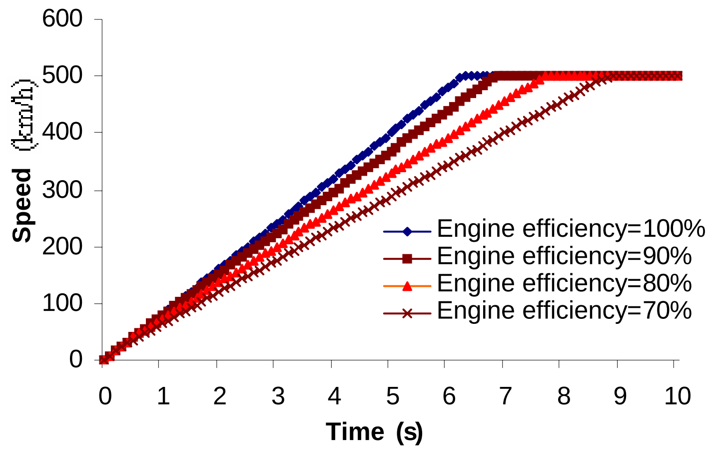
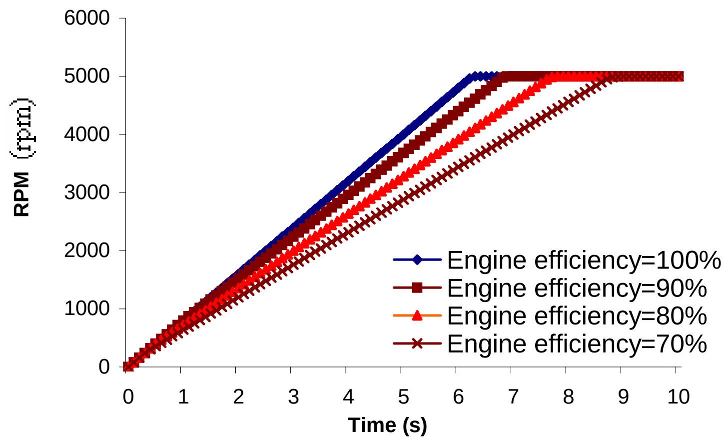


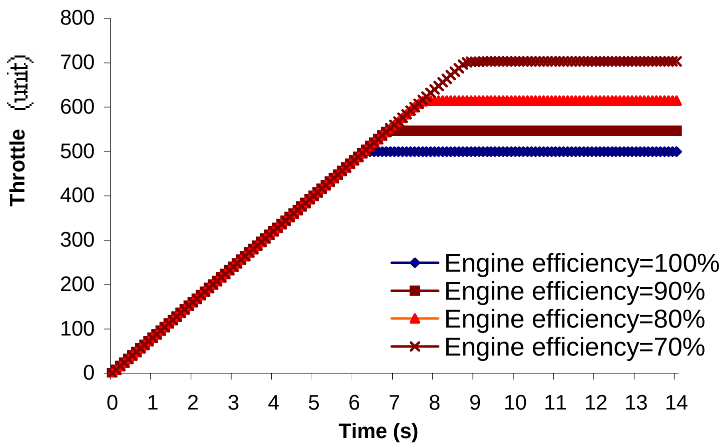
Fig (4.39) Navigator interface that handles the engine efficiency effect.



a. Speed behavior.

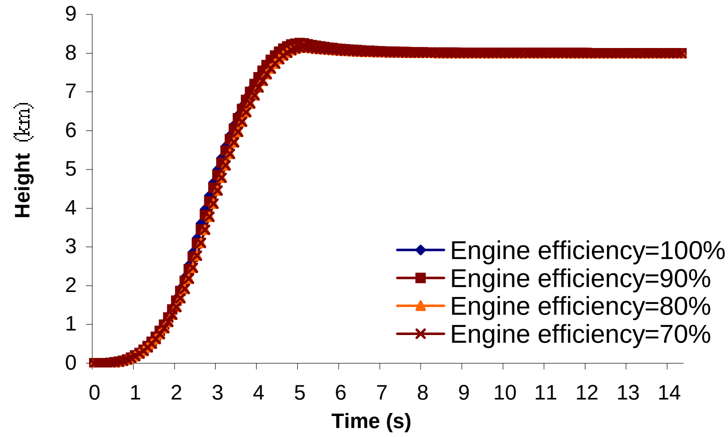


b. RPM behavior.

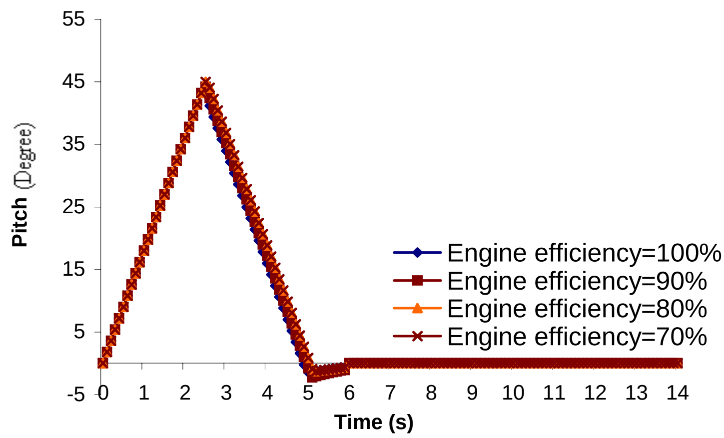


c. Throttle behavior.

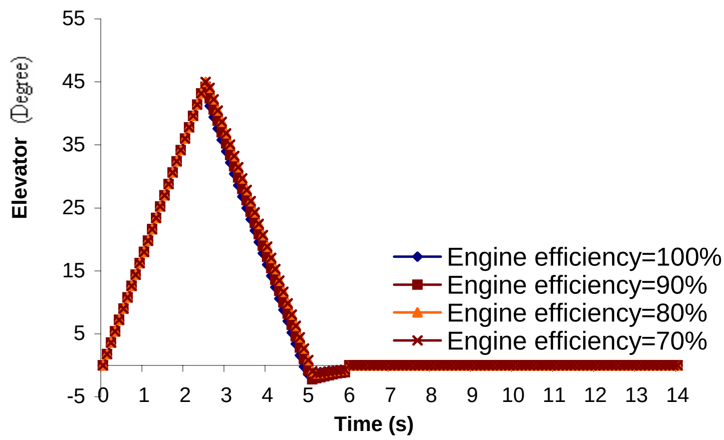
Fig (4.40) The behavior of simulated speed, RPM, and throttle for different engine efficiencies.



a. Height behavior.

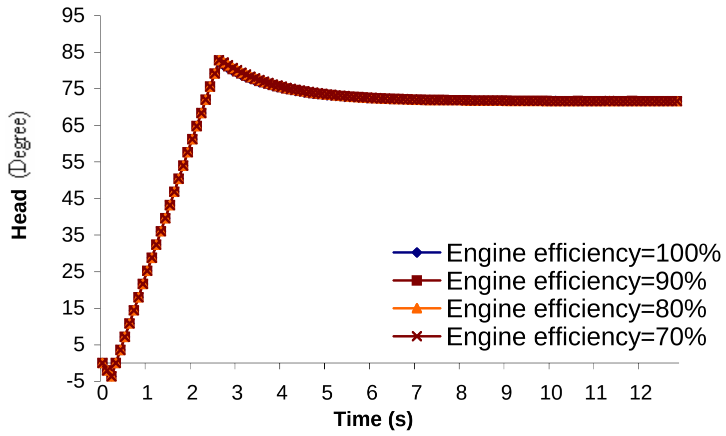


b. Pitch behavior.

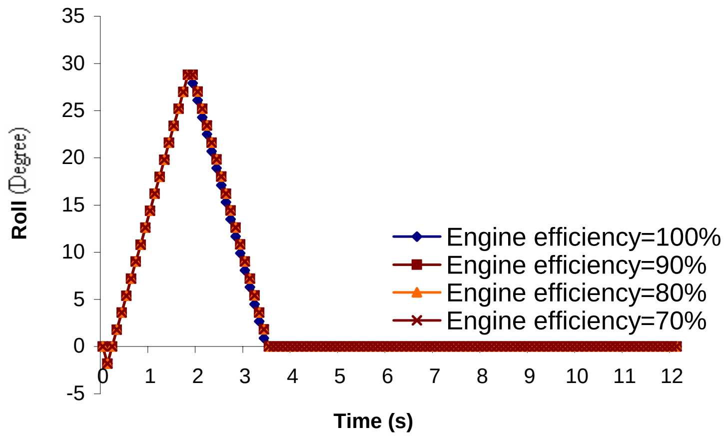


c. Elevator behavior.

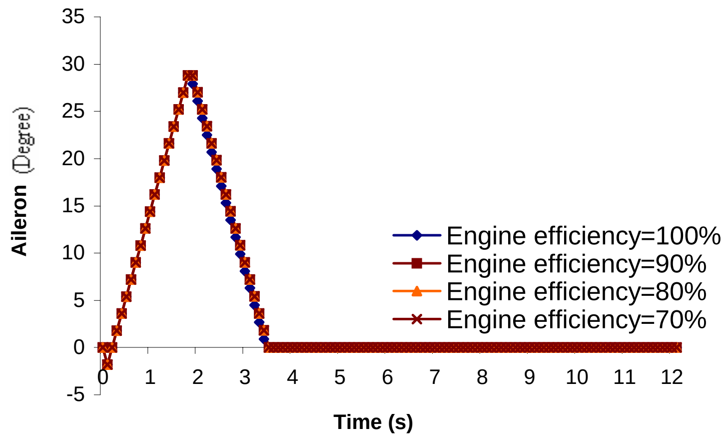
Fig (4.41) The behavior of simulated height, pitch, and elevator for different engine efficiencies.



a. Head behavior.

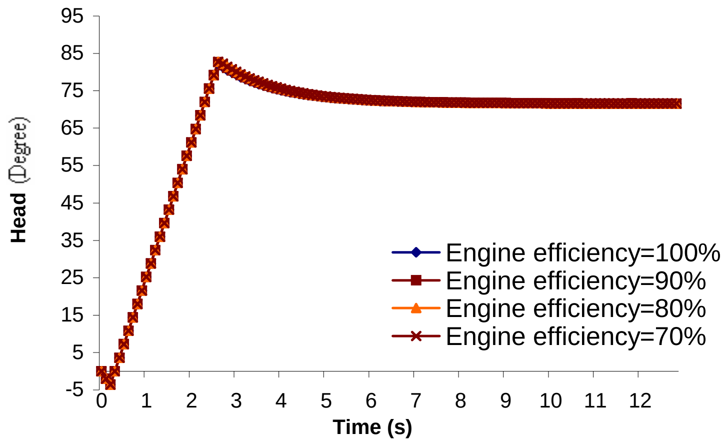


b. Roll behavior.

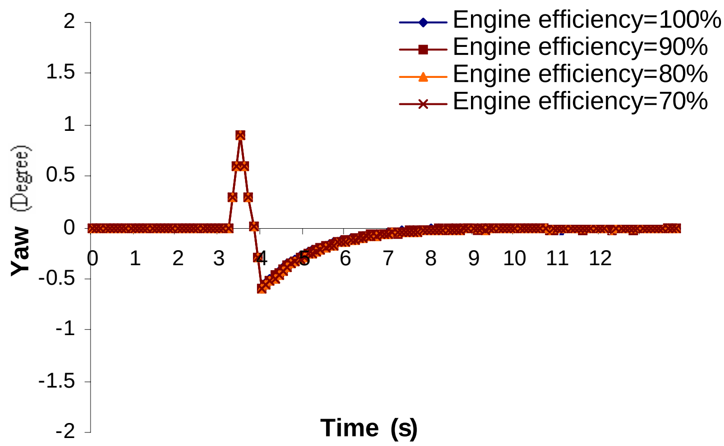


c. Aileron behavior.

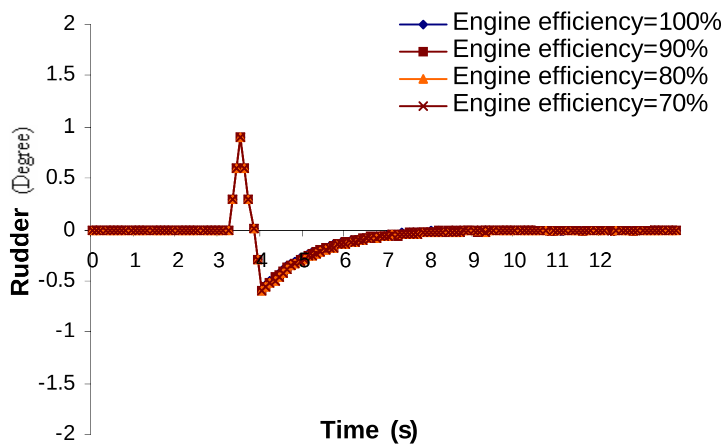
Fig (4.42) The behavior of simulated head, roll, and aileron for different engine efficiencies.



a. Head behavior.



b. Yaw behavior.



c. Rudder behavior.

Fig (4.43) The behavior of simulated head, yaw, and rudder for different engine efficiencies.

The major effect of the engine efficiency is on the speed and its related parameters. In general, all the navigation parameters had been under the control of the navigator since all of them reached to their intended conditions without any significant difference in the needed time interval to reach. The successful follow toward the intended conditions refers to the successful mathematical treatment of the engine efficiency.

It was noticed that the difference in engine efficiency causes different amounts of increase in the speed. The speed was increased by different rate for each considered efficiency. The lowest efficiency led to the least increasing rate of the speed; which had led the actual speed spend longer time to reach the intended speed.

Also, the slope of the growing portion in the speed curve decreases with the decrease of the engine efficiency. The projection of the growing portion of the speed on the time axis represents the time needed to reach the intended speed. The determined time of reaching the intended speed for the navigation, flight, and control parameters was 6.3 *s*, 7.3 *s*, 8.3 *s*, or 9.3 *s* when the engine efficiency was taken 100%, 90%, 80%, or 70%, respectively, these values are characterized by the equal differences between them. Therefore, the response time (i.e., the time interval require to reach the intended value) increases when the engine efficiency is decreased.

In addition, the behavior of the *RPM* at different engine efficiencies are very similar to that of the speed because of the linear relationship between them. Thus, all the events occurred on the speed have occurred on the *RPM*. One can notice that the throttle was 500 unit when the engine efficiency is 100%, which gave *RPM* value (5000 *rpm*) in order to make speed reach the intended value (500 *km/h*). While, when the engine efficiency decreases to be 90%, 80%, or 70% the amount of the spent throttle of the engine was increased to be around 500, 550, 620, or 710 units receptively.

During the growing portion of the throttle curve, the rates of throttle increase at different efficiencies become close to each other, and they approach to the allowed values of the maximum change of throttle that correspond to the maximum change of speed. When the engine was set fully efficient then the increasing rate of the speed became 7.5 *km*/*h*, which is equal to $MaxChgN_1$, but when the engine was set not fully efficient then the increasing rate of the speed became a *Fraction* of $MaxChgN_1$, it was 7.2, 6.4, or 5.6 *km*/*h* for engine efficiency 90%, 80%, or 70%. This means the behavior of throttle was raised by same increasing rate at all the considered efficiencies, while the behaviors of both *RPM* and speed were raised by different increasing rate according to the efficiency of the engine in use.

After the interval of throttle increases, the throttle become fixed, i.e. throttle did not change since the speed reached to its intended value. This stable state of the throttle was above the normal state in the case of less efficiency. The differences between the behaviors associated with any two successive engine efficiencies was not equal, but they were respective, this is due to inequality of the increasing rate, which had affected the time for reaching the intended speed at different efficiencies.

The behaviors of the remaining navigation, flight, and control parameters are not much differ at different engine efficiencies, they seem to be exact. The matching of the behaviors of these parameters refers to the guidance stability of the established navigator.

The maximum value of the pitch was little inversely increased with engine efficiency to compensate the fallback of the height due to the fallback of speed. The fallback appeared clearly at the region of approaching the intended height.

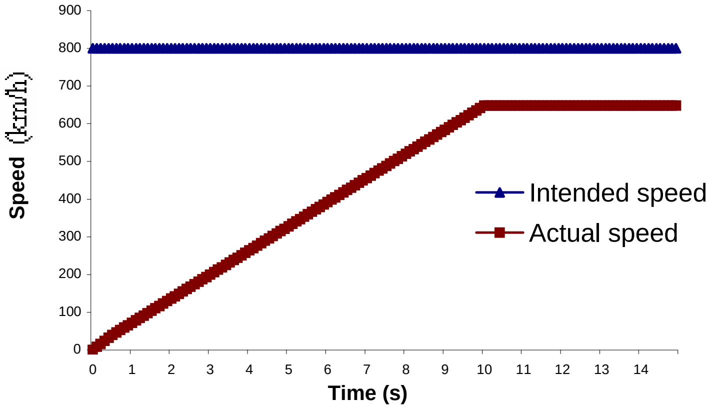
The behaviors of the head and its followed parameters are similar. The fallback of speed had caused a small amount of position deflection (ΔPos), which as a consequence earned very a small share to the head deflection during the time of turn. In general, all the flight parameters do not much affected by the engine efficiency that is due to the stable guidance of the navigator.

An interesting situation appears when the actual speed did not reach its intended value. This situation was found when the intended speed was very high (such as 800 *km/h*) and the engine efficiency was just 80%. Thus, the maximum speed that can provided by the engine was

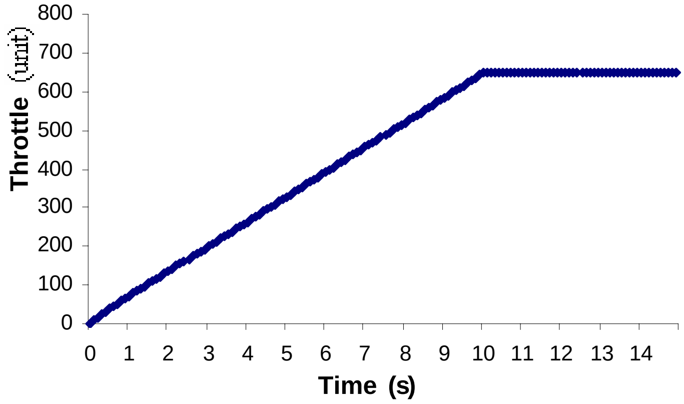
$$\begin{aligned} MaxSpeed_{80\%} &= MaxSpeed_{100\%} \times 80\% \\ &= 800 \times 0.80 = 640\ km/h \end{aligned}$$

To reach the highest possible speed the engine must work in its highest performance, i.e. the throttle must reach its maximum possible value. The *RPM* and speed show less values than the normal (full efficiency), so the speed was maintained on value less than that of intended, as shown in Figure (4.44). In such case, the differences between the behaviors of the navigation and flight parameters have been noticed clearly.

The difference between the actual and the intended speed had affected greatly the related navigation parameters. The head was affected due to the positioning deflection, which was small when the actual speed was low.



a. The actual speed is below the intended.



b. Throttle behavior for inefficient engine.

Fig (4.44) The behavior of simulated speed and throttle when the engine efficiency is 80% and the intended speed is maximum.

4.6 Results of Simulated Altitude Effect

In order to analyze the behavior of the navigator at high altitudes, the minimum and maximum allowed altitudes were assumed to be zero and *MaxHgh* respectively. And the corresponding relative powers have been taken 100% and 50% respectively. The relationship by which the relative power can be determined at any altitude, as shown in Figure (4.45), modeled by using the following linear equation:

$$P = -0.05 \times ActHgh + 1 \qquad \dots (4.3)$$

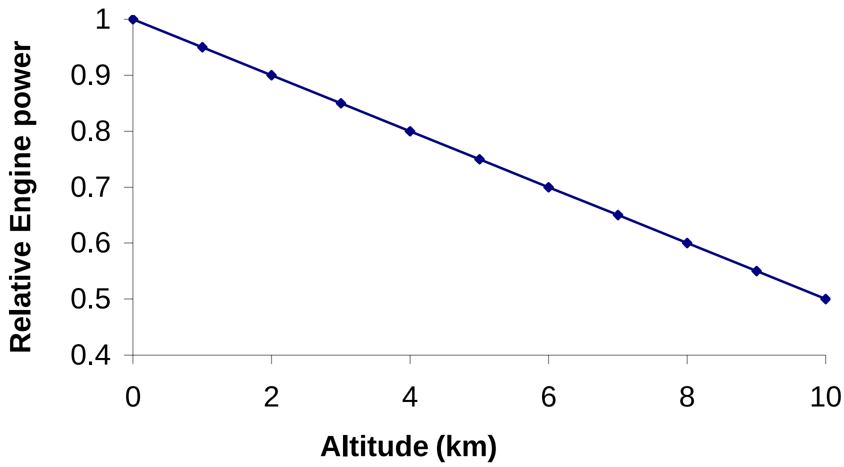


Fig (4.45) The relative engine power of the airplane versus altitude.

Figure (4.46) shows the established navigator interface that used to impose the decrease in the engine power with the increase of the altitude. The number exist beside the third check box in the navigator interface shows the instantaneous value of the engine power. During the flight simulation, it was noticed that the throttle gauge is pointing on a value greater than its normal value, and both the pointers of *RPM* and speed gauges were pointing on their normal values. The pointers of the remaining parameters were pointing to their normal states. As a general view, the navigator had navigated the simulator toward the intended situations and pass the difficulty of decreasing the engine power due to the altitude increase, and the navigation parameters reached to their intended values.

Figures (4.47-4.50) show the behavior of the navigation, flight, and control parameters for an intended speed of 400 *km/h* at three assumed altitudes (i.e., 4, 7, 10 *km*), The behavior of each parameter is compared with its behavior when the altitude is taken (4 *km*) and without effect.

According to equation (3.54), the amount of the throttle had been changed with the change of the actual height, since the engine power changes the thrust, the thrust changes the actual speed, and then the actual speed related parameters will be changed too.

The highest considered altitude is 10 *km*, which causes a reduction in the engine power down to 50%. The throttle became 200 unit which led to 2000 *RPM*, and the actual speed reach 200 km/h. At this altitude, the navigator had compensated the actual speed. The navigator had increased the throttle by the needed amount (i.e., 200 unit), which provided an additive *RPM* (2000 *rpm*) for the engine, and consequently an additive actual speed (200 *km/h*). Thus, the actual speed became 400 *km/h*, which equal to the intended value. In general, at all the considered cases of the altitude the navigator had compensated the actual speed to reach the intended value.

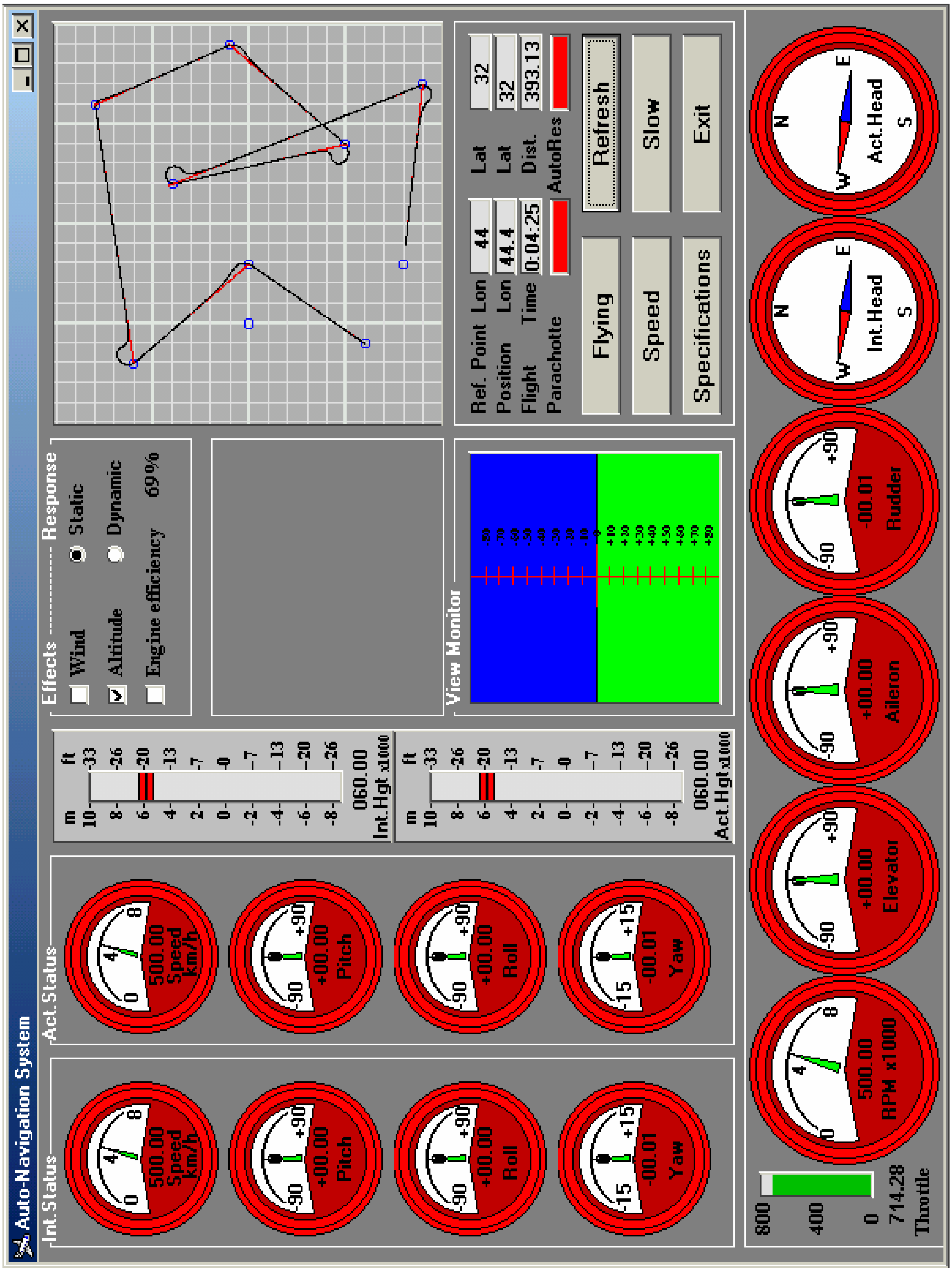
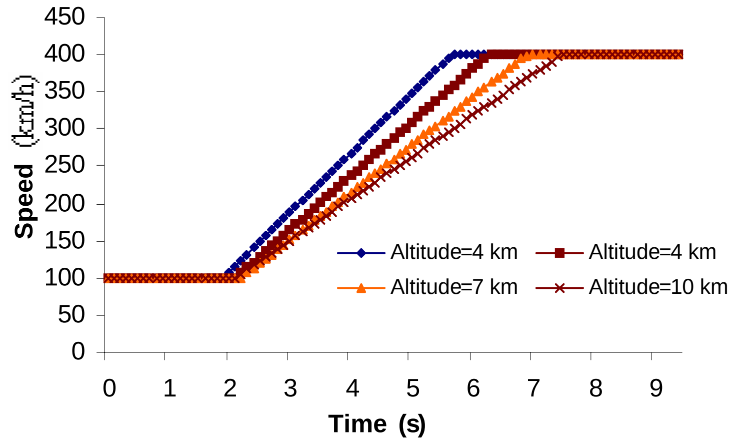
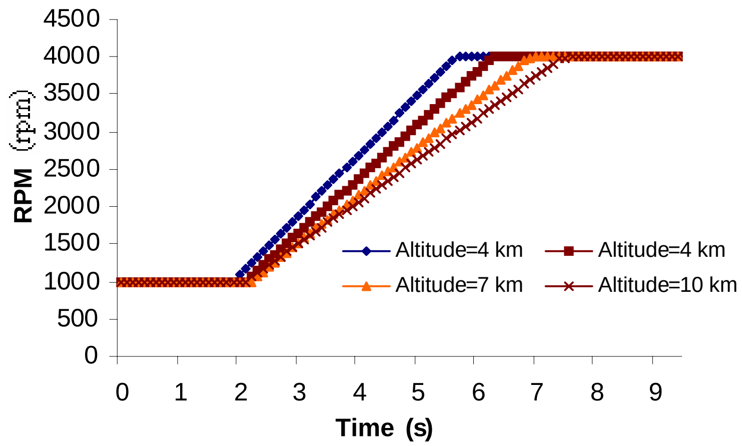


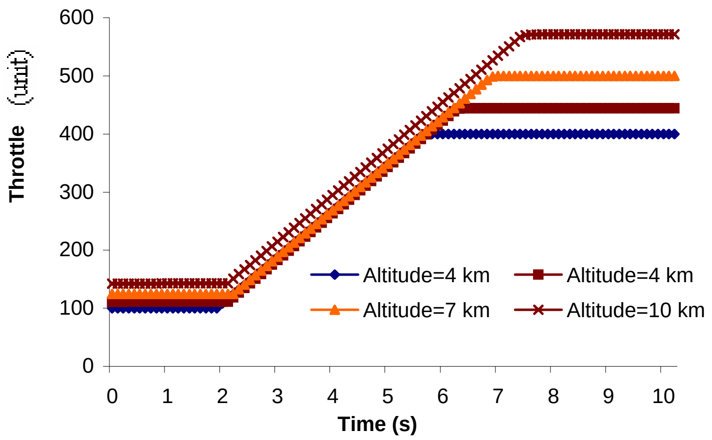
Fig (4.46) Navigator interface used to impose the altitude effect on the engine efficiency.



a. Speed behavior.

