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Boosting Aircraft Efficiency by Reversing the Load on the Horizontal Stabilizer

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Abstract: This work aims at finding how reversing the direction of THS force improves aircraft performance. In most airplanes, the trimmable horizontal stabilizer (THS) is subjected to downward air force. This downward force acts in the same direction as the weight and opposite to the lift. The produced extra lift can be used to increase the payload or extend the range of the aircraft by carrying more fuel. The proposed design is based on shifting the wings location forward in order to make the force on the THS upward instead of downward. However, the stability of the airplane will be adversely affected. To address this issue, modern control theory is applied to the airplane elevator so as to maintain longitudinal stability. An airplane model based on longitudinal dynamics was used to investigate the stability of the airplane. Both current and proposed designs are simulated first without controllers and then with active controllers. The longitudinal dynamics' equations are used to design the controllers so as to make the aircraft stable. The payload gain due to the proposed design is calculated; For a typical airliner, it is found that up to 21% increase in payload can be achieved using the proposed design. The proposed design where the load on the THS becomes upward instead of downward results in improving flight efficiency; that is, we can choose between increasing payload, extending the range, reducing the thrust, or using a smaller wing, or any combination of these benefits. In all these cases, there is an operational advantage. This advantage is translated to cost savings or higher revenues.

Keywords: Aircraft control and stability, Aircraft fuel saving, Aircraft payload, Aircraft, Trimmable horizontal stabilizer

I. INTRODUCTION

Aviation is a world-wide activity that has a major economic, social, and political impact. It has become the backbone of international travel and fast shipping.

Due to the increasing number of daily flights around the world, enhancing flight efficiency would result in huge savings for aircraft operators [1-2].

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Many researchers tried to increase the efficiency of aircraft operation using various approaches; Park et al. [3] proposed an efficient design process based on Collaborative Optimization (CO). The method could be used to increase cruise range and the number of passengers of a derivative civil jet aircraft. Lim et al. [4]
developed a design procedure for the wing using the three-dimensional Euler equation and the beam analysis. Both wing aerodynamics and structure integrity were considered.

The wing weight was reduced by 25% and the power required was reduced by 3.4%, while maintaining the stability of the aircraft. However, the above studies were based on the existing design where the load on the tail is downward.

The aerodynamic effect of aircraft tail upward lift was investigated by Laitone [5]. He showed that minimum induced drag occurred with positive tail upload. He also stated that a typical transport aircraft could save up to 5% of its fuel consumption if tail lift is realized [3]. However, the controllability of the aircraft was not considered in either works. Aft tails and canards were compared by Goldstein and Combs [6]. They showed the effects of different configurations, where the advantage of canard as compared to upward load on aft tail was revealed, because the latter required a heavy aft tail to shift the center of gravity *CG* backward, which would offset the additional lift attained. However, neither stability nor the possibility of shifting the wing forward instead of shifting the *CG* backward was discussed.

In the present work, the idea of producing lift on aft tail (THS) is studied more rigorously. It should be indicated that the design of a typical transport aircraft implies that the tail is loaded downward, with the direction of the air force (*FT*) on the THS as shown in Figure 1 [7]. However, in the proposed design, the wings are shifted forward, and the direction of the air force (*FT*) on the THS will be upward, as shown in Figure 2. Thus, additional lift will be produced; In the current design (Figure 1), the total lift is *FL*, while in the proposed design (Figure 2) the total lift is *FL* + *FT*, where *FL* is the lift force of the wing. It should be highlighted that in the proposed design, the center of gravity (*CG*) will move slightly forward due to the shifted wing.