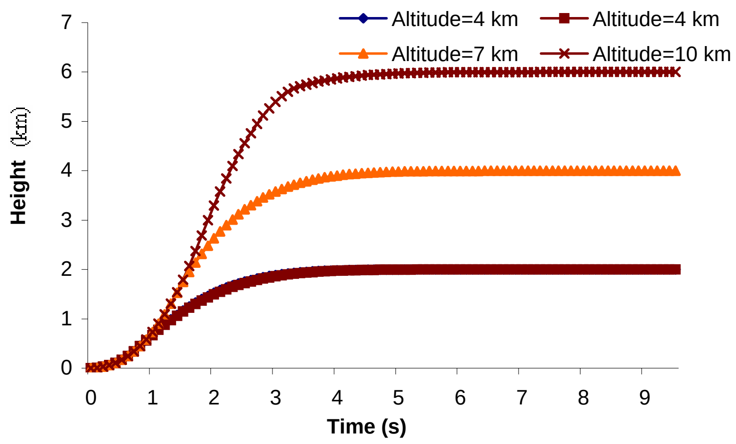


b. RPM behavior.

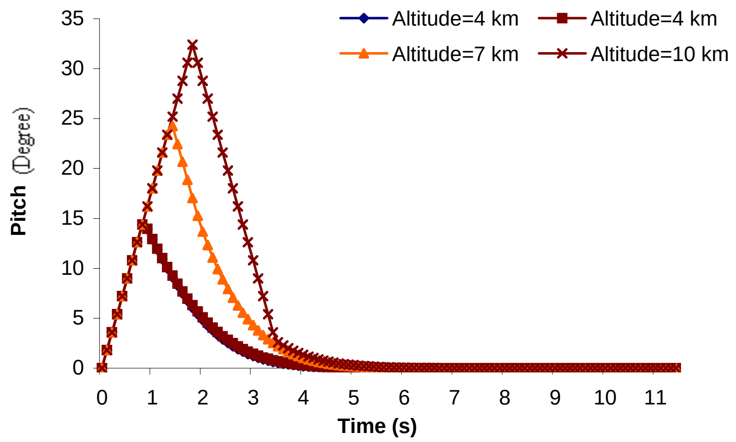


c. Throttle behavior.

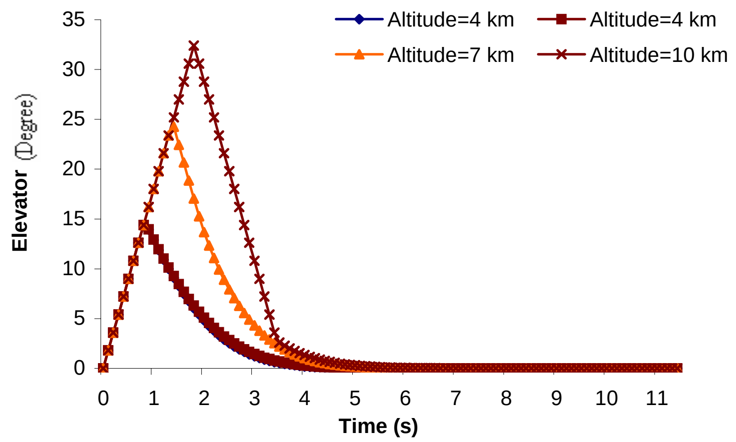
Fig (4.47) The behavior of simulated speed, RPM, and throttle at different altitudes.



a. Height behavior.

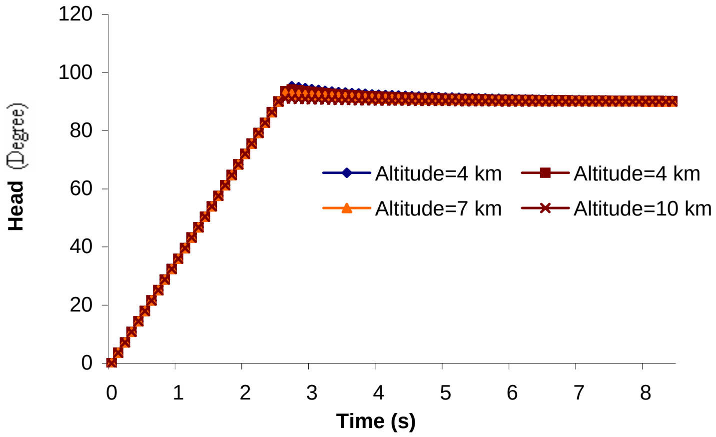


b. Pitch behavior.

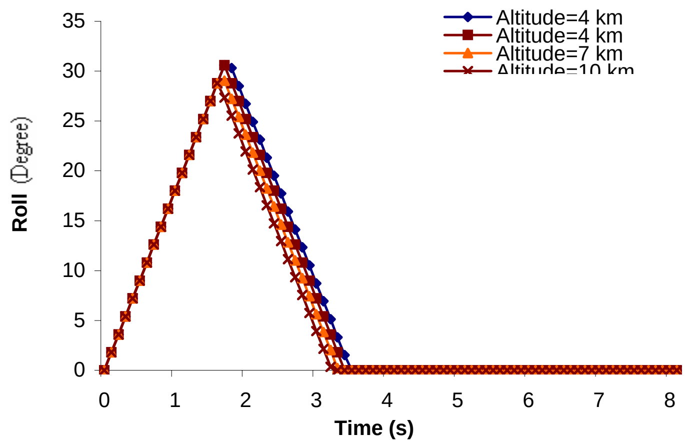


c. Elevator behavior.

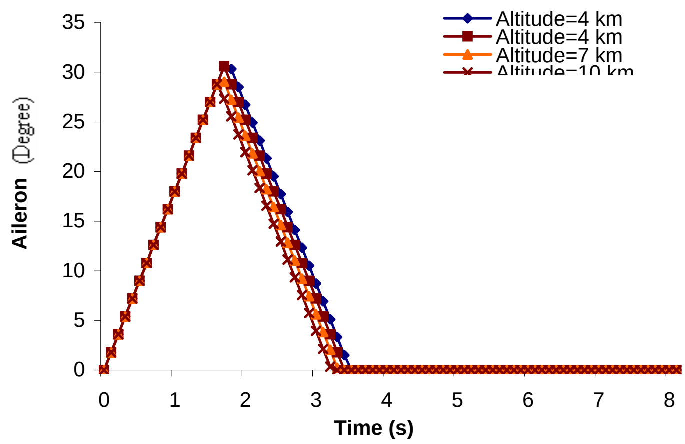
Fig (4.48) The behavior of simulated height, pitch, and elevator at different altitudes.



a. Head behavior.

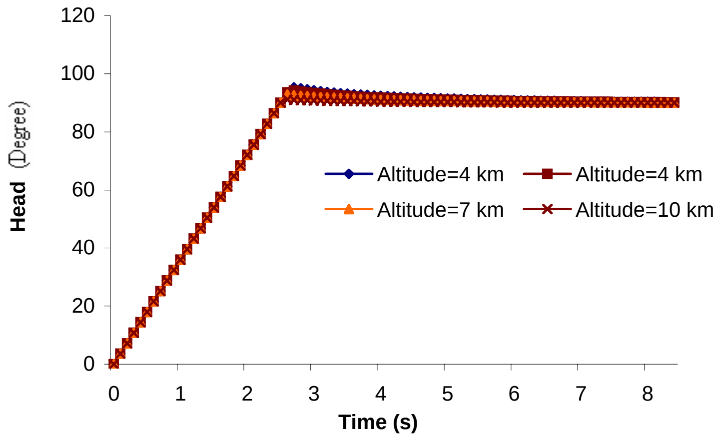


b. Roll behavior.

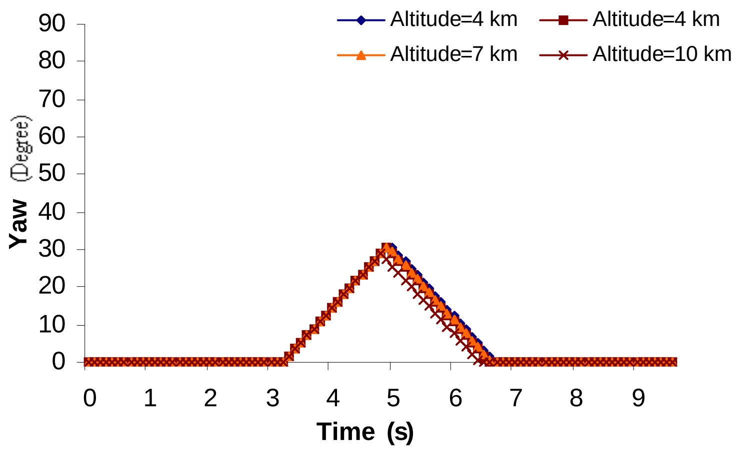


c. Aileron behavior.

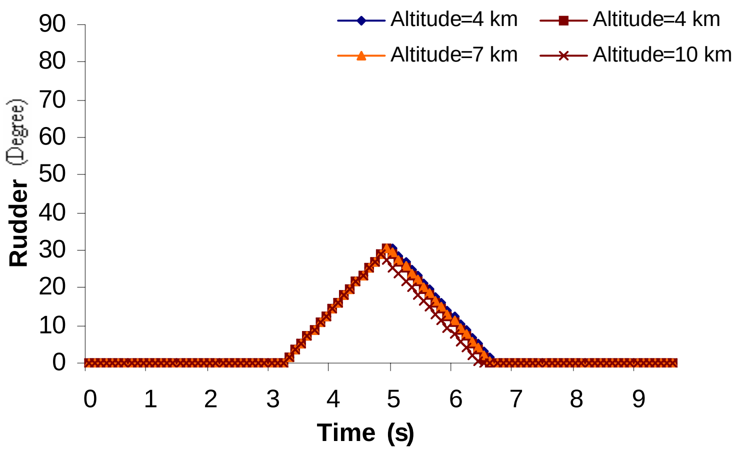
Fig (4.49) The behavior of simulated head, roll, and aileron at different altitudes.



a. Head behavior.



b. Yaw behavior.



c. Rudder behavior.

Fig (4.50) The behavior of simulated head, yaw, and rudder at different altitudes.

For the case of engine dependency on the height, the acceleration will depend on the actual height (as indicated by the simulation results), it shows difference in its highest values when the intended height had been varies. Also, the time of reaching the intended speed is different, and depends on the current intended heights. It was found 6.2 *s*, 7.7 *s*, or 8.2 *s* when the intended height was taken 4 *km*, 7 *km*, or 10 *km*, while it was 5.7 *s* for for the intended height 4 *km* when the effect of the height on the engine performance was ignored. The equal differences between the reaching time periods are due to the equal differences between the intended height and due to the linear relationship that assumed between height and engine performance.

It was noticed that the increasing rate of the height and its related parameters have not much affected by this engine performance relationship. Also, the difference between the height behaviors was proportional to the difference between the occurred and intended heights. It is shown that the pitch had different highest values for different intended height, but all of them were reaching into its normal state at same time. The same time ascribed to the linear change of the speed that reduce the time of climbing.

Moreover, the differences of the power engine due to the altitude had affected the time of reaching the actual speed to its intended value (400 *km/h*). This time became longer at higher altitude, which means the engine little increase the acceleration at higher altitude. In addition, the head related parameters did not much affected by the altitude since they have little dependency on the speed, and the speed reached the intended value at all cases, but after some time delay. Therefore, the behaviors are shifted by a small duration, which may be not significant.

When the intended speed has the maximum possible value at high altitude, the navigator failed to raise the speed, through the throttle, to reach the intended speed due to the limitations imposed on the simulator. The head was little affected due to the small position deflection at low speed. Thus, the change of the acceleration with the altitude had made the speed lower at higher altitude, and by the way, the deflection of head was less too. In such case, the low speed should be avoided because it may cause an airplane stall especially at turning. This case is presented in Figure (4.51), where high intended speed was taken 600 *km/h* at the intended height 8 *km*. The engine power of the considered altitude was taken 60%, therefore the throttle could not raise the speed to reach its intended value. Instead, the throttle had provided an actual speed equal to:

$$Actual\ speed = 600 \times 0.6 = 360\ km/h$$

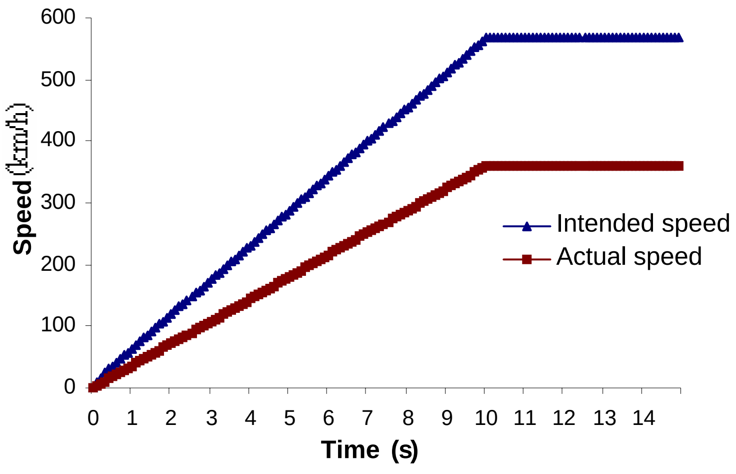


Fig (4.51) The behavior of simulated speed when the actual speed kept under the intended at high altitude.

The proper reaction of the navigator was compensating the difference of the speed by increasing the throttle to give more speed. The maximum amount of the throttle was just 120 *unit*, the actual speed reach to 480 *km/h* which is the higher speed that the engine could provided at that altitude, because the throttle had reached its maximum limit and it could not give any more.

When the simulator was climbing to reach the intended height (10 *km*), and the actual speed was planned to raise from 100 *km/h* to its maximum speed 800 *km/h*, as shown in Figure (4.52), it was found that the navigator tried to increase both the speed and height. The curvature in the second behavior at time less than 3 *s* is due to the non-uniform acceleration of the simulator during the climbing, this irregular behavior was due to the dual change of the height and speed. After the 3 *s*, the height reached its intended value. The fixing of the height made the speed increase is due to the speed deflection only, it is the reason of why making its increase became uniform, and the behavior of the height became fixed. After 6 *s*, the speed reached to the limit that the engine could be provide.

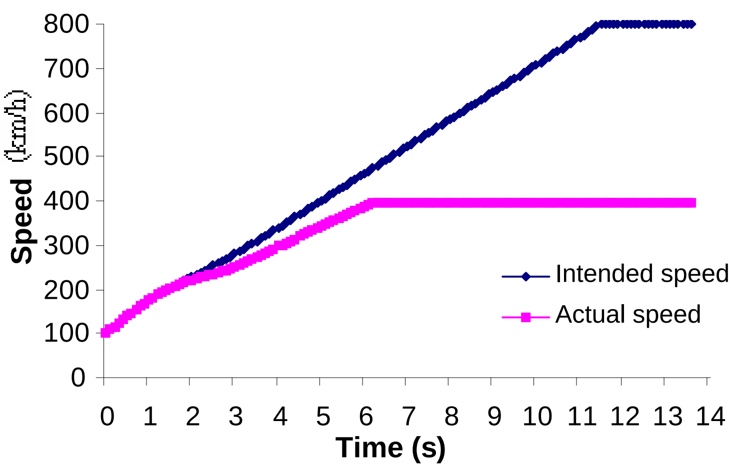


Fig (4.52) The behavior of simulated speed when both the actual speed and height are changed together, in comparison with the case of fixed height.

4.7 Further Analyses

The capability of the simulator in describing the flight situations, and sensing the surrounded conditions of the airplane during the flight encourage to produce more analyses. The analyses includes a qualitative description of the dynamic response on the considered effects of the wind, engine efficiency, and altitude, and then studying the combined effect of them on the navigator performance. In the following, more explanations about the resulting interaction of the adopted simulated effects with each other are presented according to their importance.

4.7.1 Time of Reaching

The time of reaching the simulator to the desired situation is very important for the navigation. In the present model, the time played the most important role since all the parameters are changing with respect to time. The determination of the correction command of due to positioning deflection was depended on the time response t_r, which control the speed of correction. Also, the intended speed assessment depends on the overall flight time and the displacement between the two successive way points.

In addition, the acceleration was depended on the two parameters; *Fraction* and k_{2i}. The *Fraction* is used to determine the rate of change of the flight parameters, while k_{2i} is the ratio factor (proportional factor). Both, have the ability of reducing the time as needed.

4.7.2 Dynamic Response with Wind Effect

It is shown that the wind affects significantly the flight performance. It might speedup or slowdown the flight actions (i.e., it might affect the rate of change of the flight parameters). Figure (4.53) presents how the wind that has the same direction of the flight resists the climbing; see the static handle results of both cases of existence and absence the wind effect. When the dynamic treatment was used the approach to the actual height was faster, rate of change was greater, and the overshot was less.

Therefore, The dynamic response has better capability for correcting the deflection of the flight parameters simultaneously, because its response adapts with the size of the present deflection. Moreover, the dynamic response had reduced the time, the rate of changed the maximum allowed value, and the behavior became better.

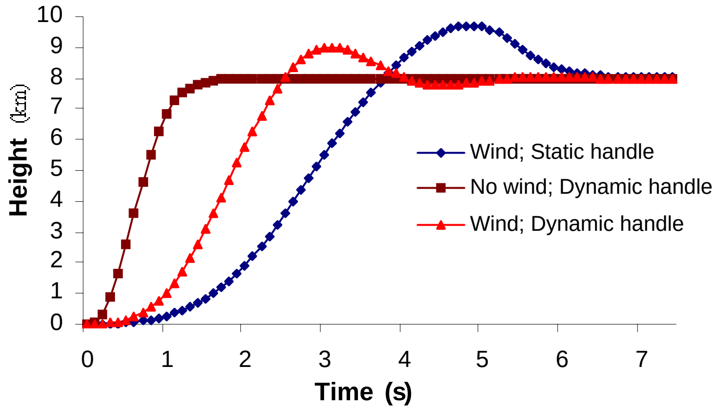


Fig (4.53) The effect of the opposite wind and static/dynamic handle on the behavior of simulated height.

4.7.3 Dynamic Response with Engine Efficiency Effect

The dynamic response is a simultaneous incentive for the mobility of the flight, since the low efficient engine makes acceleration less than that of the efficient engine, and consequently the occurred change of the flight parameters less than the intended change as shown in Figure (4.54). It was noticed the rate of change in the case of the dynamic response model greater than that of the static due to better correction. While, both the static response behaviors possesses an overshot due to the slow correction, which is greater at cases of less efficiency. The dynamic response had avoided the occurrence of both problems (i.e., the decrease of the rate of change and appearance of the overshot), which is clearly shown as a third curve in Figure (4.54).

Therefore, the dynamic response can repair the decrease on flight mobility. The simulation results indicated that the dynamic response model had compensated the fallback of the flight parameters due to the lower engine efficiency and enable the simulator to reach its desired situation with time less than that needed when using static response.

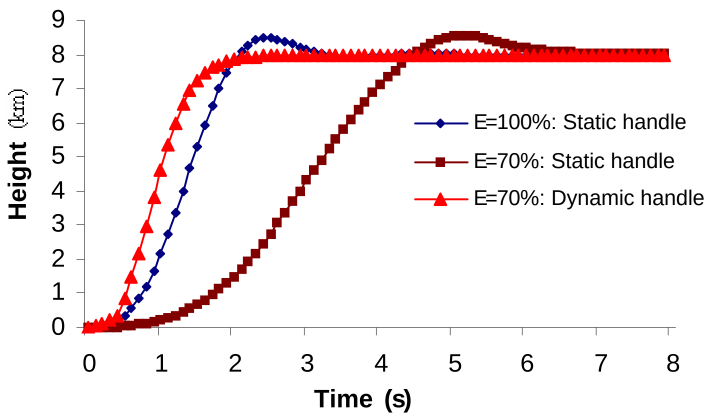


Fig (4.54) The effect of low engine efficiency and static/dynamic handle on the behavior of simulated height.

4.7.4 Dynamic Response with Altitude Effect

It is noticeable that the change of the engine power due to the effect of the altitude led to a response coefficient different from that with the case of no effect.

Figure (4.55) shows three speed behaviors at two fixed altitudes 1*km*, and 10*km*, third was considered when the response was dynamic, while the other when the response was static. It found that the static response model lead to a behavior slowed than that gained by using dynamic model. The decrease of the height rate of change is due to less engine power at higher altitude, which affected the reaching time to be longer. The results due to the dynamic response tell that a high rate of change was ordered by the navigator and led to more acceptable behavior.

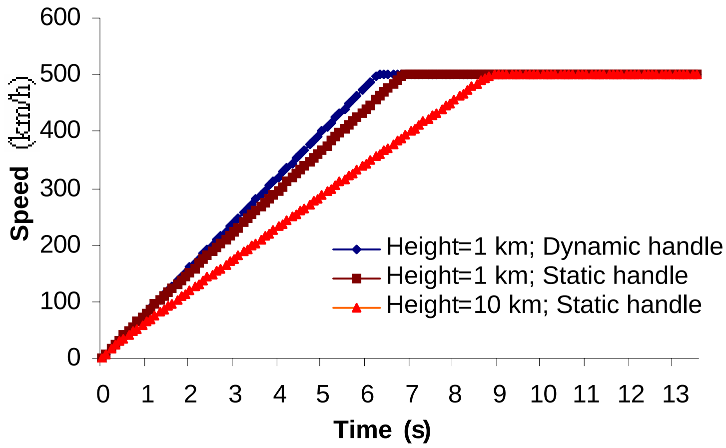


Fig (4.55) The effect of high altitude and static/dynamic handle on the behavior of simulated speed.

4.7.5 Wind Effect and Low Engine Efficiency

It is shown that the wind has a significant effect on the flight; the wind that has opposite direction with respect to the flight increases the drag, while the wind that has same direction push the airplane and increase its speed. The low efficient engine cannot generate the required *RPM* to compensate the effect of the opposite wind, and in such case the airplane cannot reach the desired speed. The simulation results showed that although the throttle increase to largest value the *RPM* was less than the intended and the speed may be less than the intended. This is due to the two combined reasons; the first was the drag effect of the opposite wind and the second was the low *RPM* that supported by the low efficient engine, see Figure (4.56).

When the wind directed in the direction of the flight it would cause an increase in the actual speed. Then, the navigator ordered the throttle to be decreased, and consequently the actual speed had decreased too. The decrease of the throttle is proportional to the difference between the actual and intended speed, so when the wind is very high the *RPM* approached to its minimum amount (i.e., slow *RPM*).

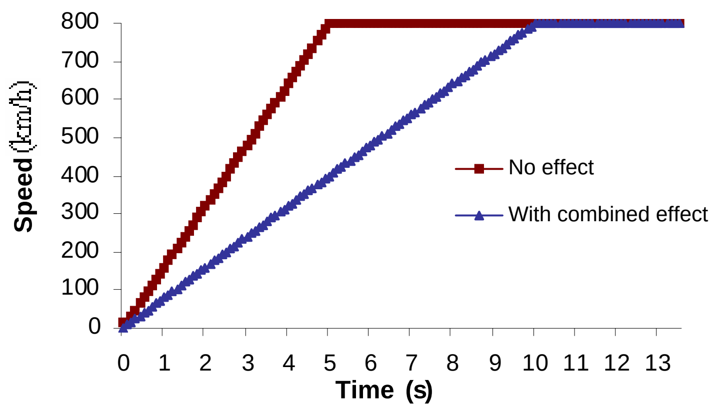


Fig (4.56) The effects of both low engine efficiency and opposite wind on the behavior of simulated speed.

4.7.6 Low Engine Efficiency and High Altitude

The power generated by inefficient engine might not be enough to provide the airplane with the necessary force to attain the desired velocity. When the flight is at high altitude, the engine power will decrease more. When the engine efficiency was assumed 80%, the intended speed 800 *km/h*, and altitude 8 *km* the simulation results indicated that the maximum actual speed did not exceed 480 *km/h*, as shown in Figure (4.57). This low speed made the climbing rate become slow.

Also, the acceleration had passed through two stages; the first was short (it take a time 4.8 *s*), within the time interval of this stage both the combined effects (the low efficiency and altitude) are working together. The interval of the second is between 4.8 *s* and 10 *s*, when the effect of the lower efficiency exists only, and the intended height was reached. After 10 *s*, the acceleration becomes zero because the throttle reached to its maximum value, so it could not cause much more increase to the speed. Therefore, the actual speed kept at a value which is less than the intended.

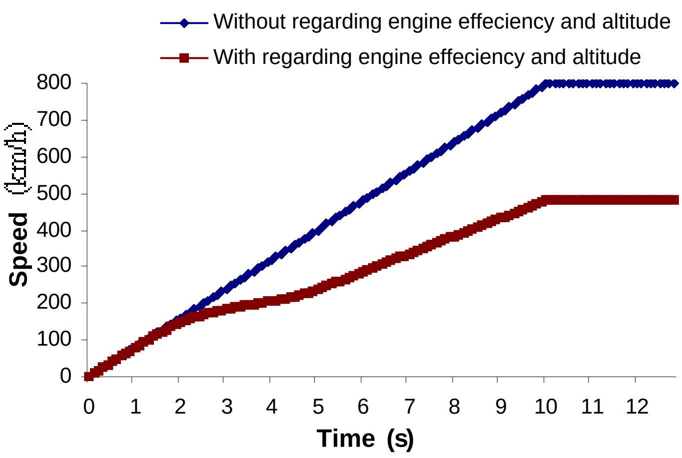


Fig (4.57) The effect of both high altitude and low engine efficiency on the behavior of simulated speed.

CHAPTER ONE

CONTROL SYSTEMS

Chapter Contents

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1.1 Introduction

The guided airplane which is called the Unmanned Aerial Vehicle (UAV) is remotely piloted airplane that can carry cameras, sensors, communication equipments, or other payloads [**Nav, 2005**]. UAV is ordered by its auto-navigation system to proceed to a destination. So, there is no need to have a pilot to drive the airplane around. Instead the airplane is auto-guided to take its vessels to a specific place. The auto-navigation system will try and use all airplane mobility to pass through the shortest possible path. The auto-navigation capability depends upon the contents of the aerial map, which at least should contain an information or database about the weather conditions, allowed routes, and another geographical data. The guidance of the airplane is done by governing the control surfaces of the airplane via commands. The computation of these commands depends on the amount of the occurred error between the actual attitude and the planned attitude during a considered time interval.

A number of UAVs are presently existed. They vary significantly according to their payload weight, carrying capability, their accommodations (volume, environment), their mission profile (altitude, range, duration) and their command, control and data acquisition capabilities. The present research work intends to model and simulate a simple auto-navigation system; this is point out to build a controller drive the airplane through predefined routes, which requires to understanding the basic concepts of the control techniques, which is covered in this chapter.

1.2 Control Systems

Control systems have played an increasingly important role in the development and advancement of modern civilization and technology. Practically every aspect of our day to day activities is affected by some type of control systems. Control systems are found in abundance in all sectors of industry, such as quality control of manufactured products, automatic assembly line, machine-tool control, space technology and weapon systems, computer control, transportation systems, power systems, robotics, and many others. Even the control of inventory and social and economic systems may be approached from the theory of automatic control [**Liu, 1998**].

The basic ingredients of a control system consist of:

1. Control objective.
2. Control system components.
3. Results or output.

The basic relationship between the three components is sequential. In more technical terms, the ***objectives*** can be identified with ***inputs***, or actuating signals and the result (also called ***outputs***), or the controlled variables. In general, the objective of the control system is to control the outputs in some prescribed manner by the inputs through their passage in the elements of the control system [**Sau, 1996**].

In the following a detailed explanation about the two common types of the control systems; open loop and closed loop systems, will be given:

1.2.1 Open-Loop Control System

The elements of an open loop control system can usually be divided into two parts: ***controller*** and the ***controlled process***, as shown in Figure (1.1). An input signal or command (r) applied on the controller, whose output acts as the acting signal (u), the actuating signal then controls the controlled process so that the controlled variable (y) will performed according to the prescribed standards. In simple cases, the controller can be an amplifier, mechanical linkage, filter, or other control element, depending on the nature of the system. In more sophisticated cases, the controller can be a computer such as a microprocessor. Because of the simplicity and economy of open loop control systems, one can find this type of system in many non-critical applications [**Pas, 1998**].

Fig (1.1) Elements of open loop control system

1.2.2 Close-Loop Control Systems

The close-loop control system has a link or feedback from the output to the input of the system. The advantage of the feedback is to provide more accurate and more adaptive control. The controlled signal y is fedback and compared with the reference input, and an actuating signal, proportional to the difference of the input and the output, must be sent through the system to correct the error. The idle-speed control of automobile and steering control of automobile are common examples for the close-loop control system. Figure (1.2) shows the close loop system for one feedback path, it has an input signal (r), output signal (y), error (e), feedback signal (b) and the load torque (T_L) [**Elg, 1998**].

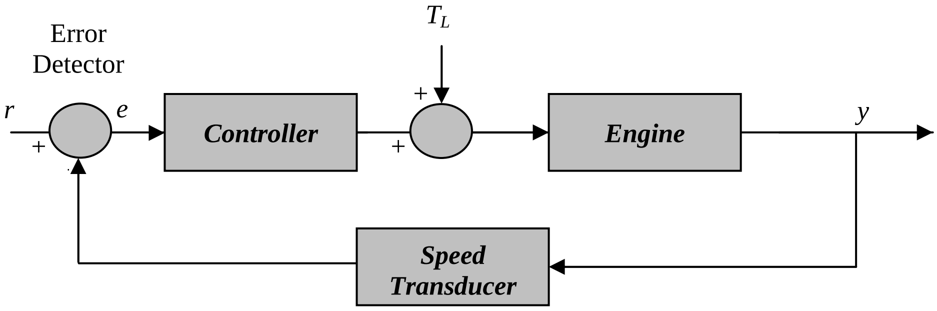


Fig (1.2) Elements of close loop control system

1.2.3 Feedback Effects

The use of feedback is shown to be for the purpose of reducing the error between the reference input and the system output. The significance of the effects of feedback in control systems is more complex than is demonstrated by the following simple example. The reduction of system error is merely one of the many important effects that feedback may have upon a system. The feedback has effects on such system performance characteristics such as overall gain, stability, sensitivity, and disturbance.

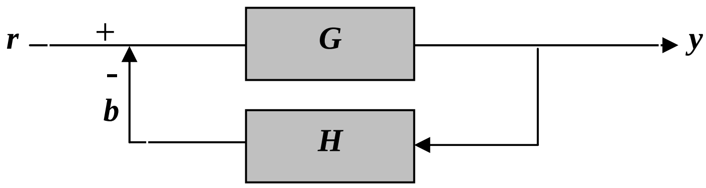


Fig (1.3) Feedback System.

The simple feedback system configuration, shown in Figure (1.3) have two parameters (G) and (H) may considered as constant gains. Therefore, the overall gain (M) of the system is given by the following relation [**Mac, 1989**];

$$M = \frac{y}{r} = \frac{G}{1 + GH} \qquad \dots \ (1.1)$$

Using this basic relationship of the feedback system structure, one can cover some of the significant effects of feedback which are described in the the following sections:

A. Effect on Overall Gain

Feedback affects the gain G of a non-feedback system by factor $1+GH$. The system shown in Figure (1.3) is said to have ***negative feedback***, since a minus sign is assigned to the feedback signal. The quantity GH may itself include minus sign, so the effect of feedback is that it may increase or decrease the gain G. In any practical control system, G and H are functions of frequency, so the magnitude of $1+GH$ may be greater than 1 in one frequency range but less than that in another [**Fai, 1998**].

B. Effect on Stability

Stability is notion that describes whether the system will be able to follow the input command (or be useful). In a nonrigorous manner, a system is said to be unstable if its output is out of control. To investigate the effect of feedback on stability one can refer to equation (1.1), if GH=-1 the output of the system is infinite for any finite input and the system is said to be unstable. Therefore, the feedback can cause a system that is originally stable to become unstable. It can be demonstrated that one of the advantages of incorporating feedback is that it can stabilize an unstable system. When the feedback system shown in Figure (1.3) is unstable due to GH=-1, then this situation is overcame by introducing another feedback loop through negative feedback gain of F, as shown in Figure (1.4), the input-output relation of the overall system will be

$$\frac{y}{r} = \frac{G}{1 + GH + GF} \qquad \dots \; (1.2)$$

It is apparent that although the properties of G and H are such that the inner-loop feedback system is unstable, because GH=-1, the overall system can be stabled by proper selection of the outer-loop feedback gain F [**Pad, 1991**].

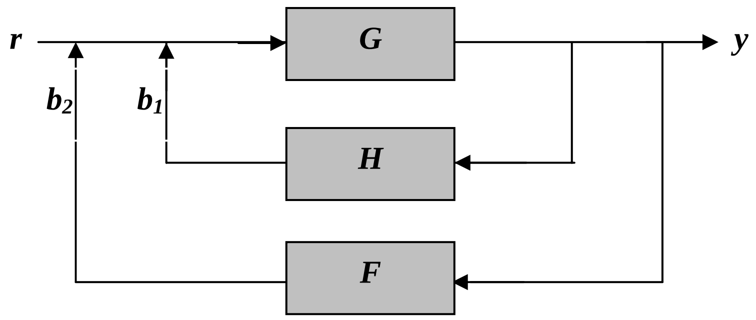


Fig (1.4) Feedback system with two feeding loops.

C. Effect on Sensitivity

Sensitivity considerations are important in the design of control systems. Since all physical elements have properties that change with environment and age, one cannot consider the parameters of a control system to be completely stationary over the entire operating life of the system [**Elg, 1998**].

In general, a good control system should be very insensitive to parameter variations but sensitive to the input commands. By referring to the system shown in Figure (1.3) G is the gain parameter that may vary, the sensitivity of the gain of the overall system M, to the variation in G is defined as

$$S_G^M = \frac{Percentage\ change\ in\ M}{Percentage\ change\ in\ G} = \frac{\delta M\ /\ M}{\delta G\ /\ G} \qquad \ldots\ (1.3)$$

where, δM denotes the incremental change in M due to the incremental change in δG. By using equation (1.1) the function S_G^M is rewritten,

$$S_G^M = \frac{\delta M}{\delta G}\frac{G}{M} = \frac{1}{1+GH} \qquad \ldots\ (1.4)$$

this relation shows that if GH is a positive constant, the magnitude of the sensitivity function can be made arbitrarily small by increasing GH. Provided that the system remains stable. It is apparent that in an open-loop system, the system will respond in a one-to-one fashion to the variation in G (i.e., $S_G^M = 1$). Practically, GH is a function of frequency, the magnitude of (1+GH) may be sensitive to parameter variations in certain cases [**Dro, 1989**].

D. Effect on Disturbance

In the design of a control system, considerations should be given so that the system is insensitive to the disturbances (noise) and sensitive to input commands.

The effect of feedback on the disturbance depends greatly on where these extraneous signals occur in the system. In many situations, feedback can reduce the effect of disturbance on system performance. System presented in Figure (1.5) has command signal (r) and noise signal (n). In the absence of feedback, H=0, the output (y) due to (n) acting alone is

$$y = G_2 n \qquad \dots \ (1.5)$$

With the presence of feedback the system output due to (n) acting alone is

$$y = \frac{G_2}{1 + G_1 G_2 H} \times n \qquad \dots \ (1.6)$$

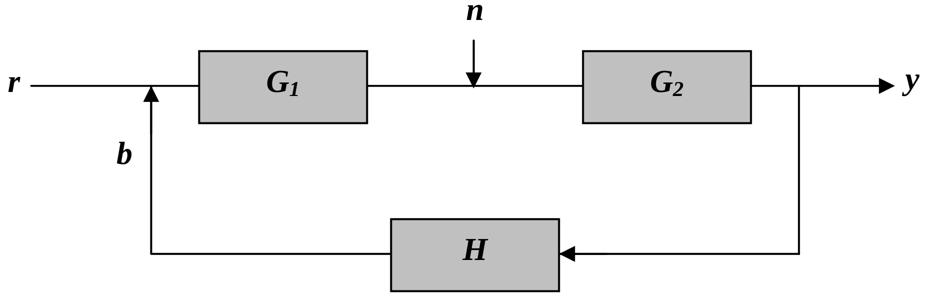


Fig (1.5) Feedback system with a noise signal.

A comparison of equation (1.6) with equation (1.5) shows that the noise component in the output of equation (1.6) is reduced by the factor $1+G_1G_2H$, if the latter is greater than unity and the system is kept stable [**Kuo, 1987**].

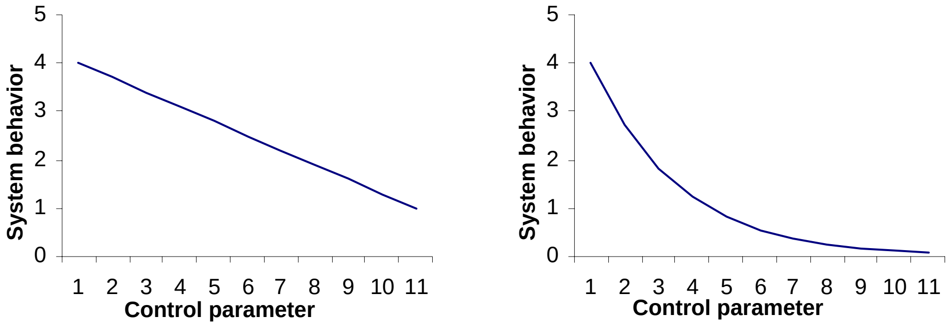
1.2.4 Feedback Types

Feedback control systems may be classified into different ways, depending on the purpose of the classification. According to the method of analysis and design, control systems are classified as ***linear*** and ***nonlinear***, ***time-varying*** or ***time-invariant***. According to the types of signal used in the system, reference is often made to ***continuous-data*** and ***discrete-data*** system, or modulated and unmodulated systems. Control systems are often classified according to the main purpose of the system. For instance, a position-control system and a velocity-control system control the output variables according to the way the names imply [**Phi, 2000**]. For example, the auto-navigation system exhibits the linear system, time-variant, and route-velocity control system. In the following some details are offered.

A. Linear and Nonlinear Systems

This classification is made to the methods of analysis and design. Linear systems do not exist in practice, since all physical systems are nonlinear to some extent. Linear feedback control systems are identized models fabricated by the analyst purely for simplicity of analysis and design. When the magnitudes of signals in a control system are limited to ranges in which system components exhibit linear characteristics then the system could be essentially considered linear, as shown in Figure (1.6-a). But when the magnitudes of signals are extended beyond the range of the linear operation the system should no longer be considered linear [**Gla, 1996**].

For linear system, there exist a wealth of analytical and graphical techniques for design and analysis purposes. Nonlinear systems are usually difficult to treat mathematically and there is no general methods available for solving a wide class of nonlinear systems as shown in Figure (1.6-b) [**Gla, 1996**].



(a) Linear system **(b) Non-linear system**

Fig (1.6) Linear and non-linear system behaviors.

In the design stage of control systems, it is practical first to design the controller by using the linear model (i.e., neglecting the nonlinearities of the system). Then, the designed controller is applied on the nonlinear system model to assess the required enhancements to improve the proposed system model [**Fai, 1998**].

B. Time-Invariant and Time-Variant Systems

When the parameters of a control system are stationary (with respect to time) during the operation of the system, the system is called a time-invariant system. Practically, most of the physical systems contain elements that drift or vary with time. For example, a guided-missile control system in which the mass of the missile decreases as the fuel on board is being consumed during flight. Also the auto-navigation system has a route and velocity, both are varying with time [**Gop, 1988**].

The analysis and design of time varying class of systems (such as unmanned aerial vehicle) are usually much more complex than that of the linear time-invariant system (like, sun tracking control system) [**Kuo, 1987**].

C. Continuous and Discrete Data System

A continuous data control system is one in which the signals at various parts of the system are all functions of the continuous time variable (t). The electrical signals may be further classified as *ac* or *dc*. The two types of signals (*ac* and *dc*) carry special significance in control systems terminology. When one refers to an *ac* control system, it is usually means that the signals in the system are ***modulated*** by some form of modulation scheme. On the other hand, when a *dc* control system is referred to, it does not mean all the signals in the system are unidirectional, then there would be no corrective control movement. A *dc* control system simply implies that the signals are ***unmodulated***, but they are still ac signals according to the conventional definition [**Rav, 1978**].

Discrete data control systems differ from continuous data systems in that the signal at one or more points of the system are in the form of either a pulse train or a digital code. Usually, discrete data control system are subdivided into ***sampled-data*** and ***digital control systems***. Sampled-data control systems refer to a more general class of discrete-data system in which the signals are in the form of pulse data. A digital control system refers to the use of a digital computer or controller in the system, so that the signals are digitally coded (such as in binary code) [**Row, 1986**].

Because digital computers provide many advantages in size and flexibility, computer control has become increasingly popular in recent years. Many airborne systems contain digital controllers that can be treated with thousands of discrete data per instant of time. Figure (1.7) shows the basic guidance close-loop control system of the guided airplane [**Ili, 1996**].

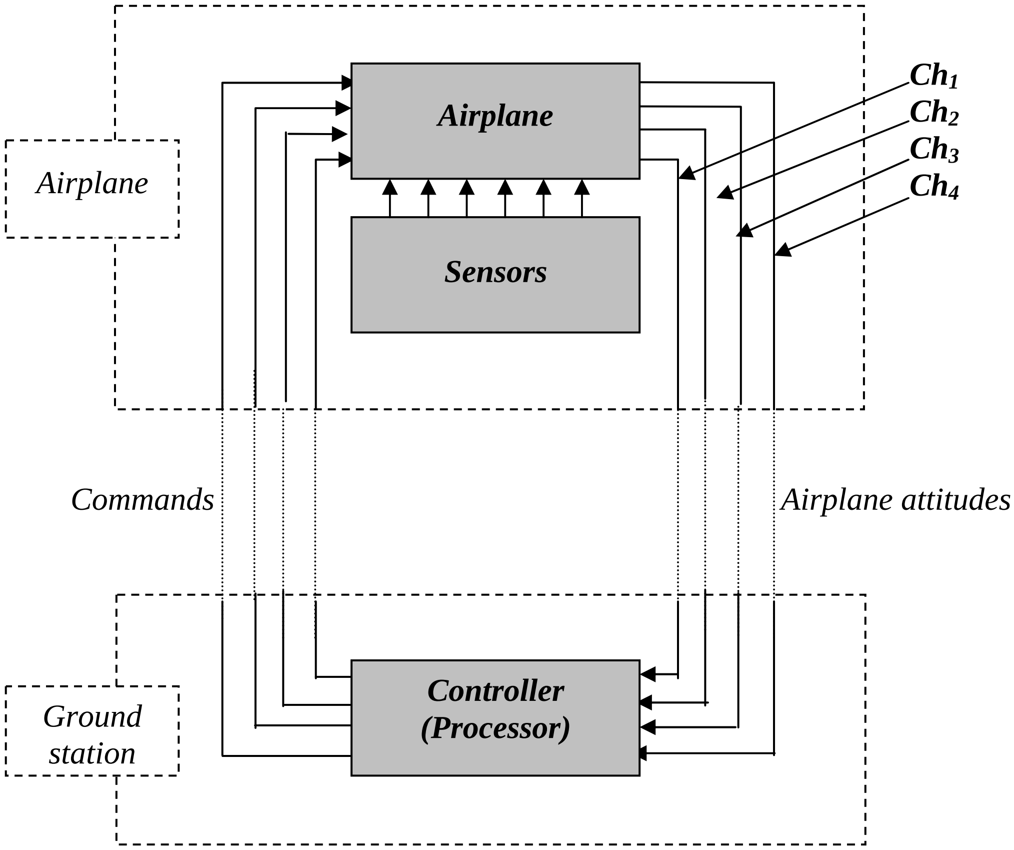


Fig (1.7) Digital auto-pilot system for a guided airplane.

It shows the basic elements of the digital auto-pilot for guidance airplane control, the controller on the ground station (or may on the airplane) feedback the actual data from the airplane's sensors by means of four specific channels, the four channels represents the degree of freedom of the airplane motion. At each controlling interval, the controller determines the occurred error and tries to correct the airplane situations by exporting correction commands [**Kuo, 1987**].

1.3 Commands

The close loop technique is the best controlling approach for the components of the control system. The controlling process is done by sending correction commands respectively. These commands are the ways by which can govern the components toward the intended conditions. Sending the command requires determining the occurred error of the system components, and then the direction of the correction commands will be opposite to the determined errors. The magnitude of the command should be proportional with the magnitude of the occurred error. That means, when the error is small, the command influence will be small, while when the command influence is large, the error should also be large. The influence is the response of the system to the correction command, where this response depends on the type of the exported command [**Ili, 1996**]. There are three command models used in the control processing, these are:

1.3.1 Linear Proportional Command Model

The command in the linear command model (*LCM*) is linearly proportional with the occurred error, according to the following relation:

$$C = k\ e \qquad \dots\ (1.7)$$

where C is the command, e is the error, and k is the proportional factor.

Figure (1.8) shows the linear relationship of the command as a function of error, where the proportional factor k of the command-error relationship is the slope of the command behavior [**Veg, 1996**]. Therefore, if the occurred error is equal to (-2) then the determined command will be equal to (+2) for k=1, and it is equal to (+4) for k=2, and so on. It is clearly that k is a magnification factor for the influence value of the command [**Ili, 1996**].

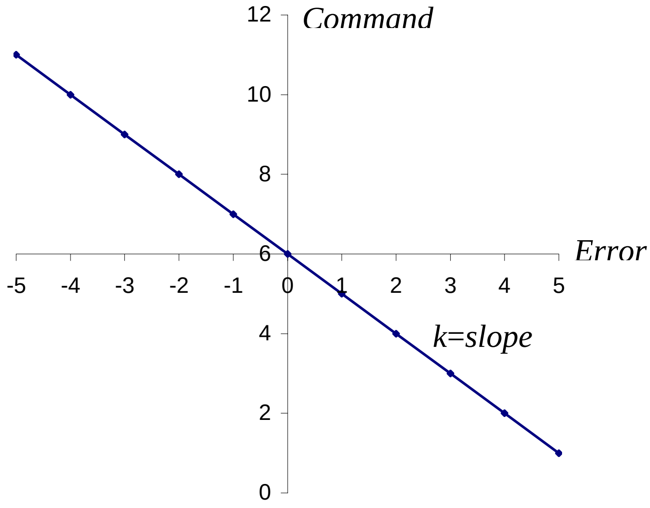


Fig (1.8) Command versus error

Small value of *k* makes the response of the system slow, so it may be not useful because of the time delay. Large value of *k* makes the response of the system high. The large values of *k* may cause transfer of the error to the opposite side of the desired value, this may generate an oscillation in the control parameters value. A proper value of *k* must be tested and chosen to give more stability for the control system with good response [**Gla, 1996**].

1.3.2 Differential Command Model

The equation of the differential model (*DCM*) contains two terms within. The first term is the linear model equation, while the second term is the derivative of the error with respect to time [**Nag, 1986**], as follows

$$C_i = ke_i + k' \frac{\Delta e_i}{\Delta t} \qquad \dots \ (1.8)$$

It is natural to expect that the added differential term will sharpen the behavior of the linear command. The sharpening is the process of making the additive term (derivative term) is equal to relatively small value, i.e. approach to zero, for all the sequenced commands that seem to be nearly equal, or smoothed value, while it is relatively large value for the sequenced commands with high differences in between. The most commonly used method of the differentiation in the command formatting is the gradient. The gradient is approximated by differences of the sequential two commands, as follow

$$C_i = ke_i + k'(e_i - e_{i-1}) \qquad \ldots\ (1.9)$$

This formulation is more desirable for a computer implementation of the gradient, because it is easier to program if the speed of computation is an essential requirement. The approximation, given above, for the value of the gradient in the command equation is proportional to the difference between two successive determined errors. Thus the gradient assumes relatively large for prominent of errors, and small for an error that are fairly smooth or being zero only [**Dro, 1989**].

1.3.3 Differential-Integral Command Model

The differential integral model (*DIM*) is suggested in order to avoid the effects of the large differences of the derivative term in the command equation. The additive third term represents the integration of the occurred error with respect to time. Mathematically, the integration has an opposite action of the derivative, so that one expects its affect oppositely compared with the derivative effect. Where the integration term will cause smoothing in the behavior of the command, especially at high differences between the successive commands [**Veg, 1986**].

Because the behavior of the command yields through discrete intervals, the integration term can be expressed by the equivalent mathematical form, which is the summation. Therefore, the equation of *DIM* will be

$$C_i = ke_i + k'\frac{\Delta e_i}{\Delta t} + k''\int e_i\, dt \qquad \ldots\ (1.10)$$

or,

$$C_i = ke_i + k'\frac{\Delta e_i}{\Delta t} + k''\sum e_i \Delta t \qquad \ldots\ (1.11)$$

or, it can be written in the following practical form

$$C_i = ke_i + k^{'}(e_i - e_{i-1}) + k^{''}(e_i + e_{i-1}) \qquad \ldots (1.12)$$

where, k, $k^{'}$,*and* $k^{''}$ are the proportional factors.

1.4 Historical Review

The practical application of the airplane guidance appeared at the beginning of the 1950's [**Nav, 2005**]. In 1950s the UAVs have been used in a reconnaissance and intelligence-gathering role, and more challenging roles were envisioned, including combat missions. In 1960s the Department of Defense (DOD) of the United States (U.S.) had developed 11 different UAVs, due to acquisition and development problems only 3 entered production. In 1970s the National Advisory Committee for Aeronautics (NACA) sponsored a national project with participation by the DOD to introduce high altitude long endurance remotely operated aircraft to routine flight in the National Airspace System (NAS). The produced UAV capable to provide imagery intelligence for the commanders on land, and see at range out to 85 km with a height of 1500 ft and flight duration of 1 hour. In 1980s Army Science Board developed UAV including high speed target drones and multi purpose aerial surveillance aircraft. Pioneer UAV in 1985 is designed to support the commanders with near-real-time imagery intelligence range up to 200 km, height of about 3000 ft and endurance 1 hour [**Unm, 2005**].

By the early of 1990s the UAVs testify a great technological developments. DOD sought UAV can satisfy surveillance requirements in close range, short range, or endurance categories. Close range was defined to be within 50km, short range was defined as within 200 km, and endurance as anything beyond. In the late of 1990s, the close range and short range categories were combined [**Hun, 2006**].

After the second millennium, the current employed categories of UAV are the endurance (EUAV) and the tactical (TUAV). Tactical category is ubiquitous with different purposes, it is optionally piloted vehicle and capable of manned or unmanned flight [**Gui, 2005**].

In 2003, the Tactical and Joint Tactical (Hunter) UAVs are designed and developed to provide ground and maritime forces with near-real-time imagery intelligence at range up to 200 km; extensible to 300 km by using another Hunter UAV as an airborne relay [**Hun, 2006**]. In 2004, the medium altitude endurance (MAE) UAV is designed by the U.S. Air Force to provide imagery intelligence at a range 500 nautical mile, while the altitude reaches to 30000 ft. In 2005, the Advanced Concept Technology Demonstration (ACTD) contributed with National Aeronautics and Space Administration (NASA) to design the high altitude endurance (HAE) UAV that known Global Hawk, which is intended for missions requiring long-range deployment and wide-area surveillance or long sensor dwell over the target area, it is reaches to a height of 60000 ft [**Uav, 2005**].

In 2006, U.S. Navy is still utilizing the Tactical Control Station (TCS) which is a software and communications links required to control the TUAV and MAE-UAV and other future tactical UAVs. TCS is prepared to produce the Micro UAV (MAV) by the U.S. Navy that used to explore the military separation and the future military operations, and to develop and demonstrate flight enabling technologies for very small aircraft (less than $15\ cm$ in any dimension) [**Mav, 2006**]. According to a statistics published in 2006, there are about 50 U.S. companies, academic institutions, and government organizations developing over 150 UAV designs. Forty of these companies have succeeded to fly own designs. Fifteen of these companies have 26 models of UAVs in, or ready for, production [**Nas, 2006**].

1.5 Aim of the Thesis

Because of the strategic significance of this science, the industrial techniques have driven some companies to conquer both marketing and knowledge in this field. Therefore, present thesis aims at modeling and design a dynamic navigator, and then examine the proposed navigator by the simulation technique at different flight situations.

1.6 Work Objectives

The followings are the main objectives of the present research.

1. Analyze, model, and design simple dynamic auto-navigation system .
2. Simulate the auto-navigation system; in this stage a flight simulator is designed to analyze performance of the navigator and response of the airplane, it will be helpful to reflect a synthesized airplane response.
3. Develop a dynamic response model, and adopt it in the simulation process rather than the static one.
4. Analyze, model, and simulate the effect of the wind, engine efficiency, and altitude.
5. Discuss and analyze the results of each step of the above points.

CHAPTER THREE

PROPOSED SIMPLE DYNAMIC NAVIGATION MODEL

Chapter Contents

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PROPOSED SIMPLE DYNAMIC NAVIGATION MODEL

3.1 Preamble

The proposed auto-navigation model based on the concept of the action-reaction, it aims at controlling the airplane during the flight. The idea is implemented by introducing a specific action according to accounted determinations, and then the expected reaction will navigate the airplane toward the desired situation. The control of the airplane mobility is described as in the followings paragraphs;

If the speed of the airplane is intended to be increased/decreased, then the proper action is increasing/decreasing the throttle, which leads to increase/decrease in thrust, the reaction will speed up/slow down the forward speed of the airplane. Also, if the airplane is intended to be climbed/descended, then the action is deflecting the elevator by an angle proportional to the direction and amount of the desired change, then the reaction will be the generation of a torque arise/descend the tail of the airplane (i.e., pitch angle has a value does not equal to zero) and then the wind lifts the airplane upward/downward. Whereas if the head of the airplane is intended to be changed, then the action is deflecting the aileron/rudder by a value is proportional to the amount and direction of the change, and the reaction is generating a torque rotate the airplane about its longitudinal/vertical axis which causes a change in its head.

The analysis in the above is employed to model the auto-navigation process as a first phase. Next phases will be design, simulate, and response development of the suggested navigator, then studying the effects of wind, engine efficiency, and altitude to examine the efficiency of the navigator performance.

3.2 Auto-Navigation System Modeling

The auto-navigation is a control process determines the error occurs in the airplane flight from the planned path at each control interval, these errors are called ***deflections***. Then, the determined correction commands based on these deflections are exported to the control surfaces to guide the airplane toward the desired situations. These commands have values equal to the deflections but in opposite direction. The auto-navigation process continues navigating the airplane through the intended route and desired situations, where the amount of deflection is reduced time by time. Figure (3.1) presents the block diagram for the main operations of auto-navigation algorithm.

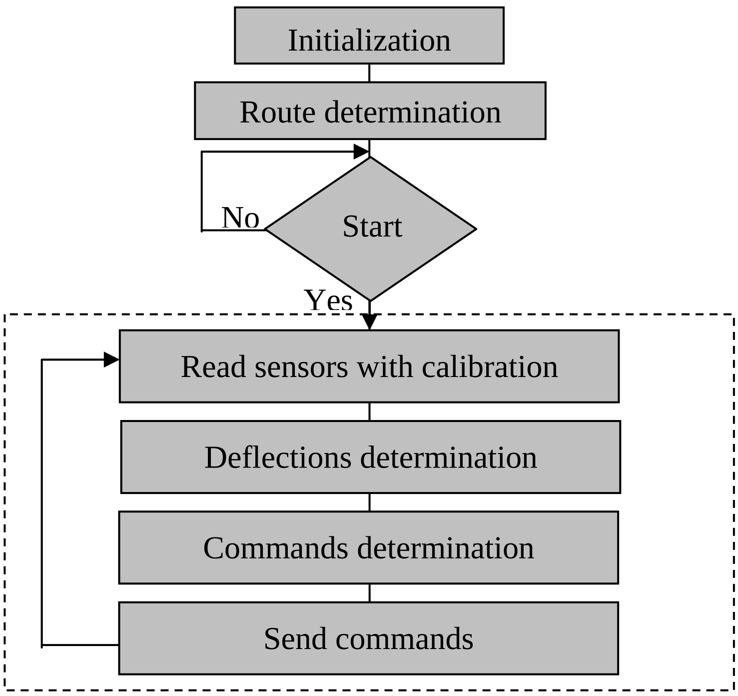


Fig (3.1) Block diagram of the auto-navigation operations.

Many terminologies will be illustrated in following sections to classify and describe the parameters included in the proposed simple model of the auto-navigation according to the strategy of Figure (3.1). An explanation to the proposed model is given, and how its parameters are related is analyzed.

3.2.1 Navigation Parameters

During the flight, the airplane swims through a space of three dimensions. The instantaneous position of the airplane is determined by the ellipsoidal coordinate system (ECS; *longitude*, *latitude*, and *height*) via specific sensors. The positioning coordinates besides both speed and head of the airplane represent the navigation parameters by which the airplane's flight attitude could be determined during the flight.

The geometrical description of the ECS is very difficult to use because its imaginary lines along the Earth's surface are curved. This difficulty can be exceeded by adopting a simplified linear approximation rather than EMS, which is applicable to short flight distances:

$$\begin{bmatrix} X \\ Y \\ Z \end{bmatrix} = \begin{bmatrix} 91 & 0 & 0 \\ 0 & 112 & 0 \\ 0 & 0 & 1 \end{bmatrix} \begin{bmatrix} Lon - Lon_o \\ Lat - Lat_o \\ Height \end{bmatrix} \qquad \dots (3.1)$$

where Lon_o and Lat_o represent the longitude and latitude of the reference point which can be chosen as a mid point within the planned flight path. This transformation is an approximation, and it is valid only for a small geographical flight region (such as Iraq), appendix (A) contains the mathematical details of this transformation. The 3-D space that determine the position of the airplane is divided into; the vertical axis Z which determine the height of the airplane with respect to the sea level, and the two planatic axes X and Y that determine the head of the airplane on the surface of the Earth with respect to the reference point.

In order to model the navigation parameters and its relationship and effect on the airplane flight, it is needed to understand how the airplane behaves during the flight. A flying airplane assumed to be moved by velocity V with respect to the coordinates X, Y, and Z as shown in Figure

(3.2), note that the positive *Y*-axis refers to the geographical north and the positive *X*-axis refers to the geographical east.