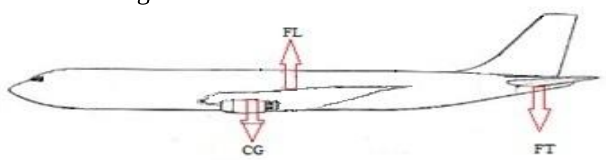

Figure 1: Current design of typical transport aircraft [7]

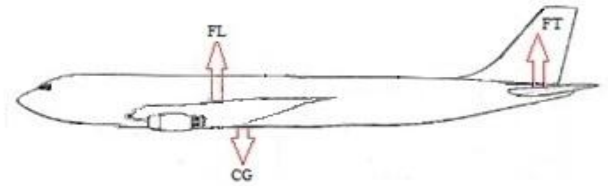

Figure 2: Proposed design where wings are shifted forward

Longitudinal stability of the aircraft is checked for both current and proposed designs. The proposed design increases lift capability but decreases the stability of the aircraft. To address this issue, modern control methods are utilized. In order to bring the load on THS upward, the wings location needs to be shifted forward. An airplane model based on longitudinal dynamics is used to investigate the stability of the airplane for both cases. Closed–loop control is introduced to the airplane elevator so as to maintain stability. Both current and proposed designs are simulated first without controllers and then with active controllers. In the first case, the current design is found more stable than the proposed design. In the second case, both designs were brought to stability using active controllers.

II. LONGITUDINAL STABILITY MODEL

Longitudinal stability of an aircraft refers to the aircraft's stability in the pitching plane (i.e. the plane which describes the position of the aircraft's nose in relation to its tail and the horizon), as shown in Figure 3 [8]. If an aircraft is longitudinally stable, a small increase in the angle of attack will cause the pitching moment on the aircraft to change, which in turn decreases the angle of attack. Similarly, a small decrease in angle of attack will cause the pitching moment to change which in turn increases the angle of attack [9]. Longitudinal static stability of an aircraft is significantly influenced by the distance (moment arm or lever arm) between the *CG* and the aerodynamic center (*AC*) of the airplane. The *CG* is established by the design of the airplane and influenced by its loading (i.e. payload, fuel, etc.), while the *AC* of the airplane corresponds to the one-quarter-chord point of the wing.

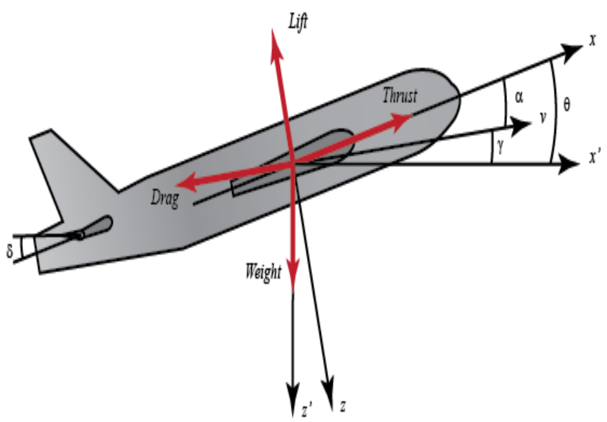

Figure 3: Longitudinal axis coordinates [8]

If the *CG* moves forward, the airplane becomes more stable (greater moment arm between the *AC* and the *CG*, and if it moves too far forward, it will be difficult to pitch the nose up, such as before landing. However, if the *CG* is too far aft, the moment arm between it and the *AC* diminishes, which in turn results in reducing the inherent stability of the airplane. Therefore, the operation manual for every airplane specifies the range over which the *CG* is permitted to move. Inside this range, the airplane is considered to be inherently stable, which means it will self-correct longitudinal (pitch) disturbances without pilot input.

In analysing stability, it should be noted that a body that freely rotates will always revolve around its center of gravity. The equations of motion of an aircraft are based on a moving coordinate system fixed to the aircraft as shown in Figure 3. The x-y-z axes are referred to as body axes. The x-axis is aligned with the longitudinal axis of the airplane. The longitudinal dynamics respond to changes in elevator deflection and thrust resulting from changes to the yoke and throttle. In Figure 3, for the plane to be in level flight, the velocity vector V must be horizontal. The plane is pitched slightly in order for the wings to develop sufficient lift to overcome gravity. The steady-state conditions are shown in Figure 3, including the speed *u* on the *x* direction.