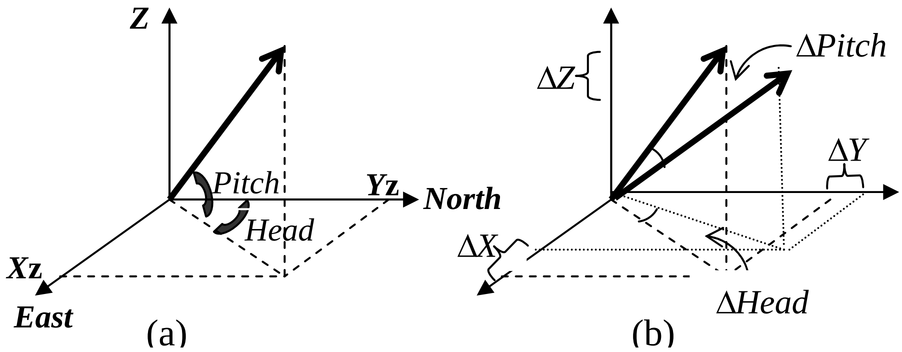


Fig (3.2) The three positioning coordinates of the airplane

The instantaneous components of the airplane's velocity could be described by the spherical coordinates as follows

$$V_x \approx \frac{\Delta x}{\Delta t} = V\ Cos(Pitch)\ Sin(Head)$$
$$V_y \approx \frac{\Delta y}{\Delta t} = V\ Cos(Pitch)\ Cos(Head) \qquad \ldots (3.2)$$
$$V_z \approx \frac{\Delta z}{\Delta t} = V\ Sin(Pitch)$$

where Δx, Δy, and Δz are the positioning deflections during one short time interval Δt, *Pitch* and *Head* are the pitch and head of the airplane. Equations (3.2) can utilized to determine the positioning deflection due to the airplane motion, which leads to determine the deflections of both height and head of the airplane as follows;

$$\Delta Height = \Delta z \qquad \ldots (3.3)$$

$$\Delta Head = \tan^{-1}(\frac{\Delta x}{\Delta y}) \qquad \ldots (3.4)$$

where, $\Delta Height$ and $\Delta Head$ are the deflections of the height and head.

3.2.2 Flight Parameters

Flight parameters describe the flight status of the airplane and its relation with the surrounding, they include the angular coordinates (pitch, roll, and yaw) and the revolution per minute of the airplane's engine.

The three angular coordinates describe the inclination of the airplane with respect to the planes pass through the center of mass of the airplane as shown in Figure (3.3). They are; ***Pitch*** angle which is the angle between the airplane longitudinal axis and the *X* axis, the change in this angle may lead to change in the height, ***Roll*** and ***Yaw*** angles are the angles of other two transverse axis of the airplane about the *Y* and *Z* axis, respectively, both of them may lead to a change in head angle, and the fourth flight parameter is ***RPM***, supports the thrust that affects the speed of the airplane.

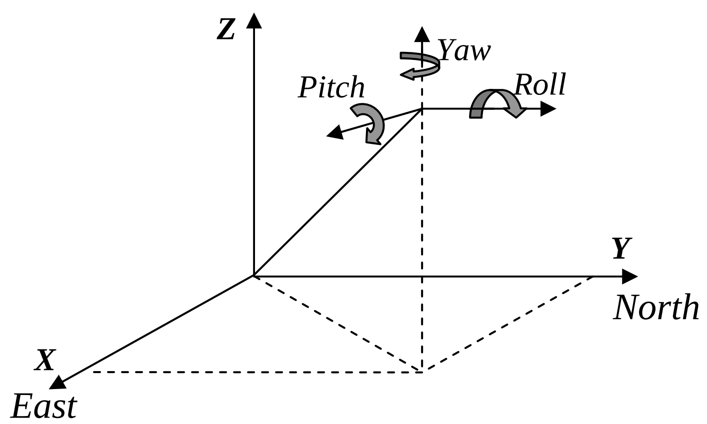


Fig (3.3) Flight parameters.

Since each flight parameter is related by a corresponding navigation parameter, any change may occur in the flight parameter will change the navigation parameters too, i.e.

$$\Delta F_i \propto \Delta N_i$$

One can use this relationship to govern the navigation parameters (i.e., airplane flight status), therefore the adoption of the linear model will lead to the following formula:

$$\Delta F_i = k_{1i} \Delta N_i \qquad \ldots (3.5)$$

where ΔF_i and ΔN_i are the deflections of the flight and navigation parameters, k_{1i} is the proportional factor, and i is an index takes the values (1 to 4), it specifies the flight and navigation parameters according to table (3.1).

Table (3.1) The correspondence of flight and navigation parameters.

i	**Flight parameters**	**Navigation parameters**
1	*RPM*	*Speed*
2	*Pitch*	*Height*
3	*Roll*	*Head* (for large changes)
4	*Yaw*	*Head* (for small changes)

The proportional factor k_{1i} is an incentive by which the navigation parameters affect the flight parameters. When k_{1i} is set 1, then the flight parameters are assumed to be changed by the same amount of change of the navigation parameters. While, when k_{1i} is set 2, then the changes in the flight parameters are assumed to be double of the changes in the navigation parameters, and so on. Therefore, it is necessary for k_{1i} to be limited by a specific range (i.e., $k_{1i,min} \leq k_{1i} \leq k_{1i,max}$) to provide a balance between the changes of the navigation and flight parameters. During the flight, k_{1i} is determined according to the current situation that the airplane undergo, for example $k_{1i} = k_{1i,min}$ when small changes need to be produced in the navigation parameters, and it is set $k_{1i} = k_{1i,max}$ when large changes need to be done.

3.2.3 Control Parameters

In addition to ***throttle***, the control surfaces; ***elevator***, ***aileron***, and ***rudder*** are regarded as the control parameters. The throttle is responsible on changing the *RPM* of the engine which in turn leads to change in the airplane speed, while the control surfaces are responsible about the change in the inclination status of the airplane with respect to its coordinates centered at the center of mass of the airplane, their change leads to change in the height and head of the airplane.

The change of the control parameters causes specific changes in the flight parameters. Whereas, each flight parameter has a direct effect on the corresponding control parameter:

$$\Delta C_i \propto \Delta F_i$$

Therefore, the control surfaces can guide the airplane to specific flight states (by changing its flight parameters) according to the linear proportional command model as follows:

$$\Delta C_i = k_{2i} \Delta F_i \qquad \dots (3.6)$$

where ΔC_i and ΔF_i are the deflections of control and flight parameters, k_{2i} is the proportional factor, and i is an index takes the values (1 to 4), it specifies the control and flight parameters according to table (3.2).

Table (3.2) The correspondence of control and flight parameters.

i	**Control parameters**	**Flight parameters**
1	*Throttle*	*RPM*
2	*Elevator*	*Pitch*
3	*Aileron*	*Roll*
4	*Rudder*	*Yaw*

3.2.4 Deflections

Deflections are the errors occur in the navigation and flight parameters. The deflection of the navigation parameters means that the airplane does not move within the planned conditions due to these deflection the control command are issued to set the flight parameters in such a way to suitably correct the navigation parameters.

To determine the deflection value, two parameters must be taken into account; the first is planned value, sometimes called the ***intended value*** (*IntValue*), and the latter is the ***actual value*** (*ActValue*). The difference between them is considered as the deflection ($\Delta Value$) of that parameter. So, the deflection of any parameter is then written by the following general formula.

$$\Delta Value = IntValue - ActValue \qquad \dots (3.7)$$

The actual value is determined by a specific sensor put on the airplane, while the intended value represents the planned value which is determined by fed the controller by using the set of control (or reference) navigation before beginning its task. It should be mentioned that the positioning deflection is modeled by special suggested geometrical method, which is mentioned in specific section “route deflection”.

A. Parameter Deflection

The deflection of the navigation parameters can be described according to equation (3.7), as follows;

$$\Delta N_i = IntN_i - ActN_i \qquad \text{… (3.8)}$$

where, ΔN_i is the deflection in the i^{th} navigation parameter, $IntN_i$ and $ActN_i$ are the intended and actual values of the i^{th} navigation parameter.

According to equation (3.8), the intended values of the flight parameters are deflect from their actual values by an amount proportional to the deflections occur in the navigation parameters as noted.

$$IntF_i \propto \Delta N_i$$

$$IntF_i = k_{3i} \Delta N_i \qquad \text{… (3.9)}$$

where $IntF_i$ is the intended values of the i^{th} flight parameter, and k_3^i is the proportional factor of the i^{th} flight parameters.

Now, after obtaining the intended values of the flight parameters, the deflections of the flight parameters can be achieved by rewriting equation (3.7) to be in the following form:

$$\Delta F_i = IntF_i - ActF_i \qquad \text{… (3.10)}$$

The deflections of the flight parameters can be corrected by exporting correction parameters as commands to the control surfaces. These commands will change the status of the control parameters by an amount proportional with the deflections occur in the flight parameters and the correction duration. The corrected state of the control parameters for each correction duration Δt can be newly formulated by taking into consideration the accumulated effect of the previous and current correction command as follows:

$$C_i = Q + \sum_{t=0}^{n} (k_{2i} \Delta F_i) \Delta t \qquad \dots (3.11)$$

where, n is the number of correction time slot, Δt is the time slot, and Q is other terms may be added to the command model.

B. Route Deflection

Route deflection is the deflection of the actual route from the intended route of the airplane. Since the route is a function of two variables (x and y), some difficulty had faced the model used to describe the current position of the airplane. It is important to suggest a model method for determining the intended route relative to the actual one, because such model is very helpful to enable determine the deflection, which occur in the position during the flight mission. The intended route is defined by some of reference points which should predefined before the airplane flight mission is started. One segment of the intended route is a set of straight lines each one connecting two adjacent reference points. The instantaneous intended position of the airplane is defined by any point lies on the intended route segment, while the actual position collected by the sensor.

Figure (3.4-a) shows the intended route is as the shortest distance between the two given way (reference) points (x_1,y_1) and (x_2,y_2). If there is a deflection in the route of the airplane, then the instantaneous position of the airplane on the actual route (x_{Act},y_{Act}) will projected on the intended route (x_{Int},y_{Int}). The suggested model aims at computing the corresponding position in the intended route depending on the known data of both the two way (reference) points and the current actual position of the airplane. Figure (3.4-b) shows the distance ΔPos, which represents the deflection of the actual position from the intended one, it is extended perpendicularly on the intended route.

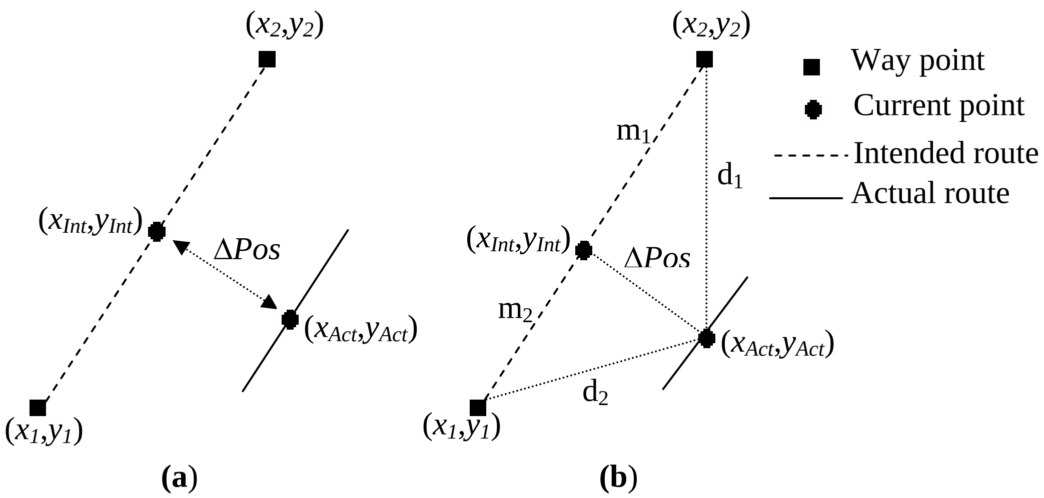


Fig (3.4) Position deflection.

By use the trigonometry, connecting the actual position with the start and end way (reference) points of the route segment, it yields four line segments m_1, m_2, d_1, and d_2. The intended route segment could be described mathematically as follows

$$\frac{x_2 - x_1}{y_2 - y_1} = \frac{x_{Int} - x_1}{y_{Int} - y_1} \qquad \text{… (3.12)}$$

let $A = x_2 - x_1$ and $B = y_2 - y_1$, then equation (3.12) can be simplified as

$$Bx_{Int} - Ay_{Int} = Bx_1 - Ay_1 \qquad \text{… (3.13)}$$

by using Pythagoras theorem for the two triangles in Figure (3.4), one find

$$d_1^2 - m_1^2 = d_2^2 - m_2^2 \qquad \text{… (3.14)}$$

where $d_1^2 = (x_2 - x_{Act})^2 + (y_2 - y_{Act})^2$

$$\begin{aligned} m_1^2 &= (x_2 - x_{Int})^2 + (y_2 - y_{Int})^2 \\ d_2^2 &= (x_{Act} - x_1)^2 + (y_{Act} - y_1)^2 \\ m_2^2 &= (x_{Int} - x_1)^2 + (y_{Int} - y_1)^2 \end{aligned} \qquad \text{… (3.15)}$$

After straightforward rearrangement of equation (3.15), the following equation was results

$$Ax_{Int} + By_{Int} = Ax_{Act} + By_{Act} \qquad \dots (3.16)$$

by solving the two equations (3.13) and (3.16), one can get the corresponding intended position components through the following equations:

$$\begin{aligned} x_{Int} &= Cx_{Act} + Dy_{Act} + E \\ Y_{Int} &= Fx_{Act} + Gy_{Act} + H \end{aligned} \qquad \dots (3.17)$$

where *C*, *D*, *E*, *F*, *G*, and *H* are constants represents the route parameters, which are given by the following form

$$C = \frac{A^2}{A^2 + B^2},\ D = \frac{AB}{A^2 + B^2},\ E = \frac{B^2 x_1 - ABy_1}{A^2 + B^2},$$
$$F = \frac{AB}{A^2 + B^2},\ G = \frac{B^2}{A^2 + B^2},\ \text{and}\ H = \frac{A^2 y_1 - ABx_1}{A^2 + B^2} \qquad \dots (3.18)$$

Now, the deflection in the position ΔPos of the airplane can be calculated as of the distance connecting two points (actual and intended) as follows

$$\Delta Pos = \sqrt{(x_{Int} - x_{Act})^2 + (y_{Int} - y_{Act})^2} \qquad \dots (3.19)$$

In order to correct the position of the airplane, the case of the deflected airplane, that shown in Figure (3.5) will be considered;

Figure (3.5-a) shows the airplane turning from the first route segment toward the second, it is deflected from the desired position (4) by a distance equal to ΔPos, to be at the actual position (3). The head in position (3) is corrected but the route still deflected by ΔPos. To correct this positioning deflection and return back to the intended position, the airplane should be deflected toward the intended route by an angle ΔH_P. So, ΔH_P should be added to the actual calculated deflection of the head

ΔH_H, which will lead to a total deflection in the head given by the following expression

$$\Delta Head = \Delta H_H + \Delta H_P \qquad \ldots (3.20)$$

The deflection of the head expressed by equation (3.20) makes the airplane pass through the continuous path shown in Figure (3.5-a). The head deflection due to positioning deflection can be calculated according to the geometry shown in Figure (3.5-b) in the following manner

$$\Delta H_P = sin^{-1}(\frac{\Delta Pos}{vt_r}) \qquad \ldots (3.21)$$

where v is the actual speed of the airplane, and t_r is the time response (predefined parameter) of the airplane that needed to correct the position.

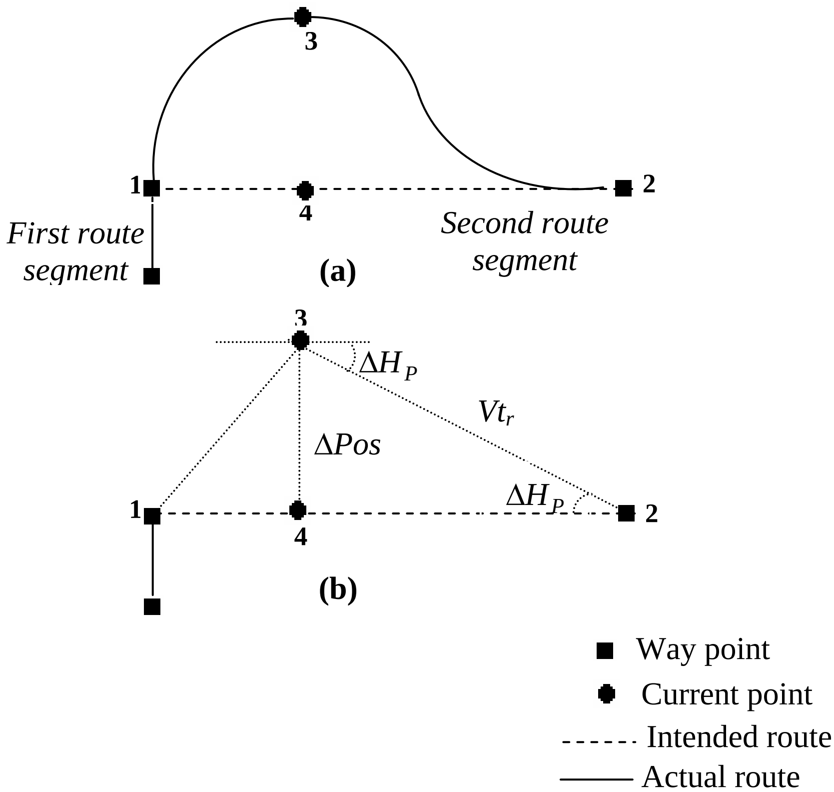


Fig (3.5) The deflection in the position.

3.3 Auto-Navigation System Design

The design of the auto-navigation system means designing a controller keeps the airplane under control and its flight satisfy certain planned conditions. This controller is called ***navigator,*** which stands for the pilot. The navigator needs to a good knowledge and deal with the control problems, especially the close loop control systems, where the events inside the loop improve quickly. Events improvement means that the state of the parameters changes from worse to better condition. The improvement remains straight forward until no change or no deflection takes place. If there is no deflection, the flight parameters reach to optimum conditions and the airplane flight is called ***level***, but the navigator should still be ready for any unsurprised changes that may happen.

There are some requirements need to be considered throughout the design phase. These requirements are; monitoring the behavior of the control and setting the input data. The navigator and its requirements are explained in the following sub-sections in details.

3.3.1 Design Requirements

The common mechanism of any designed auto-navigation system is based on determining the difference between the actual navigation attitude of the airplane and the corresponding planned attitude. The difference (or deflection) is compensated by the navigational commands issue by the system to make changes in the flight parameters such that these deflection should be reduced.

In order to sense the airplane's flight, a graphical interface is designed to display the events that the airplane pass through, these events are the spatial motion parameter and their rate of change during the flight. In addition, the interface contains a graphical map with a list of coordinates (*Lon*, *Lat*, and *Height*) of the airplane and the nearest ground stations (reference point).

The process of controlling the airplane attitude is done by controlling the navigation parameters, It is carried out by using establishing (assuming) the relationship between them, and then using the relationships to predict their changes. In the following a detailed description about the way of monitoring and setting the required changes in the navigation parameters.

A- Navigator Interface

Since the viewer cannot sense the events occur in the airplane's situations, the navigator needs to graphical interface to display the behavior of the airplane.

During the flight, the instantaneous values of the intended and actual values of the flight and navigation parameters are displayed through two adjusted gauge panels. While the control parameters are displayed on another gauge panel. The intended and actual head values are pointed on two graphical compases. Also, both the actual and intended routes are pointed as two trajectories drown on the graphical map.

The map carries information about the geographical features of the flight region. This information besides the desired conditions are put in a dedicated ***database*** established for a certain flight trip. The database has a set of records, each record includes the intended attitude of the airplane at certain route segment connecting a pair of successive way points. Table (3.3) shows an example of the airplane planned attitude registered as records in the database.

Table (3.3) Input database of an intended situations.

Way points	Position			Speed (*km/h*)	Radius of arrival (*km*)
	Longitude (*km*)	Latitude (*km*)	Height (*km*)		
1	2×91	2×112	7	100	0.5
2	5×91	4×112	10	125	0.4
3	6×91	8×112	3	100	0.6
4	12×91	3×112	15	130	0.5

The last column in table (3.3) is called the ***radius of arrival***, which helps the navigator to switch to the next way point. The process of switching is done by comparing the distance *RemDis* between the current position (x_{Act}, y_{Act}) on the actual route and the target way point (x_2, y_2) with the radius of arrival of the target way point. If *RemDis* is less than the radius of arrival then the airplane is considered arrived to the end of current route segment (reach the area of the target way point). Therefore, the auto-navigator should switch with the next way point within the planned route. The pointed circle, shown in Figure (3.6), represents the area of the arrival of the way point, and its radius is the radius of arrival of that way point.

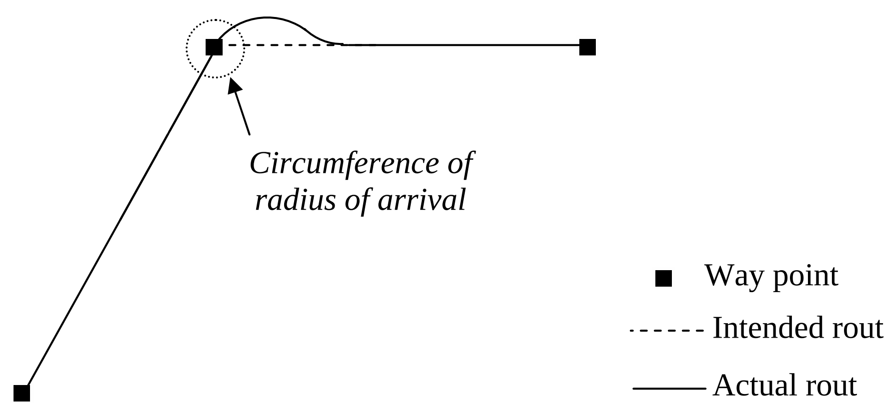


Fig (3.6) The radius of arrival

B- Sensors Calibration

The actual values of the navigation parameters are obtained by using a set of specific sensors shipped on the airplane. The readings of these sensors must be calibrated to fit the range of values of the measured parameter. Table (3.4) lists the main sensors used to measure the navigation and flight parameters in the auto-navigation process.

Table (3.4) The navigation parameters and its sensors

Parameter	**Sensor**
Speed	GPS, INS, or Airspeed
Height	GPS, INS, or Altimeter
Position	GPS, or INS
Pitch	Gyro
Roll	Gyro
Yaw	Gyro

The global positioning system (GPS) and the international navigation system (INS) provide the height of the airplane relative to the sea level, and speed relative to the ground. While the altimeter provides the actual height relative to the Earth’s surface and the airspeed determine the speed relative to the air speed.

The calibration process needs to specify the dynamic range (maximum and minimum voltage output) of each sensor, and its corresponding measured parameters. By these four known values, one can model the relationship (e.g. linear) that transforms the sensor’s voltage output to the measured physical parameter value. In the case of linear relationship

$$P = aV + b \qquad \dots (3.22)$$

where, V is the value of the sensor’s voltage output, P is the value of the parameter, a and b are the coefficients of the linear relationship.

Figure (3.7) illustrates the linear relationship between the physical parameter value of the sensor output voltage. The substitution of the maximum and minimum values of both the parameter and voltage, in equation (3.22), leads to the following two equations:

$$P_{\max} = a V_{max} + b$$
$$P_{\min} = a V_{min} + b \qquad \dots (3.23)$$

where, P_{max} and P_{min} are the maximum and minimum values of the measured parameter, and V_{max} and V_{min} are the maximum and minimum values of the sensor.

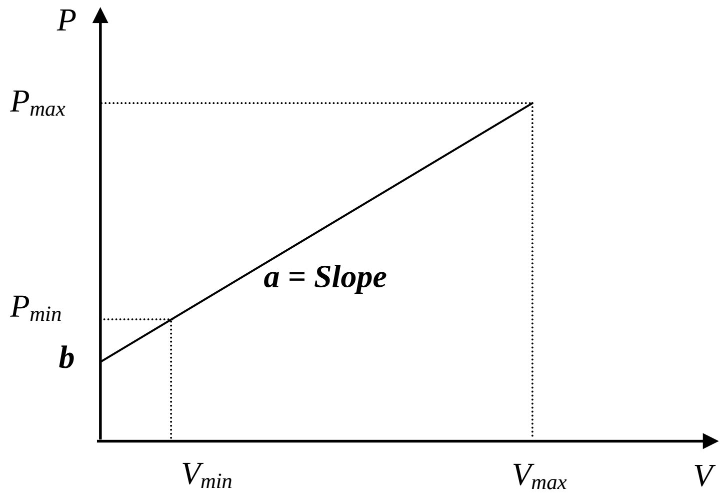


Fig (3.7) The linear relationship between the sensor's voltage output and the measured parameter value.

Equations (3.23) are two simultaneous linear equations have two unknowns, *a* and *b*, both of them can be determined by solving these two linear equations.

$$a = \frac{P_{\max} - P_{\min}}{V_{\max} - V_{\min}}$$
$$b = \frac{V_{\max} P_{\min} - V_{\min} P_{\max}}{V_{\max} - V_{\min}} \qquad \dots (3.24)$$

After the determination of the two factors *a* and *b*, their values are substituted in equation (3.22) to map the voltage output to physical parameter value.

C- Constraints on the Parameter's Rate of Change

It is important for the airplane motion to be smooth and capable to handle different flying conditions (like, bad weather). To get smooth flight the navigation parameters should change smoothly and avoid the abrupt (sudden) movements that lead to instabilities in the airplane flight.

In order to make the flight smooth, the rate change of any parameter must be restricted by a specific small limit. Therefore, the time rate of change of any navigation parameter does not allowed to be greater than a specific value, this value is called the ***maximum change***. In general, there is a maximum limit in the change of speed called ***maximum change of speed***. Also there are a ***maximum change of pitch*** for smoothing the climbing/descending motion, and ***maximum change of roll*** and ***maximum change of yaw*** for smoothing the planatic motion during the turn. The following condition illustrates the constraint applied on the flight parameters.

If $\Delta F_i > MaxCghF_i$ ***then*** $\Delta F_i = MaxCghF_i$

Else If $\Delta F_i < -MaxCghF_i$ ***then*** $\Delta F_i = -MaxCghF_i$

3.3.2 Path Tracing

One of the primary tasks done by any auto-navigator is the path tracing task. In this task, the navigation system determines the planned route coordinates at any instance of time, then the deflections between the actual airplane coordinates and the corresponding planned coordinates are determined. The objective of this task is to keep the airplane flight as more close as possible to its planned route, also this task should make the airplane flight more stable dispatched flight.

The navigator tries to navigate the airplane to pass through (or close) to several way points which represent the intended route. When the navigator start perform its task, it is initiated to navigate the airplane toward the first way point, so the first record that belong to the first way point (or first route segment) is loaded, all the required parameters in the computations are stored in this record.

During each control interval, the navigator receives the actual values of the navigation and flight parameters from the sensors, and then determines the deflections of the parameters from its set (planned values). The navigator will determine the correction commands according to the deflections in the flight parameters, then it sends these commands to the control surfaces in the airplane.

After exporting the commands, the navigator check the remaining distance between the current position of the airplane and the attended way point. If this distance is greater than the radius of the arrival then it returns to control loop and start reading the actual values of the flight parameters from sensors and repeat the command issuing process, otherwise when the distance is less or equal the radius of the arrival then the navigator will check if the current way point is the last one or not. If the way point is not the last then it returns to switch with the next way point and then reads its parameters from the next record in the database. But, if the current way point is the last, the navigator will finish the auto-navigator process by shutdown the engine and release the parachute. In the following, algorithm (3.1) presents the control operation done by the navigator.

Begin

$$\Delta Height = IntHeight - ActHeight$$
$$\Delta Speed = IntSpeed - ActSpeed$$
$$X_{Int} = C \times X_{Act} + D \times Y_{Act} + E$$
$$Y_{Int} = F \times X_{Act} + G \times Y_{Act} + H$$
$$Def = ((X_{Rout} - X_{Curnt})^2 + (Y_{Rout} - Y_{Curnt})^2)$$
$$\Delta H_P = ArcSin(Def / (t_r \times ActSpeed))$$
$$\Delta H_H = IntHead - ActHead$$
$$\Delta Head = \Delta H_P + \Delta H_H$$
$$IntRPM = k_{31} \times \Delta Speed$$
$$IntPitch = k_{32} \times \Delta Pitch$$
$if \;\; \Delta Head > 15 \; then$
$\quad IntRoll = k_{33} \times \Delta Head \quad and \quad IntYaw = 0$
$else$
$\quad IntRoll = 0 : IntYaw = k_{34} \times DeltaHead$
$end \; if$
$$\Delta Speed = IntSpeed - ActSpeed$$
$$\Delta Pitch = IntPitch - ActPitch$$
$$\Delta Roll = IntRoll - ActRoll$$
$$\Delta Yaw = IntYaw - ActYaw$$
$$Throttle = Throttle + \Delta Speed \times \Delta Tim$$
$$Elevator = Elevator + \Delta Pitch \times \Delta Tim$$
$$Aileron = Aileron + \Delta Roll \times \Delta Tim$$
$$Rudder = Rudder + \Delta Yaw \times \Delta Tim$$
$$RemDis = \sqrt{(x_{Int} - x_{Act})^2 + (y_{Int} - y_{Act})^2}$$
$if \;\; RemDis \le RadArval$
$\quad if \;\; the \; current \; way \; point \; is \; not \; the \; last \; then$
$\quad\quad Switch \; with \; the \; next \; way \; point$
$\quad else$
$\quad\quad Stop \; engine \; and \; open \; parachotte$
$end \; if$

End

Algorithm (3.1) The control calculations done by the navigator.