Since the proposed aircraft design is unstable, a control strategy will be used to maintain longitudinal stability using the elevator. For this purpose, a straight and level flight is assumed. This means a steady-cruise at constant altitude and velocity. Consequently, the thrust, drag, weight and lift forces help each other to stay in balance in the *x*- and *z*-directions. It is also supposed that any change in pitch angle will not result in a change in the aircraft speed. This assumption can be justified noting that the time of wind disturbance and corrective elevator action is not long enough to change the speed significantly. In light of these propositions, the longitudinal equations of the aircraft motion can be stated as follows [10]:

$$\dot{\alpha} = \mu\Omega\sigma\left[-(C_L + C_D)\alpha + \left(\frac{1}{\mu - C_L}\right)q - (C_W \sin\gamma)\theta + C_L\right] \quad (1)$$

$$\dot{q} = \left(\frac{\mu\Omega}{2i_{yy}}\right)\left[[C_M - \eta(C_L + C_D)]\alpha + [C_M + \sigma C_M(1 - \mu C_L)]q + (\eta C_W \sin\gamma)\delta\right] \quad (2)$$

$$\dot{\theta} = \Omega q \quad (3)$$

By simplifying and solving the equations above we can obtain a simple state-space representation that describes the longitudinal dynamics of the aircraft as will be shown later.

III. CURRENT DESIGN OF TRANSPORT AIRCRAFT

In the current design of commercial airliners, static longitudinal stability is maintained by balancing the moments of aircraft CG and tail, as shown in Figure 1. The relationship between the CG and tail moments must be such that providing that the moments are primarily balanced and the airplane is unexpectedly nosed up, the tail moment will adjust to provide a restoring effect which will in turn bring the nose down again.

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Similarly, if the nose of the plane is directed down, the adjustment in tail moment will bring the nose back up. The summation of balancing moments of the airplane is:

$$\Sigma M = I\alpha \quad (4)$$

The moment produced by wing lift (M_L) and the moment produced by THS (M_T) can be estimated using Figure 1. Taking CCW direction as positive, an expression for the total pitching moment M about the CG position can be written as:

$$M = Lx - Me - T(lt + x) \quad (5)$$

$$M = M_L - M_E - M_T \quad (6)$$

In order to represent the desired pitch rate in terms of the sum of moments, we use the following equation:

$$\ddot{\theta} = \frac{M_L - M_E - M_T}{I} \quad (7)$$

Assuming the aircraft is in steady and level flight, this leads to $M_E = M = 0$. Therefore $M_L = M_T$

A. Open-Loop Control of Current Design

The transfer function of the current design is needed in order to perform the stability analysis. The longitudinal equations of motion for a typical commercial jet are [10]:

$$\dot{\alpha} = -0.313\alpha + 56.7q + 0.232\delta \quad (8)$$

$$\dot{q} = -0.0139\alpha - 0.426q + 0.0203\delta \quad (9)$$

$$\dot{\theta} = 56.7q \quad (10)$$

Laplace transform of the modeling equations is take to find the transfer function of the afore-mentioned system, It is noted that zero initial conditions must be assumed. The Laplace transforms of the above equations are shown below:

$$sA(s) = -0.313A(s) + 56.7Q(s) + 0.232\Delta(s) \quad (11)$$

$$sQ(s) = -0.0139A(s) - 0.426Q(s) + 0.0203\Delta(s) \quad (12)$$

$$s\Theta(s) = 56.7Q(s) \quad (13)$$

Using equations 11 to 13, the continuous-time state-space model is built as:

$$\begin{bmatrix} \dot{\alpha} \\ \dot{q} \\ \dot{\theta} \end{bmatrix} = \begin{bmatrix} -0.313 & 56.7 & 0 \\ -0.0139 & -0.426 & 0 \\ 0 & 56.7 & 0 \end{bmatrix} \begin{bmatrix} \alpha \\ q \\ \theta \end{bmatrix} + \begin{bmatrix} 0.232 \\ 0.0203 \\ 0 \end{bmatrix} [\delta] \quad (14)$$

$$y = [0\ 0\ 1] \begin{bmatrix} \alpha \\ q \\ \theta \end{bmatrix} \quad (15)$$

Therefore, the continuous-time transfer function is obtained as:

$$P(s) = \frac{\Theta(s)}{\Delta(s)} = \frac{1.151\,s + 0.1774}{s^3 + 0.739s^2 + 0.9215s} \quad (16)$$

The uncompensated open-loop step response is obtained by scaling the input in order to represent an elevator angle (δ) of 0.2 radians. The resulting open-loop pitch angle response is shown in Figure 4.