3.3.3 Short Turn

At the airplane turning instances, it is very important to have a proper criterion to save the time and effort. An airplane turn is explained in Figure (3.8), where the head sensor of the airplane is calibrated to have the dynamic range [–180, 180]. This range is used to set the deflection of the intended change in the head. The positive deflection of the determined head means that the turn will be done toward the right, while the negative means that the turn will be done toward the left of the airplane.

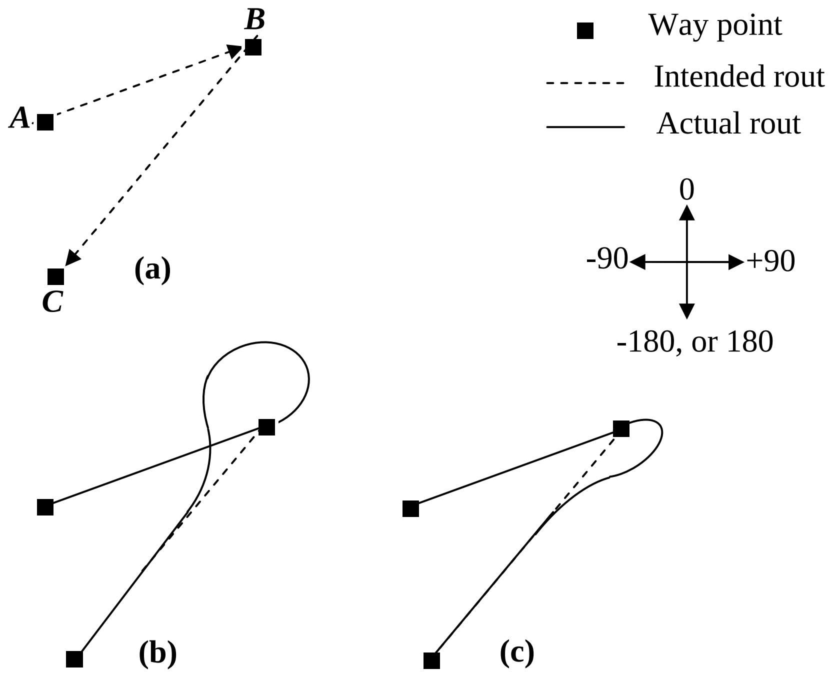


Fig (3.8) The possible turnings of an airplane.

Figure (3.8-a) shows the path of airplane flight from point A toward point B, with a head angle $H_{AB} = 70^o$, and then the airplane changes its direction toward point C by a head angle $H_{BC} = -150^o$. There are two options for the airplane to turn, Figure (3.8-b) shows the first, which is toward the left side of the airplane path with a head change:

$$\Delta H = H_{BC} - H_{AB} = -150^o - 70^o = -220^o$$

Figure (3.8-c) shows the other choice; i.e. toward the right side of the airplane path with a head change equal to:

$$\Delta H = 360^o - | H_{BC} - H_{AB} | = 360^o - | -150^o - 70^o | = 140^o$$

Obviously, the path of case (c) is shorter and easier than that of case (b). In order to avoid the long turn, the change in the head should be restricted within the range ($-180^o \geq \Delta Head \geq 180^o$) by using the conditions:

$$\begin{aligned} &if\ \ \Delta Head > 180\ \ then\ \ \Delta Head = \Delta Head\ \text{-}\ 360 \\ &else\, if\ \ \ \Delta Head < \text{-}180\ \ then\ \ \Delta Head = \Delta Head + 360 \end{aligned}$$

Applying these two conditions make the deflection that previously directed toward the left ($\Delta H_H = -220^o$) is reversed toward the right (i.e., short turn) as follows:

Since $\Delta H_H = -220^o$, that is $\Delta H_H < -180^o$

So $\Delta H_H = \Delta H_H + 360^o = -220^o + 360^o = 140^o$

3.3.4 Head Correction

The same problem of the turn may face the position correction part of the head deflection (ΔH_P). Because of ΔH_P has positive value always, it should be corrected to be directed toward the left (if ΔH_P is negative) or the right (if ΔH_P is positive) as required.

This problem is solved by assigning negative sign to the head deflection (ΔH_P) at a specific cases depending on the direction of the position correction, this direction is determined according to the amount of the changes in the coordinates (*x* and *y*) of both the current and previous way points position. The solution divide the horizontal plane of the airplane flight route into four regions as shown in Figure (3.9-a), each region is given certain type index, these indices are determined according to the following criteria.

$$\Delta x = x_2 - x_1 \text{ and } \Delta y = y_2 - y_1$$

if $|\Delta x| \le |\Delta y|$ and Δy >0 then *Type* = 0
if $|\Delta x| > |\Delta y|$ and Δx >0 then *Type* = 1
if $|\Delta x| \le |\Delta y|$ and Δy <0 then *Type* = 2
if $|\Delta x| > |\Delta y|$ and Δx <0 then *Type* = 3

where x_1 and y_1 are the components of the current way point, x_2 and y_2 are the components of the next way point.

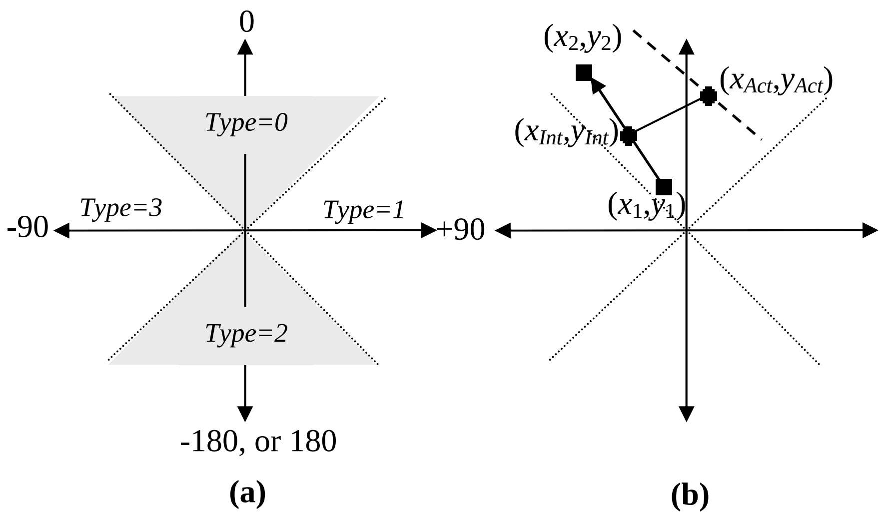


Fig (3.9) The classified four regions used in heading correction.

The direction of the airplane motion toward the right or left can be determined according to the intended change in airplane position, where the following criteria are applied (note the left turn implies set ΔH_P to be $-\Delta H_P$);

Case *type* = 0:
if $x_{Act}>x_{Int}$ then the airplane directed to the left else to the right.
Case *type* = 1:
if $y_{Act}<y_{Int}$ then the airplane directed to the left else to the right.
Case *type* = 2:
if $x_{Act}<x_{Int}$ then the airplane directed to the left else to the right.
Case *type* = 3:
if $y_{Act}>y_{Int}$ then the airplane directed to the left else to the right.

As an example, Figure (3.9-b) shows the intended route that connect the current way point (-1,2) with the next way point (-4,8), the actual and intended positions of the airplane are (1,6) and (-2,4) respectively. To decide weather the airplane turn to the left or right of the airplane, the determinations must be done as shown bellow

$\Delta x = x_2 - x_1 = -4 - 1 = -5$, and $\Delta y = y_2 - y_1 = 8 - 2 = 6$

x_{Act}=1, and x_{Int}=-2

Since $|\Delta x| < |\Delta y|$ and $\Delta y > 0$ therefore the type is 0, and since x_{Act}>x_{Int} the turn should be done to the left, which requires reversing the sign of ΔH_P. Algorithm (3.2) summarizes the implemented steps of the head correction due to position deflection.

Begin

Δx *is the deference between the two x - components of the current and next way point.*

Δy *is the deference between the two y - components of the current and next way point.*

if $|\Delta x| < |\Delta y|$ *then*

if $\Delta y > 0$ *then* $Type = 0$ *else* $Type = 2$

else

if $\Delta x > 0$ *then* $Type = 1$ *else* $Type = 3$

end if

if $Type = 0$ *and* $x_{Act} > x_{Int}$ *then* $\Delta H_P = -\Delta H_P$

else if $Type = 1$ *and* $y_{Act} < y_{Int}$ *then* $\Delta H_P = -\Delta H_P$

else if $Type = 2$ *and* $x_{Act} < x_{Int}$ *then* $\Delta H_P = -\Delta H_P$

else if $Type = 3$ *and* $y_{Act} > y_{Int}$ *then* $\Delta H_P = -\Delta H_P$

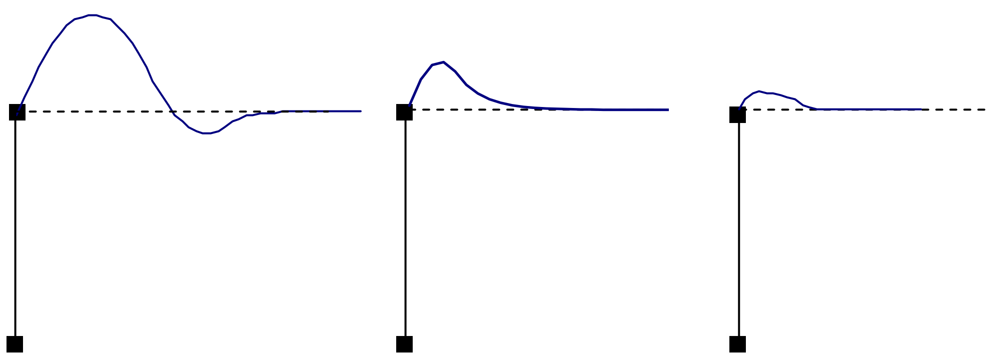
End

Algorithm (3.2) The correction of the head.

3.3.5 Overshot

Overshot is a position deflection associated with a large head correction, such as turn. The overshot may occur in three modes, as depicted in Figure (3.10). When the navigator switch to a new way point that requires large heading change, then the behavior of the airplane due to overshot could be as follows;

If the actual speed is large and the amount of required heading change is too large, then a large deflection need to be happen in the path of the airplane which may led the airplane to go far away from the intended path, as shown in Figure (3.10-a). When the actual speed is small enough and the required heading change is large, then the deflection will be small, as shown in cases (b) and (c) of Figure (3.10).



(a) Oscillated overshot. **(b)** Damping overshot. **(c)** Small overshot

Fig (3.10) Different cases of overshot.

The analysis of the above cases indicates that the amount of overshot is proportional with the amount of the actual speed of the airplane beside to the head deflection. Due to the continuous correction of the head, the case of large overshot (i.e., case a) almost causes a damping oscillation. The amount of oscillation depends on the response rate of the airplane. While in both other cases (b, and c) no oscillation happen but there is a deflection, only. The harmonic behavior of both flight and navigation parameters make the overshot gradually corrected.

In order to minimize the overshot the value of the correction command is controlled by either changing the amount of the maximum change of the flight parameters or changing the value of the proportional factor k_{2i} in the command model. The suitable maximum change of the flight parameters has narrow range, so any change in its value may affect the smooth motion of the airplane. Thus, it is better to leave it with a proper value. The variation of k_{2i} according to flight situations leads to make the response of the navigator more dynamic. Therefore, it is better for k_{2i} to be called ***response coefficient***. This suggestion gives more flexibility for the control task, especially in the turn or sudden events (like strong wind effects).

3.3.6 Route Analyzer

When there are two very close way points in the flight mission plane, it is possible that the navigator pass the second way point without need to make switching. The high speed of the airplane may make the positioning deflection due to turning very large. In case of very large deflection the next way point will not be on head of the airplane, so it is called ***lost*** since the airplane does not pass through.

The lost way point occurs when the speed of the airplane is high and the needed turn is sharp as shown in Figure (3.11-a), the third way point in the figure is lost, where one can notice that the high deflection in the airplane path make the airplane exceeds the third way point. It is easy to note that any way point satisfies the following condition may lost:

$$Def_{\max}(x, y) > Dis \qquad \dots (3.25)$$

$$Def_{max} = IntSpeed_1 \times t_r \qquad \dots (3.26)$$

$$Dis = \sqrt{(x_2 - x_3)^2 + (y_2 - y_3)^2} \qquad \dots (3.27)$$

Where, Def_{max} is the maximum position deflection in the horizontal plane, and Dis is the distance between the current (second) and next (third) way point, as shown in Figure (3.11-b).

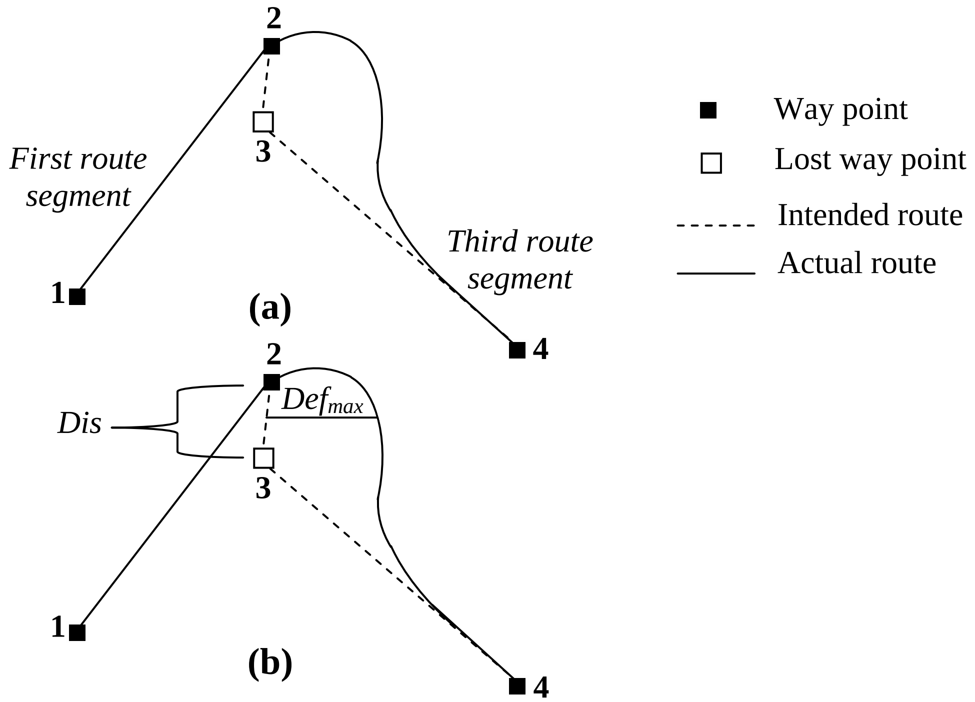


Fig (3.11) Lost way point in the intended route during a turn.

The margin of the flight of the guided airplane is just around the actual route of the airplane, and it is always trying to pass through the next way point. So, any failure in the mission of directing the airplane toward the target way point lead to failure in the flight mission. Some suggestions have been considered to solve this problem, these solutions depend on the condition of either the airplane can ignore the lost way point and switch to the forth way point or it cannot. The three suggested solutions are;

1. Try to find the lost way point by using the ***route analyzer*** before navigation, and then prepare a navigation plane to adjust the flight route to be correctly directed toward target way point.
2. Back the airplane to a point that makes the airplane able to see the lost way point.
3. Pass the lost way point and switch to the way point that follow the lost one.

The first option is safe to use. A route analyzer algorithm can be used before performing any auto-navigation task, the algorithm should make a plane to redirected the airplane flight to be toward the lost way point. This algorithm is successful when it is implemented once before the normal switching to simulate the airplane route consistency, where this analyzer could be used to detect the existence of a pair of successive way points close to each other, such that the auto-navigation system may be incapable to direct the airplane to the second way in the pair. In such case the position of one of these points (or both) should be adjusted to avoid the occurrence of such situation during the flight. Algorithm (3.3) presents how the route analyzer operates.

Begin
    For each way point
        $Def_{max,i} = IntSpeed_i \times t_r$
        $Dis_i = \sqrt{(x_{i+1} - x_i)^2 + (y_{i+1} - y_i)^2}$
        if $Def_{max,i} > Dis_i$ ***then***
            Print "The target way point is lost, please enter_
                the additions on both coordinates x, and y"
            Input x_{ad}, y_{ad}
            $x_i = x_i + x_{ad}$
            $y_i = y_i + y_{ad}$
        end if: $i = i + 1$
    loop
end

Algorithm (3.3) Analyzing the planned route.

3.4 Auto-Navigation System Simulation

The simulation is useful to test the auto-navigation system, because the auto-navigation real tests are very expensive, so it is better to simulate the airplane flight rather than make an actual one. The simulation process of the auto-navigation needs to build dedicated software to simulate the airplane flight situations, such software is called ***simulator***. The simulator should be capable to sense the actual behavior of the airplane flight under various simulated natural conditions. After ensuring a safe flight throughout the simulator, the established auto-navigator can be tested in real world.

In the simulation, the auto-navigator plays the role of the pilot that governs the flight mission, the simulator plays the role of the airplane obeys to the navigator commands. Both the simulator and navigator will exchange the flight status information and commands to make the simulation process running continually.

The simulator sends the initial actual values of the navigation and flight parameters to the navigator. On the other hand, the navigator, repeatedly, receives the data of the actual parameters, compute the intended flight parameters according to equation (3.9), and determine the deflections in the flight parameters, which are required for determining the needed correction commands according to equation (3.10). The correction commands are computed according to equations (3.11) and then they send to the control surfaces of the simulator. After that, the simulator sends the actual values of the navigation parameters to the navigator, then the navigator redetermine the correction commands and send them, and so on. The close loop between the simulator and navigator will continue till the end of the flight trip.

3.4.1 Simulator

The simulator is software emulates the flight behavior of an actual airplane under various predefined flight situations. The simulator uses a set of physical relationships to generate the proper flight behavior expectations.

The outputs (sensor's reading) of the simulator are expressed by means of four specific gauges appear in the navigator user interface. At the simulation trip, the simulator tells the navigator about the instantaneous status of each control surface and sends the actual values of the navigation and flight parameters to the navigator, and correspondingly the simulator will obey to the navigator correction commands.

In the case of actual airplane flight, there is a specific sensors carried on the airplane platform used to sense the values of the navigation and flight parameters, then the values of these parameters are transferred to the navigator. While in the case of the simulator, this process is substituted by doing some computations that employ some mathematical relationships, driven according to some physical principles, to assess the actual values of the navigation and flight parameters. The simulator assesses the actual flight parameters by using the set of known values of the control parameters that issued in the last correction round. The new values of the flight parameters depend on the accumulation of the changes (events) occur in the control surfaces, this expressed by the following linear relationship between the flight and control parameters

$$ActF_i = \sum_t \left(k_{4i} \Delta C_i\right)_t \qquad \ldots (3.28)$$

where, k_{4i} is the ith constants represents the proportional factor of the ith command channel.

The actual values of the navigation parameters can be estimated by using the estimated changes of the flight parameters at each time slot by the following formula

$$ActN_i = \sum_t k_{5i} \Delta F_i \qquad \dots (3.29)$$

Furthermore, the actual position components are determined with the aid of equations (3.2) by the following form

$$P_i = P_{0i} + \sum_t \Delta P_i \qquad \dots (3.30)$$

where P_i is airplane position coordinated (may be *X*, *Y*, or *Z*) depending on the pointer (i) which takes one of the values (1, 2, or 3, respectively), ΔP_i is the change of the i^{th} positioning which occur during the time slot.

In the following, algorithm (3.4) shows the simulator calculations during the simulation trip.

Begin

$$ActRPM = ActRPM + k_{41} \times \Delta Throttle$$
$$ActPitch = ActPitch + k_{42} \times \Delta Elevator$$
$$ActRoll = ActRoll + k_{43} \times \Delta Aileron$$
$$ActYaw = ActYaw + k_{44} \times \Delta Rudder$$
$$ActHead = k_{53} \times ActRoll \times ActSpeed + k_{54} \times ActYaw$$
$$ActSpeed = ActSpeed + k_{51} \times \Delta RPM$$
$$X_{Act} = X_{Act} + ActSpeed \times Cos(ActPitch) \times Sin(ActHead)$$
$$Y_{Act} = Y_{Act} + ActSpeed \times Cos(ActPitch) \times Cos(ActHead)$$
$$Z_{Act} = Z_{Act} + ActSpeed \times Sin(ActPitch)$$
$$ActHeight = Z_{Act}$$

End

Algorithm (3.4) The simulation calculations done by the simulator.

3.5 Dynamic Navigator Response

The response of the guided airplane can be defined as the time rate of change in the airplane attitude (in terms of navigator or flight parameters) relative to the command value. The change in the response coefficient (k_{2i}), will causes a change in the response of the parameters.

The response may consider ***static*** (when the response coefficient is given a constant value along the time of flight mission), or ***dynamic*** (when the response coefficient vary with the time according to a specific criterion). It is expected that the dynamic response is better than the static one, because it provides an accurate and more stable behavior.

Actually, the response cases whether they are slow or high both are useful to handle certain navigation tasks. When the deflection in one (or more) of the navigation parameters is large, then high airplane response is needed to correct the deflection and return back to the desired condition within short time. While in the small deflection case the high response may causes overshot and consequently oscillations, thus a slow response is needed to avoid the overshot occurrence.

The results of long analysis work to understand the effect of k_{2i} coefficients indicated that its value should be varied during the flight mission in order to make the airplane response variable and consequently more stable. The value of k_{2i} should continually update, and its new values should depend on the latest previous registered response (i.e., deflection/command), with keeping the condition $k_{2i,min} \le k_{2i} \le k_{2i,max}$ (let $k_{2i,min} = 1$ and $k_{2i,Max} = 10$). Therefore, a mathematical model for determining the needed instantaneous k_{2i} value can be put forward depending on the amount of the instantaneous deflection/command.

Almost, the variation of the dynamic response is determined either ***manually*** by using a joystick or ***automatically*** by the navigator. The manual method requires existence a ground pilot (joystick user) in the ground station. The ground pilot can improve the airplane performance as needed, he can gradually increasing/decreasing the amount of the airplane response by manually increasing/decreasing the value of the response coefficient. This response change is done only when the airplane faces difficulty for reaching the intended situation during the flight. The automatic method depends on computing the amount of the response needed to make the airplane directed toward the desired attitude within time interval. In the following a detailed discussion is given about both the manual and automatic methods.

3.5.1 Manual Method

This method depends on the functional relationship between the response coefficient and an external joystick that have at least four channels, see Figure (3.12). Each channel in the joystick has a response coefficient for a specific flight parameter versus its corresponding command. The four channels of the joystick must be calibrated to fit the allowed values of the response coefficient. The success of this method depends on the experience of the ground pilot. He can interfere only at the emergency situations that face the airplane or when the airplane shows abnormal flight. It must be taken into account that any mistake caused by the ground pilot may cause a noticeable mistake in the flight performance.

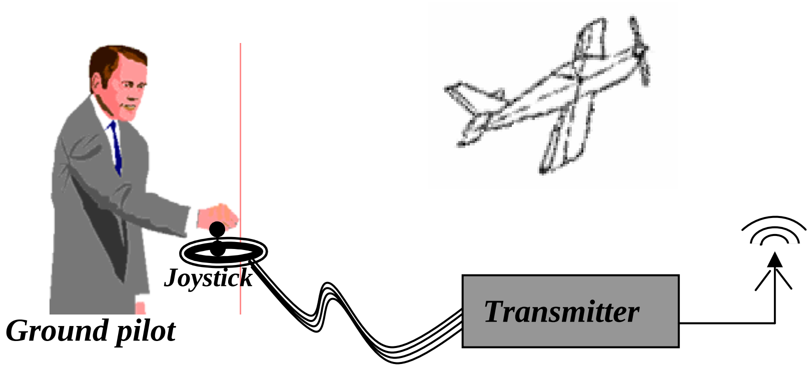


Fig (3.12) Ground pilot in the auto-navigation process.

3.5.2 Automatic Method

The automatic adjustment of response coefficient depends on the relationship between the amount of variation of the corresponding navigation or flight parameter with its associated command.

The modeling of the response coefficient as a function of deflection requires determining the maximum and minimum deflections for each navigation and flight parameter, shown in table (3.5). The absolute value of the deflections is taken since the response coefficients values are always positive.

It should be mentioned that the position deflection is embedded in the head relations, so when the position deflection is zero ($\Delta Pos = 0$) i.e. minimum, then $\Delta H_P = 0^o$, while when it has a maximum deflection $\Delta Pos = MaxSpeed \times t_r$ then $\Delta H_P = 90^o$.

Table (3.5) Maximum and minimum deflections of
the navigation and flight parameters.

Parameter	Min. deflection	Max. deflection
Speed (*m/s*)	0	200
Height (*m*)	0	12000
Head (*degree*)	0	180
RPM (*rpm*)	1000	8000
Pitch (*degree*)	0	180
Roll (*degree*)	0	180
Yaw (*degree*)	0	30

The automatic method is modeled in two different ways. The ***first*** is a deflection based method, it depends on the instantaneous deflection occur in the navigation parameters as incentive for the response. While the ***second*** is time averaging method, where the latest registered *N*-values of the parameter variating versus their commands are used to redetermine the new value of response parameter. The progress of correction versus the command decides whether the current response is slow or fast, and how should be adjusted. In the following a detailed explanation is given about the two adopted methods.

A. Deflection based Method

The modeling of the response coefficient as a function of the required correction needs careful choice for the type of the adopted mathematical model (relationship), which may be represented by any increasing function.

By adopting the linear model, the response coefficient as a function of the deflection will be given as follows

$$k_{2i} = aDef_i + b \qquad \text{… (3.31)}$$

where Def_i is the deflection of the i^{th} navigation parameter, a and b are the linear relationship coefficients.

Let us consider the following example to know how the parameters a and b are calculated, the height parameter is taken in this example because it is easy to imagine. Substituting the maximum and minimum values of both the response coefficient and the required correction (deflection), that listed in table (3.5), the following two equations are obtained

$$k_{22,max} = aDef_{2,max} + b$$

$$10 = a\,(12000) + b \qquad \text{… (3.32)}$$

$$k_{22,min} = aDef_{2,min} + b$$

$$1 = a\,(0) \quad + b \qquad \text{… (3.33)}$$

by solving equations (3.32) and (3.33) simultaneously, the coefficients of the linear model a and b can be obtained

$$a = 0.00075 \;\; and \;\; b = 1$$

therefore, the final form of equation (3.31) will be

$$k_{22} = 0.00075\, Def_2 + 1 \qquad \dots (3.34)$$

Table (3.6) shows samples of the response coefficient values for different value of deflection. Figure (3.13) shows the linear behavior of the response coefficient according to equation (3.34), it is noticeable that the influence of the response coefficient is very slow when the deflection is less than 1*km*, which is regarded as a large deflection. Therefore, this model is not suitable.

Table (3.6) Response coefficient versus deflections by linear model.

Def_i (m)	k_{22}
5	1.003
10	1.007
100	1.075
1000	1.75
6000	7
12000	10

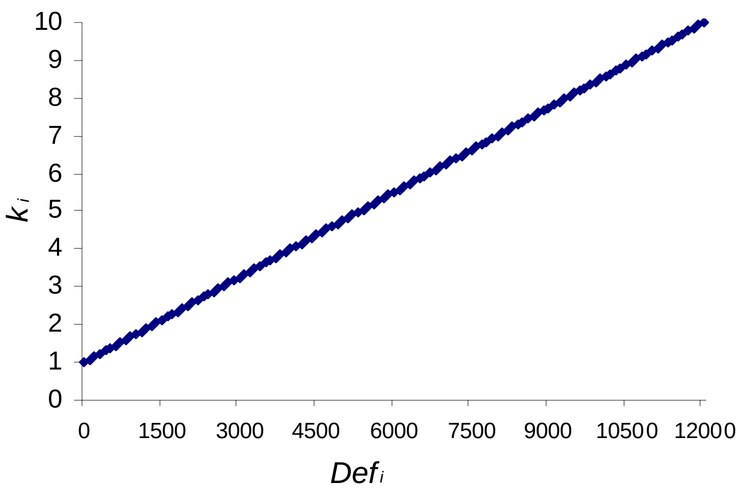


Fig (3.13) Linear model of the response coefficient.

In order to exceed this problem, one can model the response coefficient as a logarithmic function, where the response may take proper values for the small deflections. Therefore, the alternate model is denoted in the following form

$$k_{22} = a\ log\ (Def_2 + 1) + b \qquad \dots (3.35)$$

by following the same mentioned procedure, substituting the maximum and minimum values of both the response coefficient and height deflection; the following two equations are obtained:

$$k_{22,max} = a\ log\ (Def_{2,max} + 1) + b$$

$$10 = a\ log(12000 + 1) + b \qquad \dots (3.36)$$

$$k_{22,min} = a\ log\ (Def_{2,min} + 1) + b$$

$$0 = a\ log(0 + 1) + b \qquad \dots (3.37)$$

The two coefficients a and b are computed and found $a = 2.45145$, *and* $b = 1$ thus, equation (3.35) will be

$$k_{22} = 2.45145\ log\ (Def_2 + 1) + 1 \qquad \dots (3.38)$$

Table (3.7) presents some samples of response coefficient values determined according to the logarithmic model, while Figure (3.14-a) shows the behavior of the computed response coefficient according to equation (3.38).