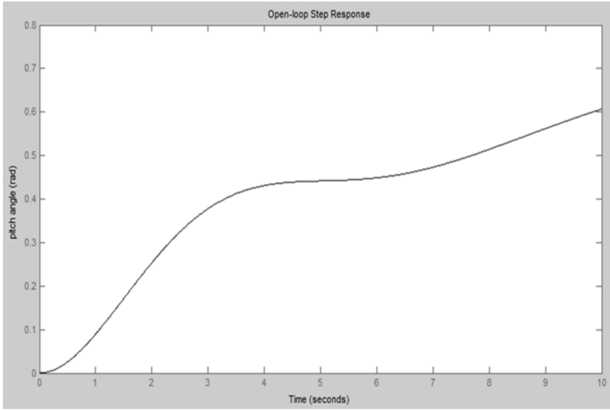

Figure 4: Open-loop response for the current design

It can be seen from Figure 4 that the open-loop response appears clearly unstable. Stability of the aircraft can be investigated by investigating the location of the poles of the transfer function. The poles are: 0.0000 + 0.0000i, -0.3695 + 0.8857i, and -0.3695 - 0.8857i. It is noted that one of the poles of the open-loop transfer function is on the imaginary axis which in turn designates that the free response of the system will not grow unbounded, but it will also not decline to zero. The other two poles are in the left-half of the complex s-plane.

Despite the fact that the free response will not grow unbounded, a system with a pole on the imaginary axis can grow uncontrolled when given an input, even when this input is bounded. This fact is in agreement with Figure 4. In this particular case, the pole at the origin behaves like an integrator. Therefore, when the system is given a step input, such as a wind disturbance affecting the aircraft, its output stays growing to infinity the same as an integral of a constant when the upper limit of the integral is made greater. This means the pitch angle will grow larger and the aircraft will be unstable.

B. Closed-Loop Control of Current Design

The transfer function for the closed-loop response is obtained as:

$$P(s) = \frac{\Theta(s)}{\Delta(s)} = \frac{1.151\,s + 0.1774}{s^3 + 0.739s^2 + 2.072s + 0.1774} \quad (17)$$

The closed-loop pitch angle response is shown in Figure 5.

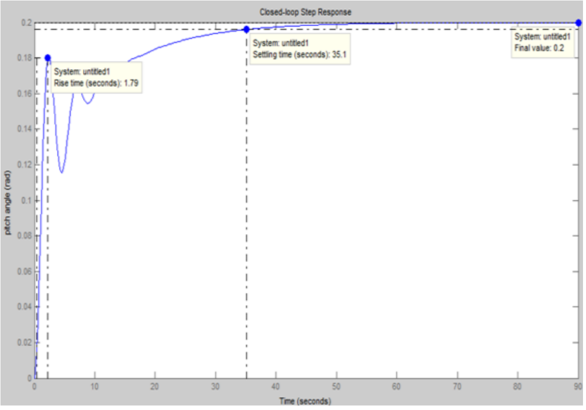

Figure 5: Closed-loop response for current design

By examining Figure 5, it is seen that adding feedback has made the aircraft stable. It is evident that the steady-state error is driven to zero with no overshoot. The poles are: -0.3255 + 1.3816i, -0.3255 - 1.3816i, and -0.0881, while the zero is: -0.1541. From these results, we can see that using a feedback signal can stabilize the current aircraft design. However, in an effort to smooth the damping response of the aircraft, a controller is introduced. The current design response using a PID controller is shown in Figure 6.

The characteristics of the response shown in Figure 6 are listed in Table 1. Taking these parameters into account, the PID controller provides a satisfactory performance of the aircraft's pitch [10].

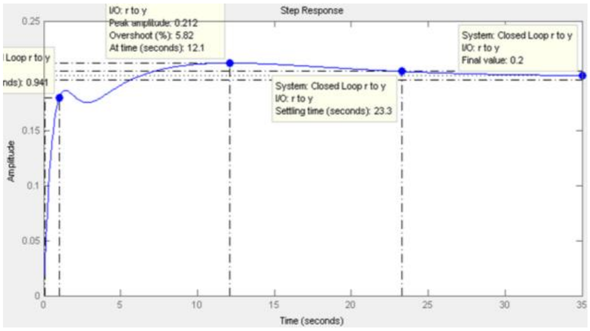

Figure 6: Current design response using PID controller

Table 1: Response of current design to PID controller

Overshoot	Rise Time	Settling Time	Steady-State Error
5%	1.2 sec	5 sec	0%

IV. PROPOSED DESIGN OF TRANSPORT AIRCRAFT

The proposed design is shown in Figure 2, where the force on the THS will become upward instead of downward. The cancelation of the downward force of the current design in addition to the created upward force will increase the amount of available lift. This is the main advantage of the proposed design, as the additional lift can be used to increase the payload or the range of the aircraft. Using Figure 2, the governing moment-balance equations can be written as:

$$M = -Lx - Me + T(lt) \tag{18}$$

$$M = -M_L - M_E + M_T \tag{19}$$

A. Open-Loop Control of Proposed Design

The same procedure used for the current design is employed here, and the continuous-time transfer function is found as:

$$P(s) = \frac{\theta(s)}{\Delta(s)} = \frac{1.151\, s + 0.5431}{s^3 - 0.113\, s^2 + 0.6548 s} \tag{20}$$

To represent an elevator angle ($\bar{\delta}$) of 0.2 radians, the uncompensated open-loop step response is obtained by scaling the input. The resulting open-loop pitch angle response is shown in Figure 7. In Figure 7, it is noted that open-loop response is unstable based on the location of the poles of the transfer function which are: 0.0000+0.0000i, 0.0565+0.8072i, and 0.0565-0.8072i. These values confirm that open-loop response is unstable. Because of that, active control is required to stabilize the aircraft.

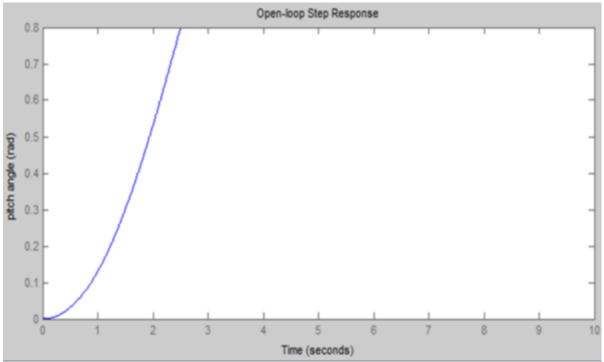

Figure 7: Open-loop response of proposed design

B. Closed-Loop Control of Proposed Design

The continuous-time closed-loop transfer function is determined as:

$$P(s) = \frac{\theta(s)}{\Delta(s)} = \frac{1.151\, s + 0.5431}{s^3 - 0.113\, s^2 + 1.806 s + 0.5431} \tag{21}$$

Using this function, the closed-loop pitch angle response is shown in Figure 8.