Table (3.7) Response coefficient versus deflections by logarithmic model.

Def_2 (m)	k_{22}
5	1.907
10	2.553
100	4.913
1000	7.355
6000	9.262
12000	10

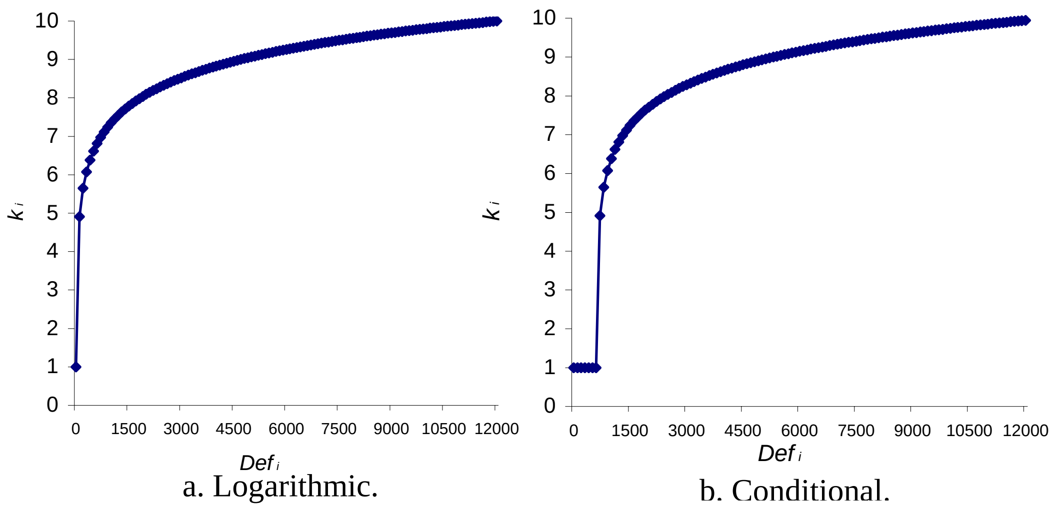


Fig (3.14) The logarithmic and conditional models of the response coefficient.

This model is very effective for the small and large deflections, but there is a problem facing this model which is when the deflection is smaller than the maximum change of the flight parameter where the response coefficient become high, this may cause unstability in the flight performance and let airplane flight and navigation parameter oscillates around their set values. This problem can be avoided by adopting a conditional model implies the logarithmic relationship for specific range of deflection, this model is expressed by the following equation and its behavior is shown in Figure (3.14-b):

$$k_{22} = \begin{cases} log(Def_2 + 1) & for\ Def_2 \geq MaxChgF_2 \\ 1 & for\ Def_2 < MaxChgF_2 \end{cases} \quad \ldots (3.39)$$

Generally, the response coefficient of the i^{th} flight parameter will be written as follows

$$k_{2i} = \begin{cases} log(Def_i + 1) & for\ Def_i \geq MaxChgF_i \\ 1 & for\ Def_i < MaxChgF_i \end{cases} \quad \ldots (3.40)$$

where $MaxChgF_i$ is the maximum change of the i^{th} flight parameter.

B. Time Averaging Method

This method deals with the previous changes occurred in the flight parameters due to their corresponding commands. It determines the response coefficient as the ratio between both the intended changes (***planned***) and the occurred (***actual***) changes of the flight parameters along specific number of previous correction intervals (time slots). This implies that the response coefficient will be a function of time.

The historical records of the response coefficient for some previous intervals are required to analyze the parameters correction behavior by comparing the actual change of the flight parameters with the intended changes that planned be occurred. The ratio between them is decided; if it has a large value then the response coefficient need to be increased, else lower response coefficient is needed.

The temporal analysis considers the airplane is moving toward the desired situation, by assuming *P* is the integral value of the planned changes in the i^{th} flight parameter during *N* intervals, and *A* is the integral value of the actual changes in the i^{th} flight parameter during the same *N* intervals, then the ratio between *P* and *A* will decreased by a small rate is proportional to the amount of the correction as follows

$$k'_{2i} = \frac{P}{A} = \frac{\sum_{i=1}^{N} (\Delta F_i)_{Planned}}{\sum_{i=1}^{N} (\Delta F_i)_{Actual}} \qquad \dots (3.41)$$

where k'_{2i} is the correction rate happen in k_{2i} during *N* intervals (i.e., $k_{2i} = k'_{2i} \times k_{2i}$). Figure (3.15) shows the behaviors of the intended correction (k'_{2i}) in k_{2i} with respect to $IntF_i, ActF_i$, ΔF_i, and predicted ΔF_i. For each set of *N* samples, *P* is the summation of planned ΔF_i while *A* is the summation of the actual ΔF_i. The value of k'_{2i} is greater than one at time less than 4.8 *s* since the predicted change (*P*) is greater than the actual change (*A*), which makes to be corrected. At time greater than 4.8 *s* both *P* and *A* are identified (i.e., k'_{2i}=1) and k_{2i} remains as it is, hence the influence of the correction rate is proper. Algorithm (3.5) shows how the time averaging method improves the value of k_{ji} according to the change of *N* intervals.

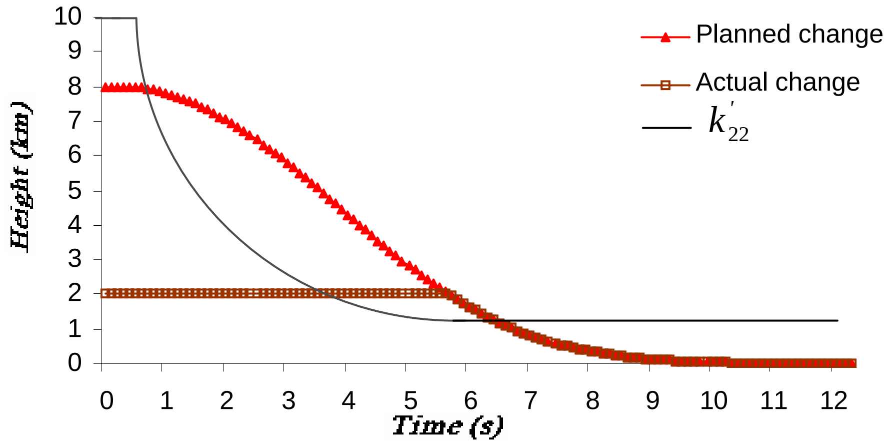


Fig (3.15) Low response for small deflection.

```
Begin
   For n=1 to N-1
      TempF(n+1)=TempF(n)
   Next n
   TempF(1) = ΔF_i
   For n=1 to N
      P=P+TempF(n)
   Next n
   If |ΔF_i|>MaxChgF_i then ΔF_i =MaxChgF_i
   For n=1 to N-1
      TempPF(n+1)=TempPF(n)
   Next n
   TempPF(1) = ΔF_i
   For n=1 to N
      P=P+TempPF(n)
   Next n
   k'_2i = P / A
   k_2i = k_2i × k'_2i
End
```

Algorithm (3.5) Improving the response coefficient using the time averaging method.

3.6 Considered Effects Modeling

The considered effects are the most important regards that may affect the behavior of the airplane during the flight. They are discussed and modeled frequently according to its importance in the following sections:

3.6.1 Wind Effect

The wind has a noticeable effect on the airplane during its flight, its effect varies under different flight conditions, and it depends on the wind relative speed, direction, and effective time of the wind. The wind effect must be studied carefully in order to understand how the airplane behaves against the wind effect.

Normally, the wind is moving from the region of high atmospheric pressure toward the region of low atmospheric pressure. It has a speed and direction. The direction of the wind is described by the region that the wind comes from. In the present analysis, the wind is considered moving with a predefined speed and direction. The sensor that measures the relative speed and direction of the wind is called ***pitot***, this sensor is putting on the airplane to sense the airspeed and direction then its reading is send to the navigator.

The picture of interest is regarding the wind as a sequence of vertical fronts move with same speed and direction. The wind strikes the airplane will deflect its position, while the wind strikes the airplane during its pitching, rolling or yawing maneuver will affect the stability of the airplane, it will generate a torque tries to rotate the airplane about the lateral, longitudinal, or vertical axis respectively as shown in Figure (3.16). These three torques are existing besides the effective wind forces that pushing the airplane toward the wind direction the reason that makes the airplane is budging from its position into another.

In order to model the effect of the wind on the navigator performance, the behavior of the flying airplane must be analyzed with regard to the wind effect, this is provide a proper reaction for the navigator in terms of a mathematical model to avoid the possible deflections and rotations due to the existing of wind effect.

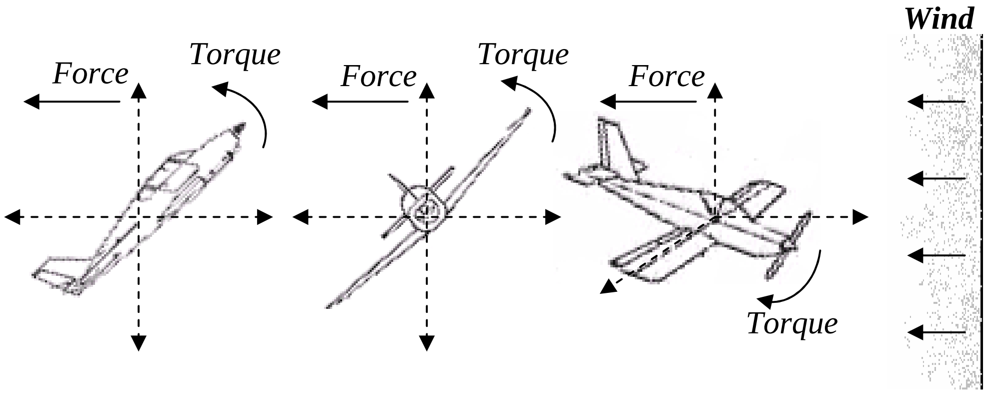


Fig (3.16) Aerodynamic forces and torques applied on the airplane by the wind.

For the aim of analyzing the behavior of the airplane against the wind effect, the general case of the airplane that inclined by any angle with respect to the wind in the three dimensional space is considered. Figure (3.17-a) shows an airplane flying with an actual speed (*ActSpeed*) and direction *Head* (i.e., velocity is $\vec{V}$) according to the adopted heading angle. The wind is moving with velocity ($\vec{W}$) and strikes the airplane by an angle (ϕ) in the horizontal plane.

The flight behavior of the airplane at different wind speed and direction indicates that the speed of the airplane decreases when the direction of the wind is opposite to the direction of the airplane, and increases when the wind has the same direction of the airplane. Also, the opposite direction of the wind affects the horizontal stability, the reason that increases/decreases the climbing rate of the airplane when the wind is in the same/opposite direction of the airplane for a specific pitch angle. Moreover, the wind that strikes the airplane from the side affect the lateral stability, so the position of the airplane will be deflected.

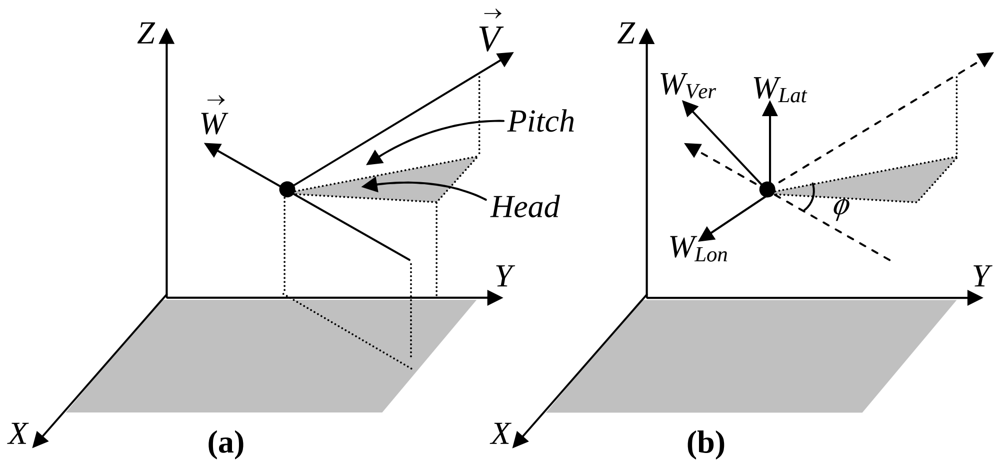


Fig (3.17) Components of wind speed with respect to the reference coordinates when the pitch and roll have non zero values.

As a result, any inclined wind strikes the airplane deflects the speed and the position of the airplane.

The wind is moving in a planatic form, thus its velocity should be analyzed into three components with respect to the airplane's coordinates that belong to the principal airplane's axes (longitudinal, lateral, and vertical axis). These components are the longitudinal component (W_{Lon}) lateral component (W_{Lat}), and the vertical component (W_{Ver}) as shown in Figure (3.17-b).

Analytically, the forces and torques applied on the airplane could be analyzed to be in the direction of the speed components of the wind. The directional stability provided by the thrust force prevents the torques of the components W_{Lon}, and W_{Ver} to be occurred. Therefore, there is only one torque is generated, which is the torque of the component W_{Lat}. This torque has a value depends on the angle *Roll*. The component W_{Lon} generates a force toward the forward/backward of the airplane depending on the angle ϕ, the component W_{Lat} generates a side force cause a shift in the airplane position toward its left/right side along the flight path depending on the angle ϕ. Also, the component W_{Ver} generates a force toward the upward/downward direction, which climb/descend the airplane depending on the angle *Pitch*.

The value of the component W_{Lat} depends on the angle ϕ, and the values of the components W_{Lon} and W_{Ver} depend on the angle *Pitch*. So the wind speed is analyzed according to the airplane coordinates in the following components

$$\begin{aligned} W_{Lon} &= W\cos(\phi)\cos(\theta) \\ W_{Ver} &= W\cos(\phi)\sin(\theta) \\ W_{Lat} &= W\sin(\phi) \end{aligned} \qquad \dots (3.42)$$

where, $\theta = 180 - ActPitch$ when the wind is opposite to the direction of the airplane, and $\theta = -ActPitch$ when the wind has the same direction of the airplane as shown in Figure (3.18).

The component W_{Lon} increases/decreases the actual speed of the airplane depending on the angles θ and ϕ, therefore, the true speed (resultant of both actual and wind speed) is given as follows

$$True\ speed = ActSpeed + W_{Lon} \qquad \dots (3.43)$$

The torque caused by the component W_{Ver} is equal to zero when the pitch angle is zero, this means the wind has little (insignificant) effect on the height of the airplane. But when the actual pitch has a non zero value, then the component W_{Ver} will be analyzed into three sub-components with respect to the reference coordinates (X, Y, and Z) as shown in Figure (3.19-a), these components are; the two planatic components W_{VerX} and W_{VerY} budge the airplane on the horizontal plane, and the vertical components W_{VerZ} that increases/decreases the height of the airplane. Thus the three sub-components of W_{Ver} are given by the following

$$W_{VerX} = W_{Ver} \cos(ActPitch)\cos(\phi^{'})$$
$$W_{VerY} = W_{Ver} \cos(ActPitch)\sin(\phi^{'}) \qquad \ldots (3.44)$$
$$W_{VerZ} = W_{Ver} \sin(ActPitch)$$

Where $\phi^{'}$ is the angle between the Y-axis and the projection of W_{Ver} on the X-Y plane (i.e., $\phi^{'} = Head + Roll + 180^{o}$; $-180^{o} \le \phi^{'} \le 180^{o}$).

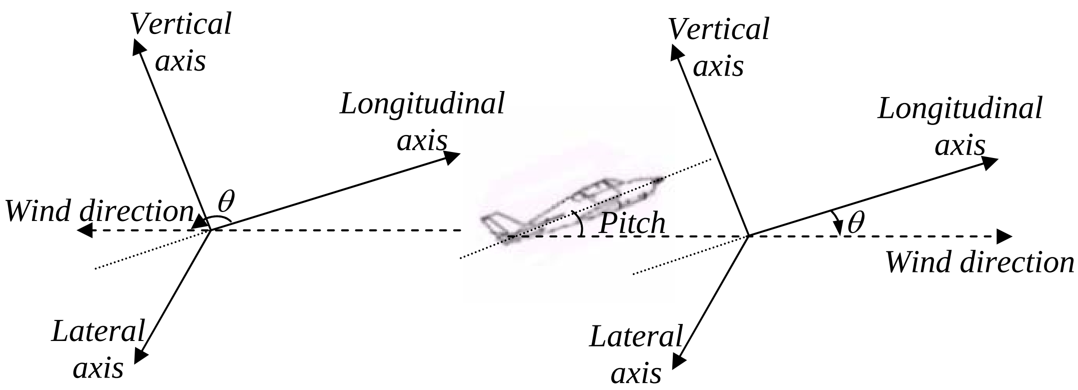


Fig (3.18) Determination of angle θ.

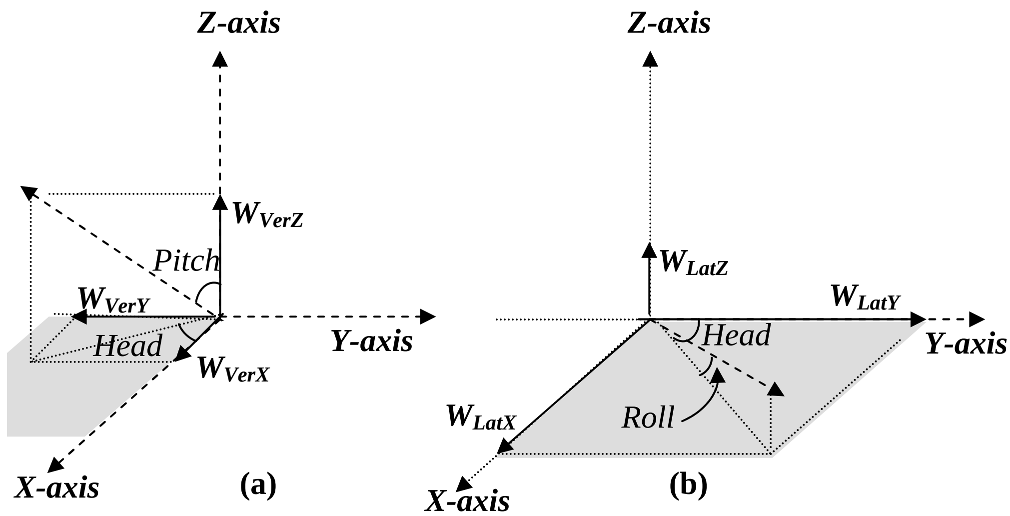


Fig (3.19) Wind speed components with respect to the reference coordinates X, Y, and Z.

The component W_{Lat} provides the airplane by both the shifting force and rotating torque. It is analyzed into three sub-components with respect to the reference coordinates as shown in Figure (3.19-b). These components are; the two planatic components W_{LatX} and W_{LatY} budge the airplane along the *X*-*Y* plane, and the vertical component W_{LatZ} is responsible on generating the torque which rotate the airplane around the longitudinal axis along its direction. The share of each one of the shifting force or rotating torque depends on the actual *Roll*. Therefore, the mathematical representation of the three sub-components of W_{Lat} is given as follows

$$\begin{aligned} W_{LatX} &= W_{Lat}\cos(\rho)\cos(\phi^{''}) \\ W_{LatY} &= W_{Lat}\cos(\rho)\sin(\phi^{''}) \\ W_{LatZ} &= W_{Lat}\sin(\rho) \end{aligned} \qquad \ldots (3.45)$$

where, $\rho = ActRoll \ \{0^o < |ActRoll| < 90^o\} \ \ \rho = 0^o$ for $ActRoll = \pm 90^o$, and $\phi^{''}$ is the angle between the *Y*-axis and the projection of the lateral component W_{Lat}, (i.e., $\phi^{''} = Head + 90^o;\ -180^o \le \phi^{''} \le 180^o$).

Therefore, the total positioning deflection due to the lateral and vertical components of the wind speed at Δt on the *X*-*Y* plane are;

$$\begin{aligned} \Delta X &= (W_{VerX} + W_{LatX})\Delta t \\ \Delta Y &= (W_{VerY} + W_{LatY})\Delta t \end{aligned} \qquad \ldots (3.46)$$

While the rate of change of the height of the airplane due to the vertical component of wind speed is

$$\Delta Z = W_{VerZ}\Delta t \qquad \ldots (3.47)$$

The generated torque due to the lateral component of the wind speed will rotate the airplane by an angle ΔR_T, this angle can be determined depending on the wind speed component W_{Lat} as

$$\Delta R_T = \omega \Delta t = \frac{W_{LatZ}}{r} \Delta t \times \frac{180}{\pi} \ (Degree) \qquad \dots (3.48)$$

where ω is the angular velocity measured in radian, and r is the distance between the center of the airplane and the end of the wing, ΔR_T takes a value within the range $\{-90^o \leq \Delta R_T \leq 90^o\}$.

In order to suggest a good performance for the navigator, one must first imagine how the airplane behaves without the existence of the navigator; assume the airplane is unguided (i.e., freezing the navigator guidance). The deflection of the airplane flight will increase as long as the wind effect exist. After each time duration the speed, height, and position are deflected due to the applied wind forces, the lateral torque deflects the roll, which may causes a deflection in the head angle of the airplane. While, the deflections increase continually the airplane's attitude keep changes far away from the intended one.

With the existence of the navigator, the resistant reaction of the navigator versus the wind action will be as follows: First, the navigator detects the deflections that occurred in the flight attitude of the airplane, then it exports correction commands for the aileron to take an angle ΔR_P in order to stop the positioning deflection due to the planatic lateral components W_{LatX} and W_{LatY}.

$$\Delta R_P = \frac{(W_{LatX} + W_{LatY})}{r} \Delta t \times \frac{180}{\pi} \ (Degree) \qquad \dots (3.49)$$

ΔR_P lies on the range $-90^o \leq \Delta R_P \leq +90^o$.

Later, the navigator will face the effect of the torque by making the aileron taken the angle ΔR_T in direction is opposite to direction of the effect (i.e., the sign of ΔR_T is reversed) in order to stop the torque effect on the lateral stability when the *Roll* has a non zero value, i.e. both ΔR_T and ΔR_P are added to the aileron while just ΔR_P is added to the intended roll in the navigator computation with a step of sign reversion of both ΔR_P and ΔR_T .

After stopping the effect of the lateral force and torque, the navigator exports a command for the throttle to compensate the change in the actual speed due to the longitudinal force. Already, the remaining deflections are the deflections of the height and position only, they corrected by the navigator as mentioned before. The issue of these corrections continues as long as the wind effect exists. As a result, the throttle and aileron stay in the abnormal state.

It is very necessary for the control surfaces to return back to the normal state when the effect is vanished, i.e. wind speed is zero. This includes that the throttle should set to a value that make the actual speed is equal to the true speed, and the aileron is set zero. The sudden stop of the wind effect must be taken into accounts during the simulation process.

According to the above analyses algorithm (3.6) was established to describe the behavior of the simulator against the effect of the wind, and shows how the control parameters return back to the normal state by reducing the wind effect at any time during the flight.

Begin

$ActSpeed = ActSpeed + \Delta Speed$

If *Wind effect* is regarded ***then***

$\phi = Head_{wind} - Head \ (-180^o \le \phi \le 180^o)$

$if \ |\phi| < 90 \ then \ \theta = -Pitch \ else \ \theta = 180 - Pitch$

$W_{Lon} = W \cos(\phi)\cos(\theta)$

$W_{Ver} = W \cos(\phi)\sin(\theta)$

$W_{Lat} = W \sin(\phi)$

$TrueSpeed = ActSpeed + W_{Lon}$

$\phi' = Head + Roll + 180 \quad (-180^o \le \phi' \le 180^o)$

$W_{VerX} = W_{Ver} \sin(Pitch)\sin(\phi')$

$W_{VerY} = W_{Ver} \sin(Pitch)\cos(\phi')$

$W_{VerZ} = W_{Ver} \cos(Pitch)$

$\phi'' = Head + 90^o \ (-180^o \le \phi'' \le 180^o)$

$if \ |Roll| < 90 \ then \ \rho = Roll \ else \ \rho = 0$

$W_{LatX} = W_{Lat} \cos(\rho)\sin(\phi'')$

$W_{LatY} = W_{Lat} \cos(\rho)\cos(\phi'')$

$W_{LatZ} = W_{Lat} \sin(\rho)$

$\Delta R_T = W_{LatZ} \times r \times \Delta t$

$\Delta R_P = \sqrt{W_{LatX} + W_{LatY}} \times r \times \Delta t$

$\Delta X = (W_{LatX} + W_{VerX}) \times \Delta t$

$\Delta Y = (W_{LatY} + W_{VerY}) \times \Delta t$

$\Delta Z = W_{VerZ} \times \Delta t$

$if \ \Delta R_P = 0 \ then$

$X = X + \Delta X$

$Y = Y + \Delta Y$

end if

$Z = Z + \Delta Z$

else

$\Delta ActSpeed = ActSpeed - TrueSpeed$

$TrueSpeed = TrueSpeed + \Delta ActSpeed$

end if

End

Algorithm (3.6) The calculations of expected deflections due to wind effect.

3.6.2 Engine Efficiency Effect

The engine capabilities should depend on some technical specifications of the airplane, they should taken into account during the design of the airplane. Some of the engine specification are put after studying the forces may imposed on the airplane, for example thrust should dominates the drag force as needed. Therefore, there is a maximum thrust, which should push the airplane to reach its maximum speed.

There are two important factors which affect the engine efficiency; the ***first factor*** is the quality of the fuel in use, where the bad quality of the fuel decreases the efficiency of the engine performance, since fuel firing will be not completed and it causes a waste of carbonate composites that dirt the valves in the firing chamber. Dirty valves do not work properly, so the engine cannot reach its optimal performance. The ***second factor*** is the consumption of the engine due to its old life.

The engine with low efficiency makes the airplane takes long time to reach its desired attitudes, the airplane response becomes slower. In such case, the high changes in the control parameters may cause a stall the airplane flight. The inclusion of the engine efficiency in the considered auto-navigation model needs a careful analysis to the effects of some engine status on its efficiency, studying the behavior of the airplane at different engine efficiencies, and finally the analysis lead to a proper mathematical model to describe the effect of degraded engine efficiency on the flight.

Technically, the engine efficiency (E) can be defined as the ratio between the output power (P_{output}) to the input power (P_{input}), which may be written as follows

$$E = \frac{P_{output}}{P_{input}} \times 100\% \qquad \ldots (3.50)$$

The input power is related by the quantity of the spent fuel (throttle) to make the engine reach the desired revolution per minute, while the output power P_{output} is related by the generated thrust (T) that needed to support the airplane by the maximum speed (S), which given by the following form.

$$P_{output} = T \times S \qquad \dots (3.51)$$

In other words, the output power is the ability of the engine to provide the specific maximum torque, which given as follows.

$$P_{output} = 2\pi N\tau \qquad \dots (3.52)$$

where N is the number of revolution per minute (*RPM*) of the engine, and τ is the torque of the engine [***prismtechnologies***].

It is very necessary to distinguish the most important terms that describe the performance of the engine which are; the ***throttle*** and the number of the ***revolutions per minutes*** (*RPM*). The throttle determine the input power since it express the rate of the consumed fuel, whereas the *RPM* is related to the output power which in turn affects the resulting thrust and consequently the actual speed. The low efficient engine spent plenty of fuel in order to provide the required *RPM* that derive the airplane to reach the desired speed. Therefore, when the engine is not efficient, a large throttle is always needed to reach the required *RPM* (on consequently the required actual speed). The relation between them depends on the engine efficiency.

Analytically, the engine with high efficiency is able to push the airplane from any position to another at shortest possible time, this implies the time rate of change of the speed is equivalent to the time rate of the intended changes as given in equation (3.6) previously. While the engine of low efficiency provides the airplane by a speed less than the normal by a fraction, this fraction depends on the percentage ratio of the engine efficiency.

Therefore, when the engine is not fully efficient, then the airplane will move slower. In fact the change of the flight parameters will be slower, and it depends on amount of the degraded engine efficiency. The engine efficiency has a direct effect on the speed of the airplane since it affects the thrust (i.e., *RPM* of engine). Also, it affects the remaining flight parameters indirectly since they are affected by the speed. Therefore the relationship that relate the throttle with the speed is needed to be updated with regarding to the engine efficiency, so equation (3.6) for the first (i=1) flight parameter (*RPM*) in the simulator calculations will be reformed to be in the following form.

$$\Delta RPM = k_{2i} \times E \times \Delta Throttle \qquad \ldots (3.53)$$

where E is the percentage ratio of the engine efficiency. Equation (3.53) let the navigator changes the throttle control surfaces in order to reach with the airplane to the desired *RPM*. Therefore, the navigator increases the amount of the spent fuel to provide the *RPM* that needed to push the airplane by the desired speed. As a result, the engine efficiency affects the time rate of changing the height and the head indirectly beside its direct effect on the speed.