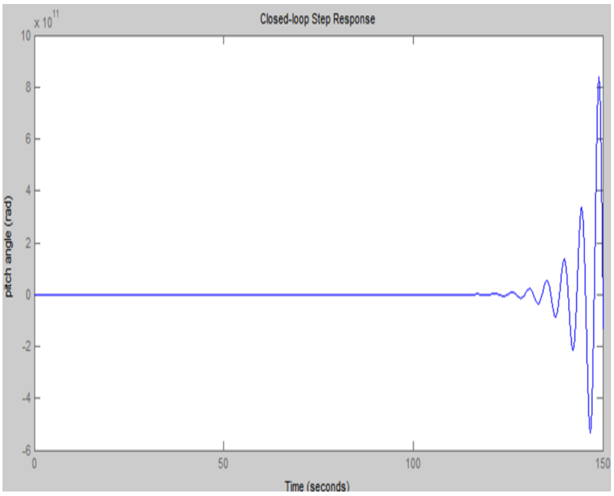

Figure 8: Closed-loop response of proposed design

In Figure 8, it is clear that after about 120 seconds the aircraft starts to oscillate and lose stability.

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To investigate further, the poles of the closed-loop response are determined as: 0.1981 + 1.3707i, 0.1981 - 1.3707i, and -0.2832 + 0.0000i, while the zeros are determined as: -0.4719. These values confirm the instability of the aircraft. Therefore, a proper controller is needed to establish aircraft stability.

V. ACTIVE CONTROL OF PROPOSED DESIGN

The proposed aircraft design carries clear operational advantage. However, the open-loop and closed-loop responses are found instable as shown earlier. In an effort to stabilize the proposed design, modern control methods are employed. Both PD and PID controllers were investigated to see how they can affect the stability of the proposed design. The controllers were installed into a closed-loop model of the proposed design. This loop is shown in Figure 9, where C(s) stands for controller and P(s) stands for plant, which in this case is the aircraft model.

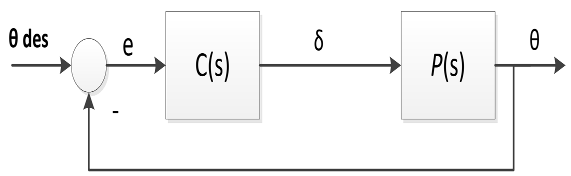

Figure 9: Active control of closed-loop model

VI. PD CONTROL OF THE PROPOSED DESIGN

The PD Controller is installed in a closed-loop model of the proposed design. The control system architecture is shown in Figure 10, where F stands for Pre-filter, C for Compensator, G for Plant, and H for Sensor. By tuning the controller parameters, stability can be achieved.

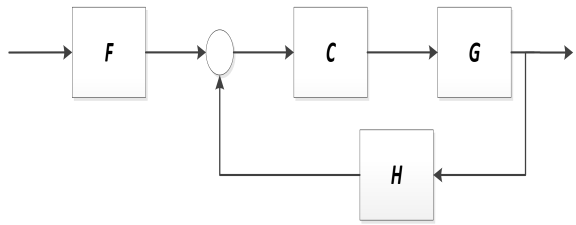

Figure 10: Block model of the proposed design

The transfer function for the PD controller is defined as:

$$C(s) = K_p + K_d s = \frac{K_p s^2 + K_d s}{s} \qquad (22)$$

The PD controller is tuned to make the aircraft model stable. Single-Input Single-Output (SISO) design is used where "F" is set to 0.2. As shown before, the closed-loop response for the aircraft without active controller is unstable. The system is tuned so that the response time is minimized. Bandwidth is set to 9.5 rad/s and phase margin is set to 65 deg. The resulting aircraft response is shown in Figure 11.

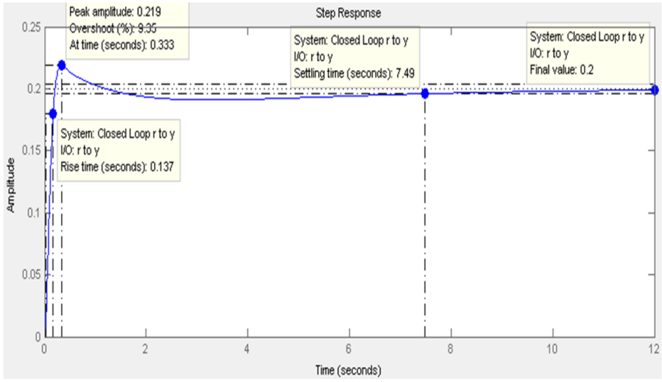

Figure 11: Proposed design response to PD controller

The compensator transfer function is:

$$C(s) = \frac{8.76s + 3.65}{0.036s + 1} \qquad (23)$$

The open-loop poles and zeroes of the function are -13.3+8.29i, -13.3-8.29i, and -0.762, while the gain margin is -36 dB and the zero is -0.423. From these results, it is shown that the aircraft pitch angle is stable with the characteristics listed in Table2.

Table 2: Response of proposed design to PD controller

Rising Time	**Maximum Overshoot**	**Settling Time**	**Steady-State Error**
0.137 sec	9.35%	7.49 sec	0%

A. PID Control of Proposed Design

The PID controller transfer function is:

$$C(s) = K_p + \frac{K_i}{s} + K_d s = \frac{K_d s^2 + K_p s + K_i}{s} \qquad (24)$$

In order to make the aircraft stable, the pitch angle "P" is tuned using SISO system tool. The same steps used for the PD controller were followed; Bandwidth was set to 7 rad/s and Phase margin was set to 76 deg. The compensator was updated with this data and the new response is shown in Figure 12. The open-loop poles are -0.904, -6.26, and -65.9 and the zero is -0.0642, while the gain margin is -32.5 dB. Consequently, the compensator transfer function is:

$$C(s) = \frac{5.88s^2 + 2.72s + 0.147}{0.014s^2 + s} \qquad (25)$$

Based on the above results, the aircraft pitch angle becomes stable with the characteristics listed in Table 3. Other types of controllers such as P, I, and PI were tested. However, they failed to stabilize the aircraft.

Table 3: Response of proposed design to PID controller

Rise Time	**Maximum Overshoot**	**Settling Time**	**Steady-State Error**
0.217 sec	8.26 %	6.76 sec	0 %

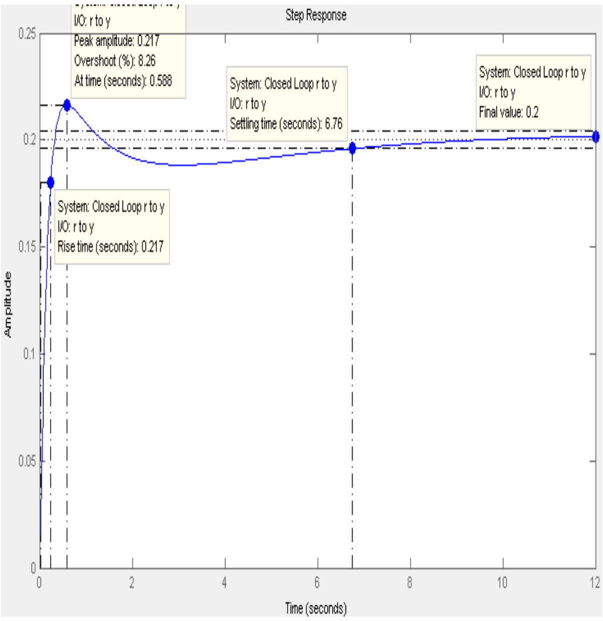

Figure 12: Proposed design response to PID controller

VII. ROBUSTNESS OF THE PROPOSED DESIGN

The proposed design is represented using a state-space block model as shown in Figure 10. The block parameters are obtained from the state-space matrix and the mathematical model. PID controller is used to test the system response. The disturbance is represented as a sine wave, where the mathematical representation of the sine wave signal is:

$$O(t) = 0.3\sin(2\pi) \tag{26}$$

The plot of the disturbance signal is shown in Figure 13 and the response is shown in Figure 14.