3.6.3 Altitude Effect

The altitude has a great effect on the engine power, and consequently, the airplane performance will be affected. The power of the airplane’s engine decreases as the altitude increase, this is due to the fact that the concentration of oxygen decreases at higher altitudes, this makes the required oxygen for complete fuel firing is greater than that exist, which affects engine power. Thus, there is an allowed altitude for each type of engine (or airplane).

All types of the airplane's engine consume a large amount of the fuel and oxygen to generate the required thrust that the airplane depends on it during the arising and flying. The consumption of fuel means firing it with existing plenty of oxygen. The complete firing required a specific amount of oxygen proportional with the amount of the fuel in the firing chamber, in the case when the oxygen is less than the needed amount, the firing will be not completed. Little available oxygen will weaken the power of the engine, and then the thrust seems to be less than that of the need.

Any decrease in the engine power makes its performance is not sufficient and its *RPM* may become less than the standard margin. Therefore, it should be noted that the increase in the spent throttle causes a corresponding increasing in the *RPM*, but with a rate less than the normal. As a result, *RPM* will be followed the throttle always but with small fallback.

In order to modeling the effect of the altitude on the flight performance, it is needed to analyze this effect and then decide what the flight parameters are affected, and what is the range of this effect.

The concentration of air in the atmosphere varies with altitude. It has high concentration at low altitudes and it decreases with the increase of altitude. The adopted distribution function of the air concentration has exponential behavior with the altitude.

It is taken into consideration that the oxygen consist 21% of the air composition at the Earth surface, and it is assumed that the oxygen concentration take the same picturing (exponential) of the air distribution, but with less concentration at each altitude as shown in Figure (3.20) for both air and oxygen.

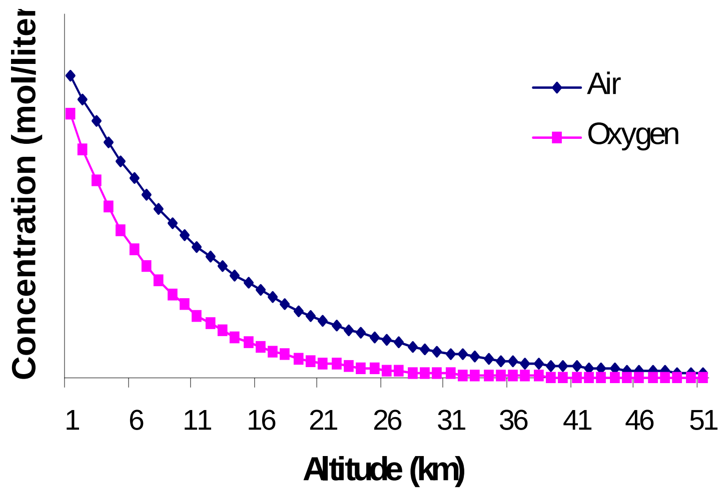


Fig (3.20) The concentration of the air and oxygen in the atmosphere.

The different designs of the airplane lead to different capabilities of flight. The airplane capabilities are explained in a catalog assigned with each design of the airplane. The catalog contains a power chart describes the limit of the altitude that the airplane can reach. The altitude that less than the pointed limit is allowed for fly, while the altitude that is greater than the limit is forbidden because small concentration of oxygen may lead to shutdown the engine. As a typical example, the power chart describes the linear relationship between the relative power of the engine as a function of the altitude is shown in Figure (3.21). The relative power is the ratio between the power of the engine at a specific altitude to the power of the same engine at zero altitude. Thus, the relative power is taking a value less than or equal to unity.

The mathematical representation for decreasing the engine power at high altitude takes the same treatments of the engine efficiency mentioned before, since both are effect on the output power (i.e., *RPM*) that needed to generate the intended thrust.

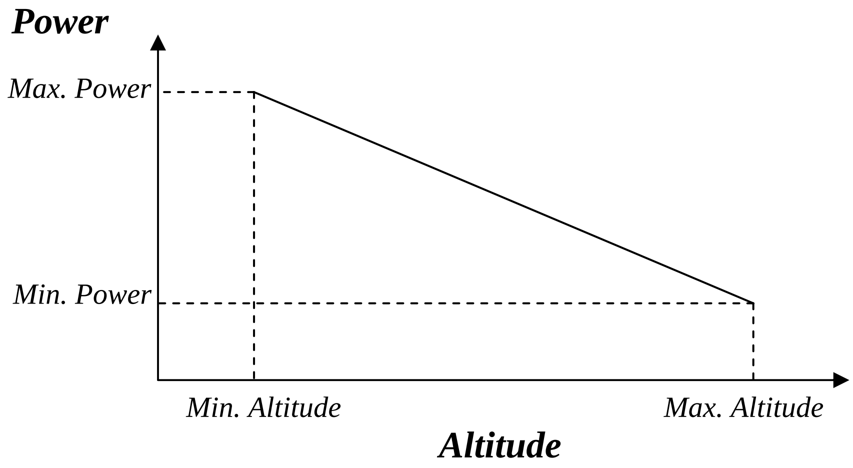


Fig (3.21) The relative engine power chart of the airplane versus altitude.

The mathematical treatment of the altitude effect should take into account that the simulator senses the changes of the flight parameters. These changes should have values less than the intended changes by an amount represents the percentage of the engine power (*P*). The following relationship was driven from the power chart corresponding to the actual height of the airplane

$$\Delta RPM = k_{2i} \times P \times \Delta Throttle \qquad \ldots (3.54)$$

The driving of the engine relative power (P) at any altitude require knowing the upper (P_{max}) and lower (P_{min}) limits of the engine relative powers that corresponding to the allowed highest (Hgh_{max}) and lowest (Hgh_{min}) altitude. These limits help to represent the power chart quantitatively, and then describe any value of the power corresponds to any height by the following relationship.

$$P = \frac{P_{max} - P_{min}}{Hgh_{max} - Hgh_{min}} \times ActHgh + \frac{Hgh_{max} \times P_{min} - Hgh_{min} \times P_{max}}{Hgh_{max} - Hgh_{min}} \qquad \ldots (3.55)$$

CHAPTER TWO

AERODYNAMICS AND AEROPLANE

Chapter Contents

CAPTER TWO

AERODYNAMICS AND AEROPLANE

2.1 Preamble

The objective of the aerodynamics is to decipher the natural laws and to give methods for calculation of the airflow properties. In turn, these properties allow to determine practical quantities, such as the forces applied on the airplane. Also, one can determine some proportional parameters; like the engine specifications, the amount of fuel and oxidizer that ignited in the combustion chamber, and the flow velocity and pressure at the nozzle exit [**Ber, 1998**].

The basic principles of aerodynamics can lead to the calculation of the exit flow velocity and pressure, which in turn lead to the calculation of thrust. For these reasons, the study of aerodynamics is vital to the overall understanding of flight [**Kue, 1986**].

There are certain physical laws that describe the behavior of airflow and define the various aerodynamic forces acting on the airplane's surface. These principles of aerodynamics provide the foundations for a good understanding of what may be termed the "Theory of flight" [**Nel, 1989**].

Aerodynamics involves the use of advanced mathematics and physics. This chapter presents only the basic principles of aerodynamics and their application to the flight of airplane without the necessity of advanced mathematical analysis. Aerodynamics can be more easily understood by the individual whose primary concern lies within the flight of the airplane.

2.2 Physical Properties of the Air

The aerodynamic forces acting on the airplane's surface are due in great part to the properties of the air mass in which the surface is operating. Air is mixture of several gases, it has one-fifth Oxygen and four-fifths Nitrogen. Pure, dry air contains about 78 percent (by volume) Nitrogen, 21 percent Oxygen, and 0.9 percent Argon. In addition, air contains about 0.03 percent Carbon dioxide and traces of several other gases, such as Hydrogen, Helium, and Neon. The distribution of gases in the air is shown in Figure (2.1) [**Elt, 1966**].

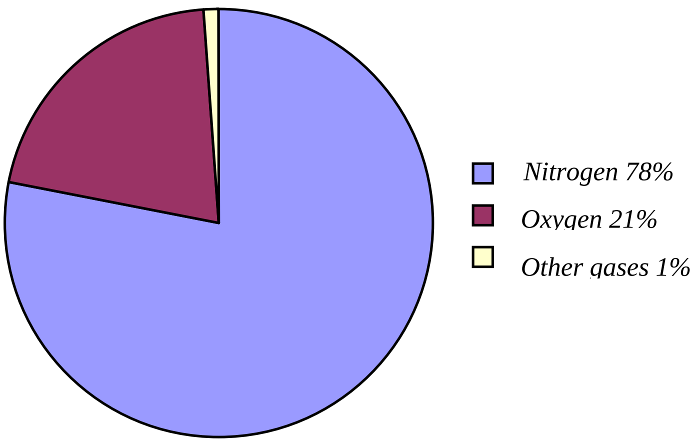


Fig (2.1) Distribution of gases in the atmosphere

2.2.1 Static Pressure

The atmosphere is the whole mass of air extending upward hundreds of miles. It may be compared to a pile of blankets. Air in the higher altitudes, like the top blanket of the pile, is under much less pressure than the air at the lower altitudes. The air at the Earth's surface may be compared to the bottom blanket because it supports the weight of all the layers above it. The static pressure of the air at any altitude results from the mass of air supported above that level [**Bai, 2001**].

Since air has weight, it is easy to recognize that the pressure of the atmosphere will vary with altitude. This is illustrated in Figure (2.2). Notice that at 20000*ft* (6097.56 *m*) the pressure is less than the half sea level pressure. This means, that more than half of the atmosphere lies below the altitude of 20000 *ft* even though the “outer” half extends hundreds of miles above the earth [**Del, 2001**].

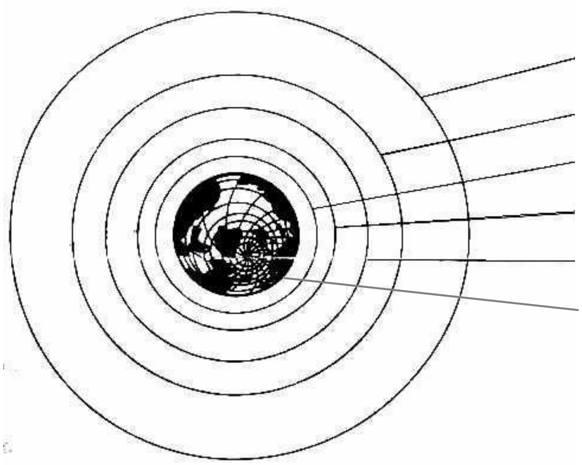


Fig (2.2) Pressure of the Earth’s atmosphere at various altitudes.

2.2.2 Temperature

Under standard conditions, temperature decreases at approximately 1.98^oC for each increase of 1000 *ft* (304.88 *m*) of altitude until an altitude of 38000*ft* (11585.44 *m*) is reached. Above this altitude the temperature remains at approximately –56.5^oC [**Lin, 2002**].

Literatures on meteorology often state that the temperature normally decreases with altitude at a rate of approximately 0.5C^o per 100 *m*, or about 1F^o per 300 *ft*. This amounts to a decrease of about 1.52C^o for each increase of 1000 *ft*, which is different from the decrease under standard conditions. It must be remembered that the literatures using the forgoing values are discussing average rather than standard conditions [**Chr, 2000**].

2.2.3 Density

The density of the air is a property of great importance in the study of aerodynamics. Air is compressible, as the air compressed, it becomes more dense because the same quantity of air occupies less space. Density varies directly with pressure and remains constant with the temperature [**Sed, 1985**].

A general gas law defines the relationship of pressure, temperature, and density of gases when there is no change of state or heat transfer. So density varies directly with pressure, inversely with temperature. On a hot day, air expands and becoming less dense, conversely on a cold day, the air contracts, and become more dense. Changes in air density affect the flight of an airplane. With the same thrust, an airplane can fly faster at a high altitude, where the density is lower than at a low altitude, where the density is greater. This is because the air offers less resistance to the airplane when it contains a smaller number of particles of air per unit volume [**Ste, 1999**].

2.2.4 Humidity

Humidity is a condition of moisture or dampness. The higher the temperature of the air, the more water vapor it can hold. The water vapor weight is approximately five-eighths as much as an equal volume of perfectly dry air. Therefore, when air contains 5 parts of water vapor and 95 parts of perfect dry air, it is not as heavy as air containing no moisture. This is because water is composed of Hydrogen (an extremely light gas) and Oxygen. Air is composed principally of Nitrogen, which is almost as heavy as Oxygen. Assuming that the temperature and pressure remaining the same, the density of the air varies with the humidity. On damp days the density is less than it is on dry days [**Kla, 1987**].

2.3 Physical Forces by Air

The physical forces that support an airplane in flight may be explained by two basic laws of physics, Bernoulli's principle and Newton's third law of motion [**Sed, 1985**]. In the following some details are given about each of them.

2.3.1 Bernoulli's Principle

Bernoulli discovered that as the air velocity increases, the pressure decreases, and as the velocity decreases, the pressure increases. Actually, Bernoulli's principle states that the total energy of a particle in motion is constant at all points on its path in a steady flow. The most appropriate means of visualizing the effect of airflow and the resulting aerodynamic pressure is to study the fluid flow within a closed tube. Figure (2.3) shows the airflow at position (1) in a tube has a certain velocity and static pressure. As the airstream approaches the constriction at position (2), certain changes must take place. Since the airflow is enclosed within the tube, the mass flow at any point along the tube must be the same and the velocity or pressure must change to accommodate this continuity of flow. As the flow approaches the constriction of position (2), the velocity increases to maintain the same mass flow [**Sul, 1998**].

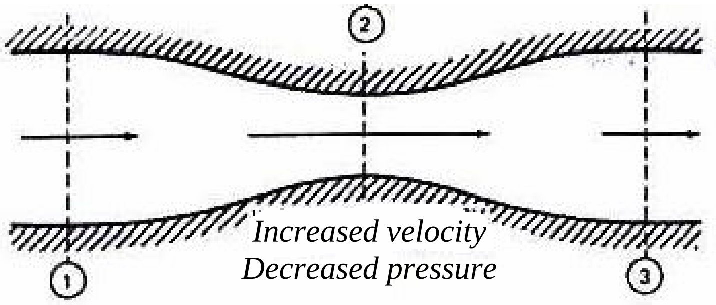


Fig (2.3) Bernoulli principle

As the velocity increases, the static pressure will decrease. The total energy of the airstream in the tube is unchanged. However, the airstream energy may be in two forms. The airstream may have a potential energy which is related to the static pressure, and a kinetic energy represented by its dynamic pressure (velocity). As the total energy is unchanged, an increase in velocity (kinetic energy) will be accompanied by a decrease in static pressure (potential energy). Therefore, it can be said that the sum of static and dynamic pressure in the flow tube remains constant, and any change in dynamic pressure produces the same magnitude change in static pressure [**Dou, 1986**].

2.3.2 Newton's Third Law

Newton's third law of motion states, for every action there is an equal and opposite reaction. This also plays a role, along with Bernoulli's principle, in the development of lift. When there is an angle between the wing and the direction of the airstream, the air is forced to change its direction. If the wing is tilted upward against the airstream, the air flowing under the wing is forced downward. The wing therefore applies a downward force to the air, and the air applies an equal and opposite upward force to the wing, this is ***lift***, which is illustrated in Figure (2.4), where the angle through which the airstream is deflected in by any lifting surface is called the ***downwash*** angle, it is important when control surfaces are studied, because they are normally placed to the rear of the wings where they are influenced by the downward deflected airstream known as the downwash [**Dil, 2004**].

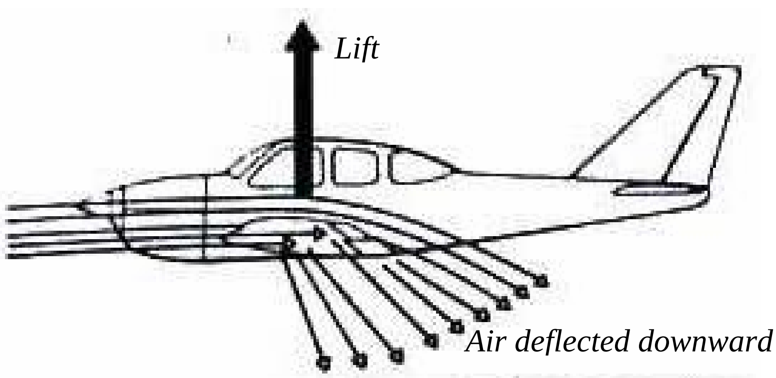


Fig (2.4) Wing deflecting the air downward.

2.4 Airfoils

An airfoil is defined as any surface or wing designed to obtain reaction from the air through which it moves [**Meg, 1988**].

If the wing of an airplane were sawed through from the leading edge to the trailing edge, the side view of the section through the wing at that point would be its airfoil section. An airfoil profile is merely the outline or shape of an airfoil section. Figure (2.5) illustrates five airfoil profiles of different shapes together with their chords. A ***chord*** is defined as a reference line directly across an airfoil from the leading edge to the trailing edge [**Alt, 2005**].

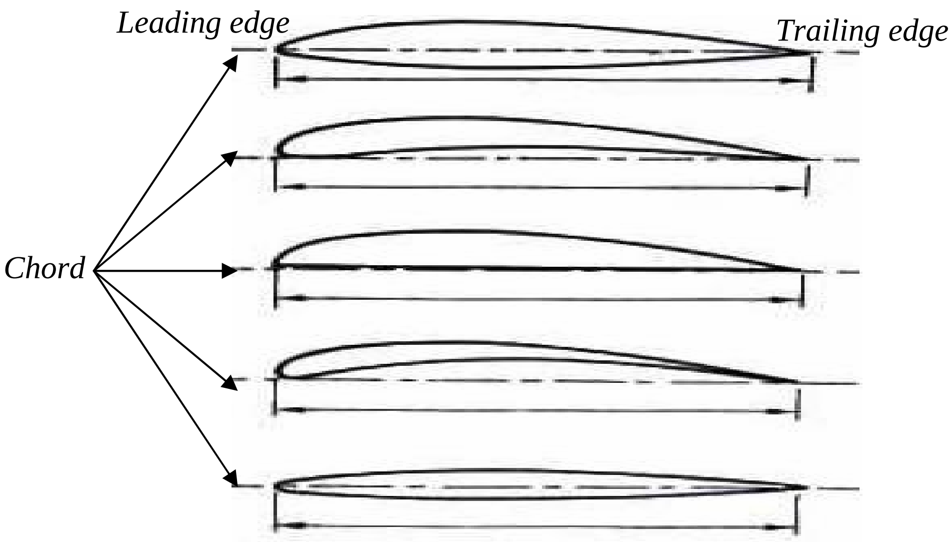


Fig (2.5) Airfoil profile of different shapes.

The angle formed by the intersection of the wing chord line and the horizontal plane passing through the longitudinal axis of the airplane is called the ***angle of incidence*** of a wing, as illustrated in Figure (2.6).

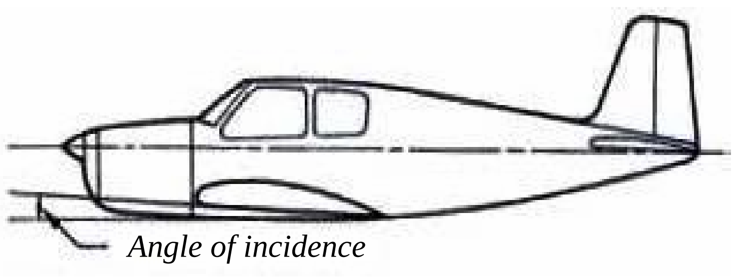


Fig (2.6) Angle of incidence.

Airplanes are usually designed with a positive angle of incidence in which the leading edge of the wing is slightly higher than the trailing edge. The correct angle of incidence is essential for low drag and longitudinal stability [**Mil, 2005**]. The lateral angle of the wing with respect to a horizontal plane is called the ***dihedral***, as shown in Figure (2.7). Positive dihedral exists when the tip of a wing is a horizontal plane passing through the root of the wing. Negative dihedral exists when the tip of the wing is below the horizontal plane passing through the root of the wing. The positive dihedral provides stability for the airplane [**Ebe, 1999**].

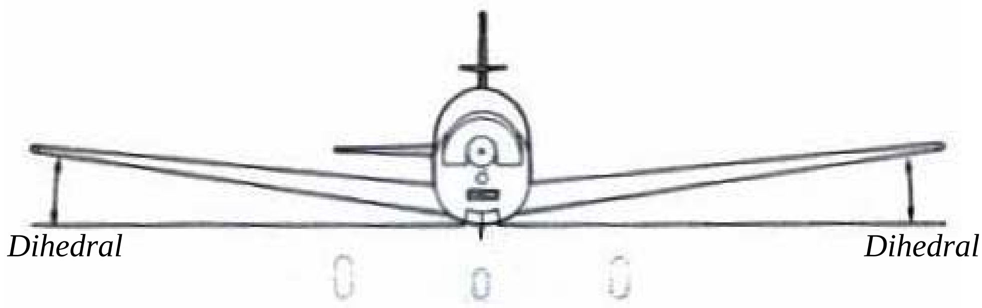


Fig (2.7) Dihedral of airplane wings

2.5 Airflow

When the air strikes the leading edge of the wing, the passage of the air is obstructed and its velocity is reduced. Some particles of the air flow over the upper surface and some flow under the lower surface, but all separating particles of air must reach the trailing edge of the wing at the same time. The effect produced by a wing moving through the air is illustrated in Figure (2.8) [**Duu, 1970**].

Air particles that pass over the upper surface have farther distance to go and therefore must move faster than those passing under the lower edge. In accordance with Bernoulli's principle the increased velocity above the wing results in a lower static pressure than that existing below the wing. Since there is a difference of pressure, the greater must prevail, and therefore an upward force exerted on the wing, this force is called lift [**Meg, 1988**].

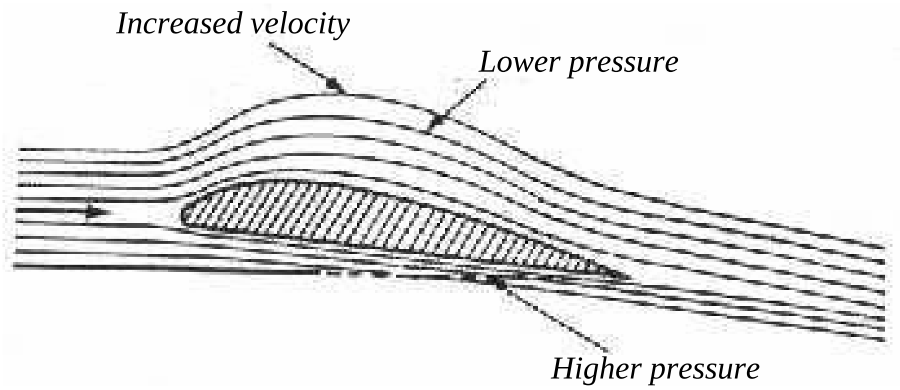


Fig (2.8) Pressure created by a wing in flight.

All designs of the profile aim at avoiding air resistance that acting on the wing, besides to perform a high quality flight. The acute angle between a reference line in a body and the line of the relative wind direction is the angle of attack, as shown in Figure (2.9).

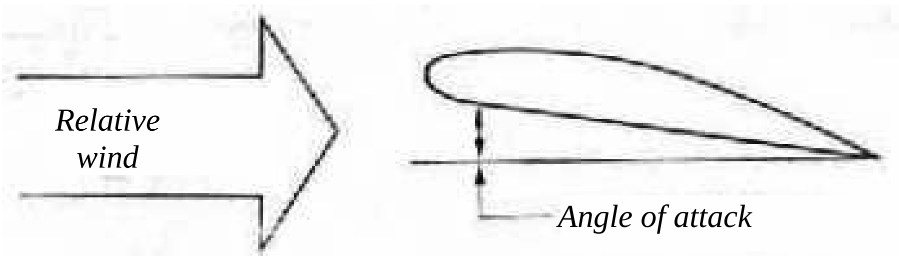


Fig (2.9) Angle of attack

The arrowhead at the left in Figure (2.9) represents the relative wind. The relative wind is the velocity of the air with respect to a body in it, it is usually determined from measurements made at such a distance from the body such that the disturbing effect of the body upon the air is negligible. In other words, the relative wind refers to the velocity of the air before it strikes the leading edge of the airfoil and divides the flow around it. In calm or still air, the direction of the relative wind is opposite to the flight path of the airplane with reference to the ground. The term ***center of pressure*** referred to the point at which the chord of an airfoil section intersects the line of action of the resultant aerodynamic forces and about which the pressures balance. Figure (2.10) illustrates the center of pressure point [**San, 2006**].

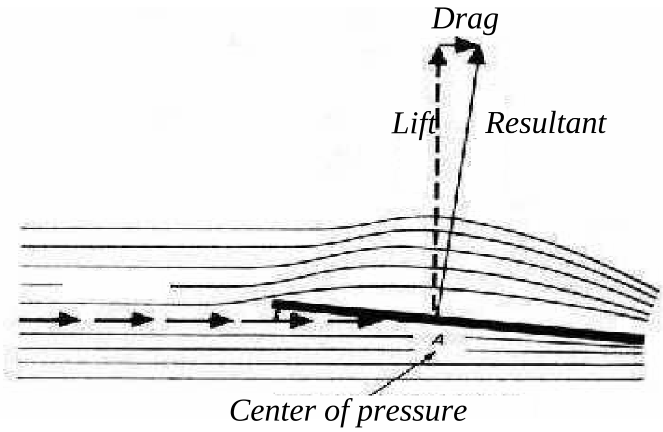


Fig (2.10) Flat plate in a stream of air

There are some conditions the wing pass through, which have more interest in the flight studies. These regarded conditions are discussed in the following subsections.

2.5.1 Laminar Flow

Laminar flow employs the concept that air is flowing in thin sheets or layers close to the surface of a wing with no disturbance between the layers of air. This means, there is no cross-flow of air particles from one layer of air to another. Also, there is no sideways movement of air particles with respect to the direction of airflow [**McC, 1979**].

Laminar flow is most likely to occur where the surface is extremely smooth and especially near the leading edge of airflow. Under these conditions the boundary layer will be very thin. The boundary layer is the layer of air adjacent to the airfoil surface. The air velocity in this layer varies from zero on the surface of the airflow to the velocity of the airstream at the outer edge of the boundary layer. The cause of the boundary layer is the friction between the surface of the wing and the air. Figure (2.11) presents that the airflow at the leading edge of a smooth-surface wing will be laminar, but as the air moves toward the trailing edge of the wing, the boundary layer becomes thicker and laminar flow diminishes [**Sch, 1989**].

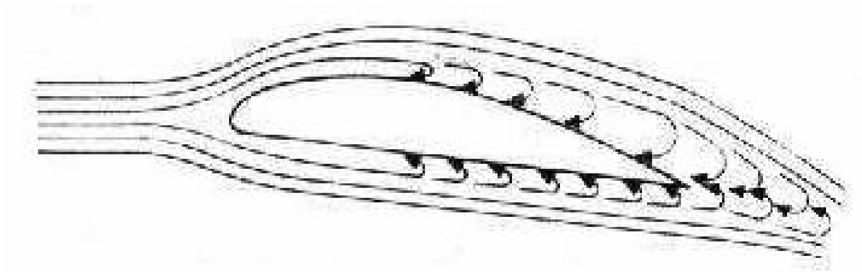


Fig (2.11) Development of the boundary layer as a result of skin friction

2.5.2 Turbulent Flow

Osborne Reynolds found that at low speed the flow is smooth but at high speed the flow is turbulent. He found a value which he called the critical ***Reynolds number*** (*R*). The flow is laminar for speed values below the critical R and turbulent for the values above the critical R. This value worked well for the flow of liquids inside circular tube, but it has to be handled differently when it is applied to the flow of air around objects which are unconfined, such as airfoils [**Asm, 1999**].

Whether a laminar or turbulent boundary layer exists around an airfoil, it depends on the combined effects of velocity, viscosity, density, and the size of the chord. It is the combined effects of these important parameters which produce the Reynolds number. The formula for the Reynolds number of any airfoil is

$$R = \rho V (L / \mu) \qquad \ldots (3.1)$$

where, ρ is air density, V is velocity, L is the dimension (usually the chord length), and μ is the coefficient of viscosity. High Reynolds numbers obtained with large chord surfaces, high velocities, and low altitude. Low Reynolds number is determined for small chord surfaces, low velocities, and high altitudes [**Bir, 2004**].

2.6 Drag Force

When an airplane flies through air, then air is moved. When any mass is moved or accelerated, force is required, and the application of force produces an equal and opposite force (reaction), this is in match with Newton's laws of motion. The impact of the air against the surfaces of the airplane applies force which tends to hold the airplane back. This force is called ***drag***. Specifically, drag is a retarding force acting upon a body in motion. Drag may classified into two main types: ***induced drag*** and ***parasite drag*** [**Ber, 1998**].

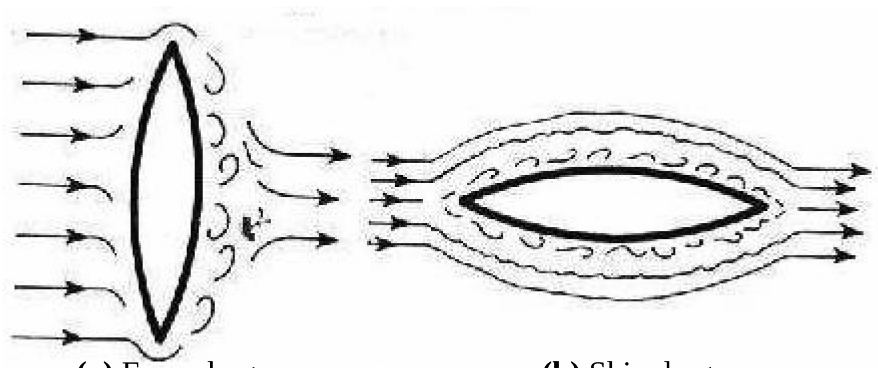
Induced drag is the undesirable but unavoidable. It is a product of lift, it increases in direct proportion with increases in the angle of attack. The greater the angle of attacks (up to the critical angle), the greater amount of lift is developed and the greater is the induced drag. The airflow around the wing is deflected downward, producing a rearward component to lift, which is the induced drag. The amount of air deflected downward increases greatly at higher angles of attack. Therefore, the higher the angle of attack the slower the airplane is flown, the greater the induced drag [**Wan, 2004**].

Parasite drag is the resistance of the air produced by any part of the airplane that does not produce lift. Several factors affects parasite drag. When each factor is considered independently, it must be assumed that other factors remain constant. These factors are:

a. The more streamlined an object is then less parasite drag is produced.
b. The more dense the air moving past the airplane, the greater the parasite drag becomes.
c. The larger size of the object in the airstream, the greater the parasite drags.
d. As speed increases, the amount of parasite drag increases. If the speed is doubled, 4 times as much drag is produced.

Parasite drag can be further classified into ***form drag***, ***skin friction***, and ***interference drag*** [**UAV, 2005**].

Form drag is caused by the frontal area of the airplane components being exposed to the airstream. Figure (2.12) illustrates how the side of the airfoil is exposed to the airstream. Skin friction drag is caused by air passing over the airplane's surfaces, and it increases considerably if the airplane surfaces are rough or dirty [**Atk, 2004**].



(a) Form drag **(b)** Skin drag

Fig (2.12) Form drag and skin friction drag

Interference drag is caused by interference of the airflow between adjacent parts of the airplane such as the intersection of wings and tail sections with the fuselage. Fairings are used to streamline these intersections and decrease interference drag [**Tot, 1999**].

2.7 Lift Force

Lift is an arising force that rises the airplane due to the speed effect. The effects of lift and drag can be determined analytically by employing force vectors as shown in Figure (2.13-a), if the lift vector is represented by the line *AB* and the drag vector by the line *AD*, then the resultant of the two forces is *AC*. The resultant force is the combined net result of the two forces of lift which acts upward and perpendicular to the relative wind, and the drag which is parallel to the relative wind, Figure (2.13-b) illustrates the vector components of both lift and drag [**And, 2000**].

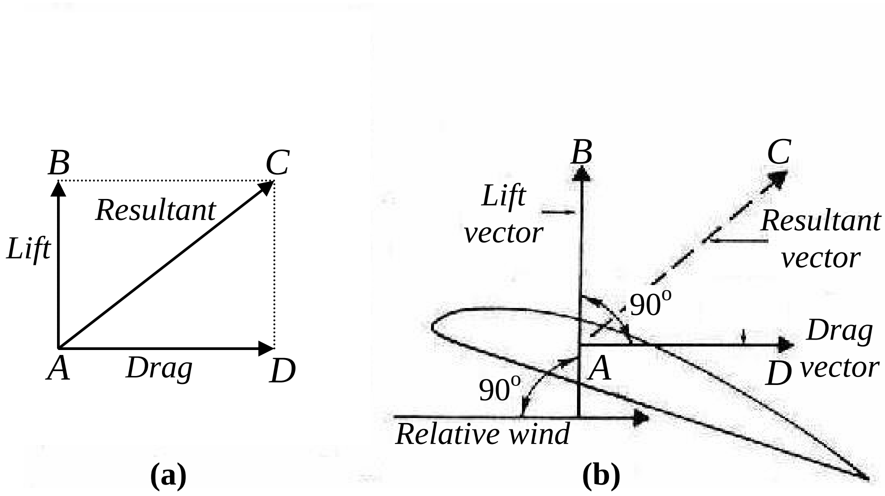


Fig (2.13): Relationship between relative wind, lift, and drag.

As shown in Figure (2.13-b) the resultant of lift and drag have the direction and magnitude equal to that of the force that created by the difference in pressure between the top surface and the bottom surface of an airfoil [**Kue, 1986**].

As the angle of attack of an airfoil increases, so do the components of lift and drag, the drag will increase in proportion more than the lift will. The ratio of lift to drag at any angle of attack is a measure of the flight efficiency because lift is a beneficial force and drag is a detrimental force. However, drag must be accepted as a necessary evil to produce lift [**Aut, 1987**].

There are other factors affect the lift force alone, these parameters depend on the geometrical design of the wing and its surrounding conditions, in the following sub-sections a brief discussion about these factors is given.

2.7.1 Wing Area

One of the factors that determine the total lift of an airfoil is the area of the surface exposed to the airstream. If the area is small, the region of pressure differential is small and there is little lift. On the other hand, if the area is large, the region of pressure differential is great and there is a large amount of lift [**Fri, 1999**].

2.7.2 Airflow Velocity

A positive angle of attack causes an increase in velocity and a decrease in pressure on the upper surface of a wing, and a decrease in velocity and an increase in pressure on the lower surface. This is in accordance with Bernoulli's principle [**Sim, 1987**].

If the air flows slowly around the airfoil, a certain amount of lift is generated. If the velocity of the airstream increases, the pressure differential increases and the lift increases. Velocity means the airspeed with respect to the airfoil [**Dre, 1999**].

2.7.3 Air Density

On hot days the density of the air is less than on cold days, while on wet days, the density of the air is less than on dry days. Also, density decreases with altitude. When the density is low, the lift will also be comparatively low. Therefore, greater airspeed is required for a particular airplane when the density is lower. The airspeed must increase as the density decreases in order to maintain the airplane at the same angle of attack [**Saf, 1964**].

2.8 Forces in Flight

The major forces acting on a flying airplane are lift, weight, drag, and thrust. The weight acts vertically downward toward the center of gravity of the airplane. The lift acts in a direction perpendicular to the direction of the relative wind from the center of pressure. In straight and level flight the lift and weight must be equal. When the airplane is flying at a constant speed, the forces thrust and drag must be equal. Thrust is provided by the engine and propeller or by high-velocity gases ejected from the tail pipe of a jet engine. Figure (2.14) illustrates the forces acting on an airplane flying in straight and level flight at a constant speed. In this case all the forces are in balance [**Sim, 1987**].

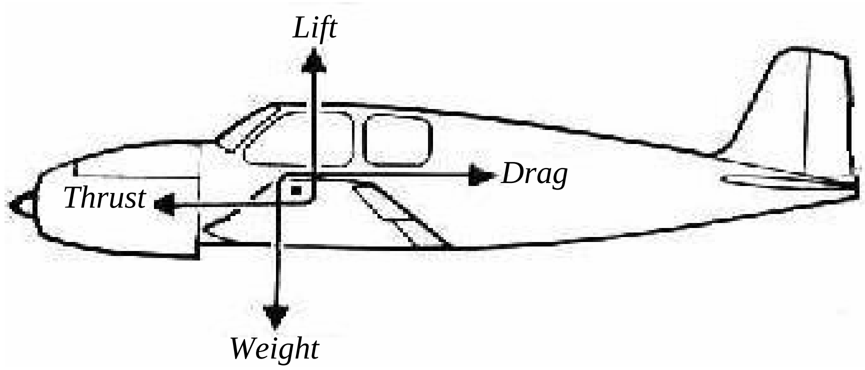


Fig (2.14) Forces on an airplane

Thrust is merely the force that drives the airplane forward, it is defined as the forward directed pushing or pulling force developed by an airplane engine. Drag is the force which opposes the forward motion of the airplane. The total drag of the airplane is in opposition to thrust [**Kro, 1980**].

Assuming that the airplane is flying straight and level in calm air, the drag acts parallel to the direction of the relative wind. As long as thrust and drag are equal, the airplane flies at a constant speed. If the engine power is reduced, the thrust is decreased and the speed of the airplane is reduced. If the thrust is lower than the drag, the speed of the airplane become less and less until it finally become lower than the speed required to maintains level flight then airplane will descend. Conversely, if the power of the engine increased, the thrust is increased and the airplane gains speed (accelerates), the drag increases until eventually become equals to the thrust, then the airplane flies at a constant speed. These facts indicats that "*in straight and level flight at a constant speed the vector sum of the components of the forces acting on an airplane equals zero*" [**Meg, 1988**].

2.9 Loads in Flight

During level flight the forces exerted on an airplane are at a minimum. A changes in speed during straight flight does not produced any appreciable change in load, but when a change is made in the airplane's flight path, an additional load is imposed upon the airplane structure. ***Load factor*** is the ratio of the total load supported by the airplane's wing to the actual weight of the airplane and its contents. Two effective cases in which load factor is affected [**Nee, 1987**], they are

2.9.1 Climbing Effect

During the flight time the airplane is curved, because of the inertia the airplane attempts (i.e., force itself) to follow straight flight. This tendency to follow straight flight rather than curved flight generates a force known as centrifugal force which acts toward the outside of the curve [**Meg, 1988**].

Curved flight produces a positive load (downward) as a result of increasing the angle of attack and, consequently, the lift. Increased lift always increases the positive load imposed upon the wing. Once the angle of attack is established, the load remains constant [**Pat, 2000**].

2.9.2 Turning Effect

In turn, the load factor on an airplane increases similarly to those experienced in pulling out of a dive. In turn of 45^{o} bank, the gravity makes to pull the airplane down with a force of 1 *g* (*g* refers to the pull of gravity) while the centrifugal force is pulling the airplane horizontally with a force of 1 *g*. When these forces are resolved, a resultant of 1.41 *g* is found. This means that the wing is required to carry 1.41 times the weight of the airplane, i.e. the load factor is 1.41, in 60^{o} bank the load factor is 2, 70^{o} bank makes the load factor is nearly 3, and an 80^{o} bank makes it almost 6. It is obvious that an airplane should not be turned with a bank so steep such that the safe load factor of the airplane is exceeded [**Nee, 1987**].

2.10 Airplane Axes

While being supported in flight by lift and propelled through the air by thrust, an airplane is free to revolve or move around its three axes, namely the longitudinal axis, the lateral axis, and the vertical axis. These axes are illustrated in Figure (2.15) [**And, 2000**].

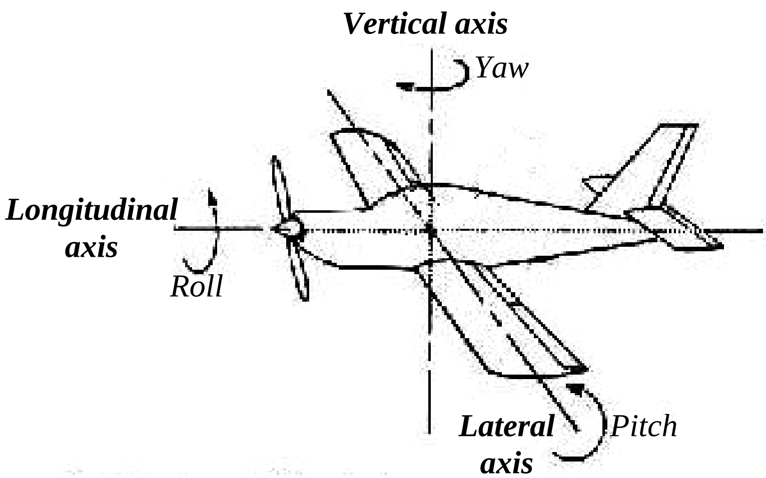


Fig (2.15) Axes of the airplane.

The axis extends lengthwise through the fuselage from the nose to the tail is the longitudinal axis. The axis extending through the fuselage from wing tip to wing tip is the lateral axis. The axis which passes vertically through the fuselage at the center of gravity is the vertical axis. During flight, an airplane is rotated about the three axes by means of the three primary flight controls. The aileron controls roll about the longitudinal axis, the elevators controls pitch about the lateral axis, and the rudder controls yaw about the vertical axis [**Pal, 1985**].

2.11 Airplane Stability

Because of the ability of the airplane to revolve about its three axes, all airplanes must poss stability in varying degrees for safety and ease of operation. Stability is the inherent ability of a body to develope a force tends to return the body to its original position. There are four kinds of stability [**And, 2000**], they are

2.11.1 Static Stability

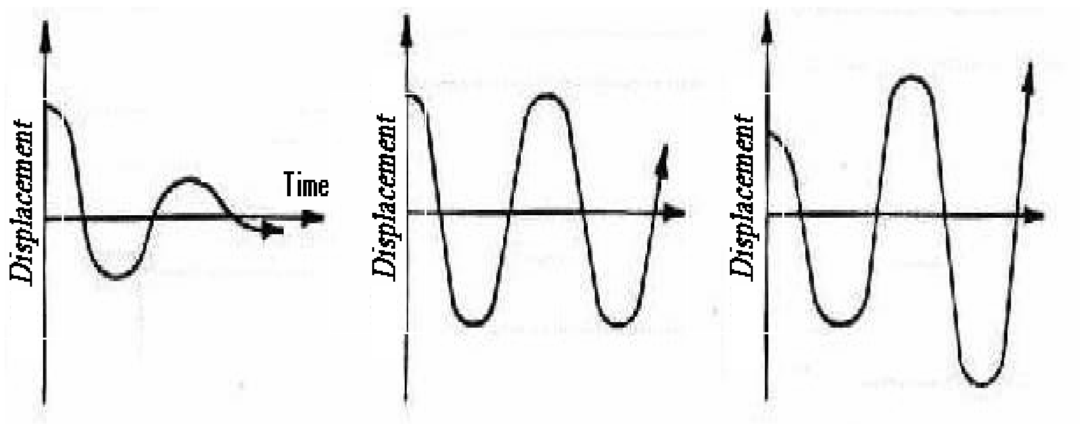
An airplane is in a state of equilibrium when the sum of all forces and all moments is equal to zero. When an airplane is in equilibrium, there are no accelerations of flight. If the equilibrium is disturbed by a gust or deflection of the controls, the airplane will experience acceleration because of the unbalance of moment or forces [**Kro, 1980**].

The static stability of an airplane is defined by the initial tendency to return to equilibrium conditions when the airplane is slightly disturbt from equilibrium. If an object is disturbed from equilibrium and has the tendency to return to equilibrium then a ***positive static stability*** exists. If the object has a tendency to continue in the direction of disturbance then a ***negative static stability*** (sometimes called static instability) exists. If the object subjected to a disturbance has neither the tendency to return and nor the tendency to continue along the disturbance direction then ***neutral static stability*** exists. This is an intermediate condition which could occur when an object displaced from equilibrium remains in equilibrium along the displaced state [**Tay, 1999**].

2.11.2 Dynamic Stability

Dynamic stability is determined by the resulting motion with time. If an object is disturbed from equilibrium, then the time history of the resulting motion is described by the dynamic stability of the system [**Nel, 1989**].

Dynamic stability is the property which dampens the oscillations set by a statically stable airplane. Enabling the oscillations to become smaller and smaller in magnitude until the airplane eventually settles down to its original condition of flight. Figure (2.16) illustrates the relationship of dynamic and static stability. The existence of static stability does not necessarily guarantee the existence of dynamic stability. However, the existence of dynamic stability implies the existence of static stability. An airplane must demonstrate the required degrees of static and dynamic stability if it is to be operated safely [**Ett, 2002**].



(a) Positive stability. **(b)** Nuetral stability. **(c)** Negative stability.

Fig (2.16) Types of airplane's dynamic stability.

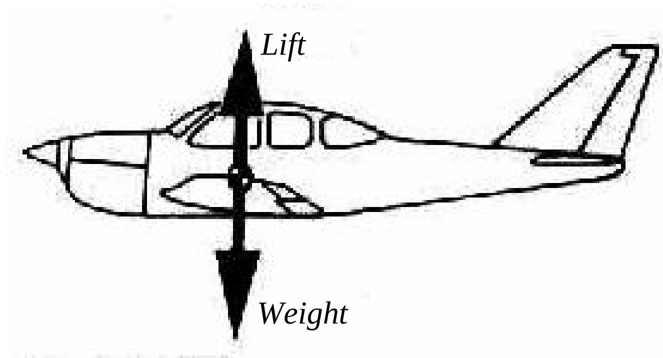
2.11.3 Lateral Stability

Lateral stability is the stability of an airplane about the longitudinal or roll axis. An airplane that tends to return to a wings-level attitude after being displaced from a level attitude by some force such as turbulent air is considered to be laterally stable. If the airplane rolls slightly to the right, the tips of the right wing moves downward and the lift of the wing increases. At the same time, the left wing lift is decreased because the angle with the horizontal plane is increasing. The airplane is subjected to more lift from the right wing and less lift from the left wing. This causes the airplane to roll back to the left to resume a level position [**Bar, 2003**].

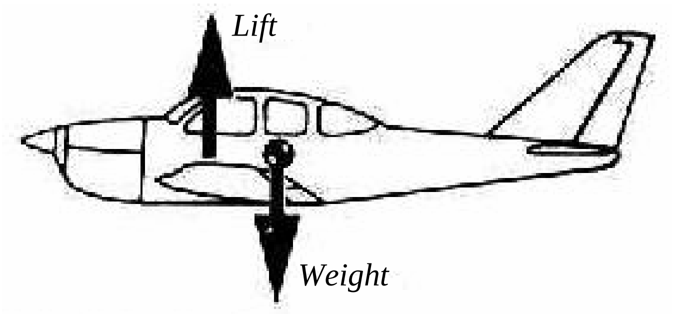
2.11.4 Horizontal Stability

The stability of an airplane about the lateral axis is horizontal. If the airplane is put into a dive or climb and then the control released, the airplane should return to level flight automatically. This depends on the location of the center of gravity with respect to the location of the center of lift, these two locations determine to a great extent the horizontal (longitudinal) stability of the airplane. Figure (2.17) presents different cases of these locations [**Liv, 2003**].

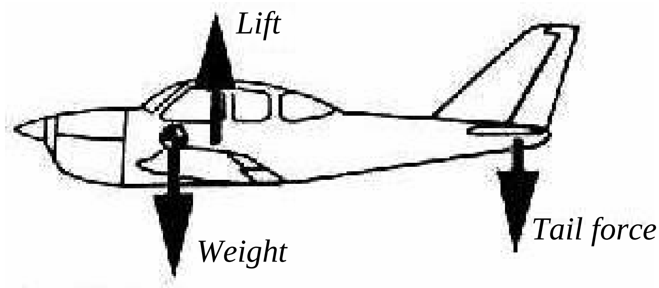
Figure (2.17-a) illustrates the neutral longitudinal stability, the center of lift is directly over the center of gravity or weight. Figure (2.17-b) shows the center of lift in front of the center of gravity. This airplane would display negative stability and an undesirable pitch-up moment during the flight. Figure (2.17-c) shows an airplane with the center of lift behind the center of gravity. This produces negative stability, some force must balance the down force of the weight [**Kro, 1980**].



a. Neutral stability



b. Negative stability.



c. Positive stability.

Fig (2.17) Types of airplane horizontal stability.

2.11.5 Directional Stability

It is the stability of an airplane about the vertical axis. This means that the airplane will return to the straight flight path after being turned (yawed) one way [**Nel, 1989**].

During its straight flight or turn if the airplane is yawed out of its flight path either by pilot action or turbulence, then the relative wind would exert a force on one side of the vertical stabilizer and return the airplane to its original direction of flight [**Fur, 1998**].

2.12 Airplane Controls

An airplane is equipped with certain fixed and movable surfaces or airfoils, which are provided for stability and control during flight. Each of the named airfoils is designed to perform a specific function in the flight of the airplane. The principal fixed airfoils are the wings, the vertical stabilizer, and the horizontal stabilizer [**Pal, 1985**].

The movable airfoils, called ***control surfaces***, are the elevator, aileron, and rudder. The elevator, aileron, and rudder are used to steer the airplane in flight to make it go where the pilot wishes it to go and to cause it to execute certain maneuvers. Figure (2.18) presents the three control surfaces of the airplane, each of them discussed with more details in the following [**Sim, 1987**]:

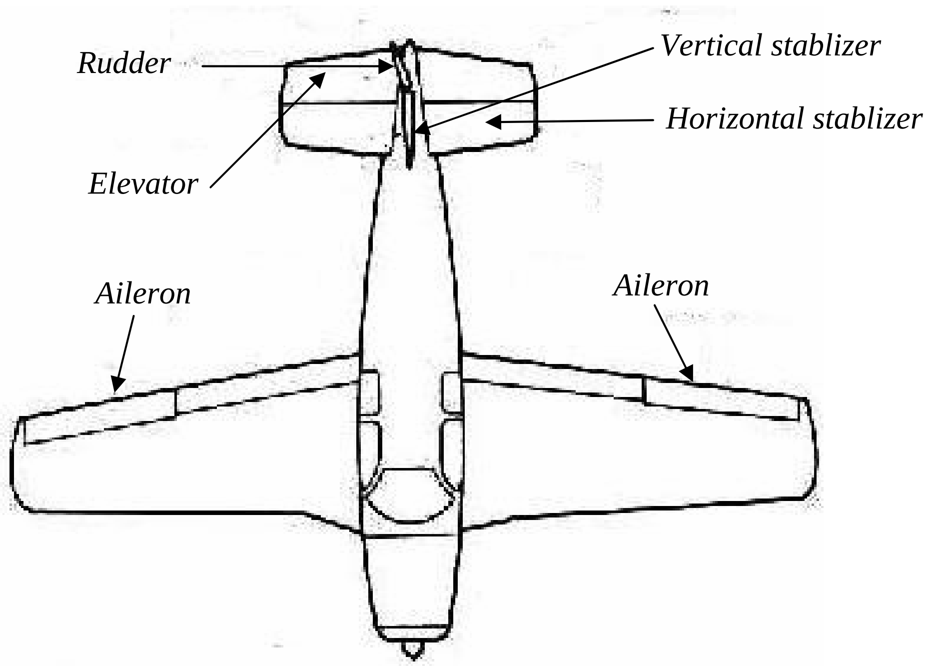


Fig (2.18) Control surfaces of an airplane.

2.12.1 Elevator

It is a control surface, usually, attached to the trailing edge of the horizontal stabilizer of an airplane, designed to apply a pitching moment on the airplane. A pitching moment is a force tending to rotate the airplane about the lateral axis to produce ***nose up*** or ***nose down***, as shown in Figure (2.19). When the elevators are raised, the force of the relative wind on the elevator surfaces tends to press the tail down, thus causing the nose to pitch up and angle of attack of the wings to increase [**Sim, 1987**].

The power developed by the engine determines the rate of climb of an airplane rather than the position of the elevators. As a matter of fact, if the elevators are held in a fixed position, the throttle alone can be used to make the airplane climb maintain level flight [**Kim, 1997**].

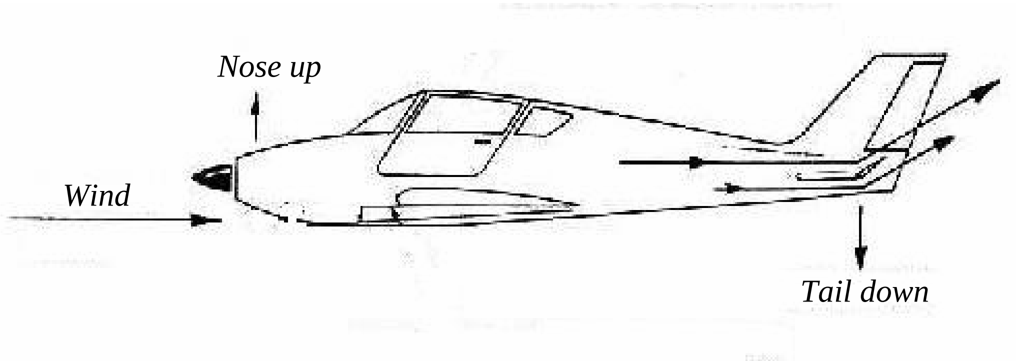


Fig (2.19) Action of elevator.

2.12.2 Aileron

An aileron may be defined as a moveable control surface attached to the trailing edge of a wing to control the airplane rolling (i.e., rotation about the longitudinal axis). They are rigged so that when one is applying an upward force to one wing, the other is applying a downward force to the opposite wing, as shown in Figure (2.20) [**Nai, 2001**].

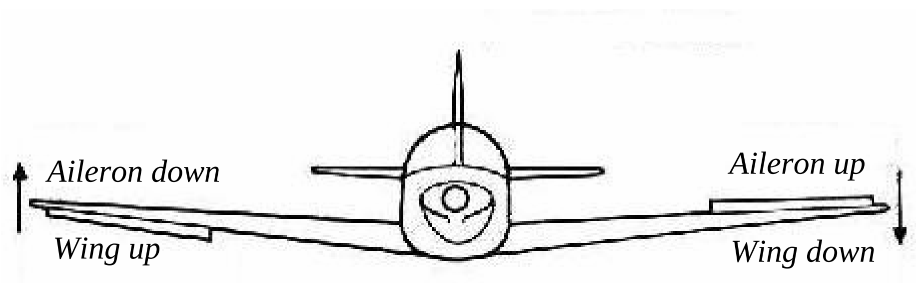


Fig (2.20) Action of aileron.

2.12.3 Rudder

It is a vertical control surface that is usually hinged to the tail post aft of the vertical stabilizer and designed to apply yaw moments to the airplane (i.e., to make it turn to the right or left about the vertical axis). When the rudder swings to the right, thus bringing an increase of dynamic air pressure on its right side. This increase in the dynamic air pressure on its right side, causes the tail of the airplane swing to the left and the nose turn to the right. The operation of a rudder is shown in Figure (2.21) [**Meg, 1988**].

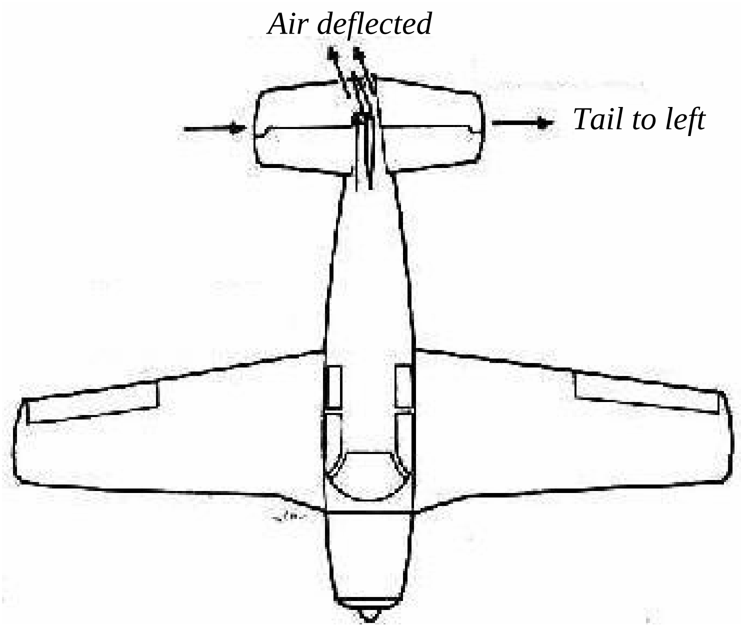


Fig (2.21) Action of the rudder.

From Newton's first law of motion, if the rudder is applied to an airplane in flight, the airplane will turn, but it will continue to travel in the same direction as before unless a correcting force is applied. Thus, with rudder only, one can find that the airplane ***skids***. In order to prevent this skid in a turn, aileron is used to bank the airplane. Therefore, properly executed turn require the use of all three of the primary controls [**Sar, 2002**].

2.12.4 Throttle

Throttle can be regarded as fourth controlling tool (surface). It represents the amount of the released energy from firing the fuel, which revealed as thrust. The thrust is the force that pushes the airplane forward. Therefore, the climbing/flight can be done, only, when the thrust is greater/equal to a specific value that make the lift force greater/equal to the weight of the airplane [**And, 2000**].

2.13 Stalling Conditions

When an airplane is in flight, there are number of flight conditions that may lead to a stall. This stall is not often encountered, because under ordinary conditions it is not necessary to pull an airplane up sharply enough to cause a stall [**Doh, 2000**].

The lift increases as the angle of attack increases. This angle corresponds to the burble point at which the streamline flow being to break down over the upper surface of the wing and burbling beings at the trailing edge. So, it is called ***stalling angle*** [**Nel, 1989**].

At angles greater than the angle of maximum lift, the lift decreases rapidly and the drag increases rapidly. By assuming both wing area and air density remains constant, as the angle of attack increases, the airspeed decreases too. Hence the least possible airspeed exists the angle of maximum lift. Point out that the ***stalling speed*** of an airplane is the minimum speed at which the wing maintains lift under a certain set of conditions [**McC, 1979**]. If an airplane is puled up sharply until its forward speed diminishes to a point where lift is less than gravity, then the airplane will begin to lose altitude. Stalls are more likely to occur during turns than in level flight, because greater lift is required to maintain level flight in a turn [**Sha, 2002**].

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DYNAMIC NAVIGATOR FOR UAV

A Thesis
Submitted to the College of Science of Al-Nahrain University as a Partial Fulfillment of the Requirements for the Degree of Doctor of Philosophy
in
Physics

By

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Dedicated to

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Certification

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