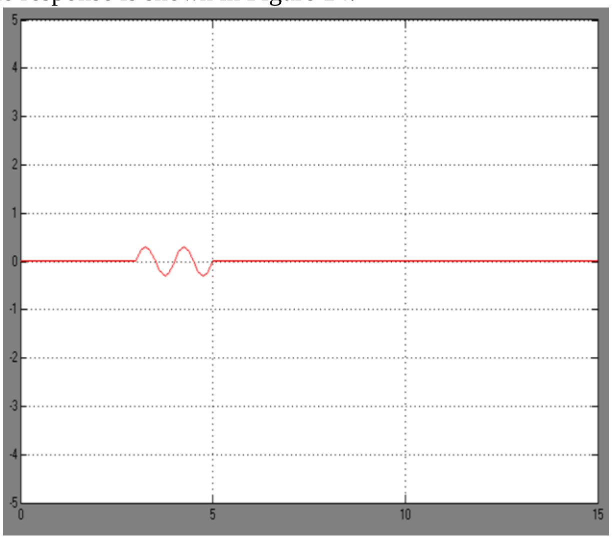

Figure 13: Sine wave disturbance signal

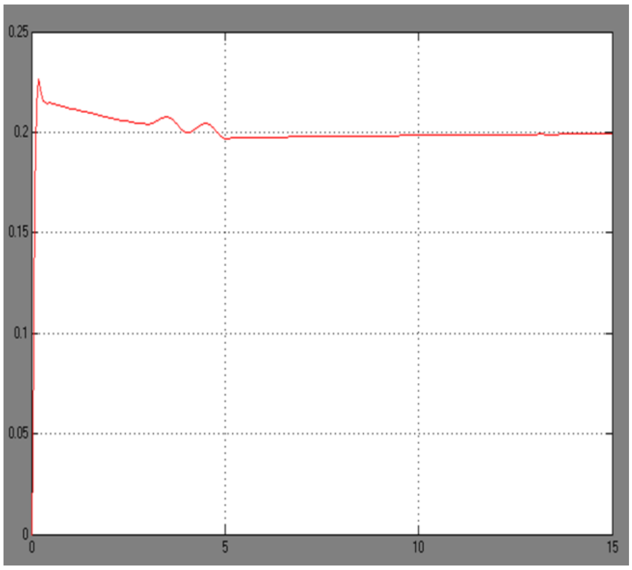

Figure 14: System response due to sine wave disturbance

In Figures 13 and 14, the proposed design performs very well. The aircraft remains stable and it instantly returns to its required steady attitude.

VIII. A PRACTICAL APPLICATION

In order to reveal the benefits of the proposed design and how it can make the aircraft carry more payload at the same fuel consumption; we worked on a real aircraft with its actual parameters. The chosen aircraft is the Airbus A330-200 powered with two Pratt &Whitney PW4000 turbofan engines [7]. The aircraft is assumed to be cruising in a straight and level flight at flight level "FL350" which equals 35,000 feet. The aircraft flies with its maximum weight of 230 tons and at a speed of Mach 0.8, which equals 979.2 km/hr. From A330 manufacturer manual [7], at these flight conditions, the fuel consumption in kg/hr for each engine is 3,249 kg/h, which equals 1,194 gallons per hour. The aircraft is modeled as a beam and the aerodynamic forces are applied to it as shown in Figure 15. Both aircraft current and proposed designs will be analyzed and compared.

IX. CURRENT DESIGN ANALYSIS

Similar to Figure 1, the distribution of forces specific to this aircraft is shown in Figure 15.

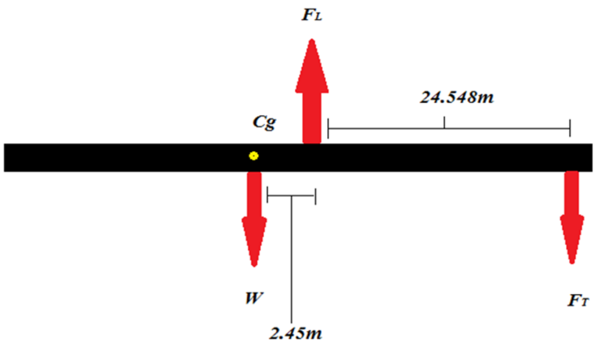

Figure 15: Aircraft forces in the current design

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The Lift force is defined as:

$$F_L = C_L\left(\frac{1}{2}\rho V^2 S\right) \quad (27)$$

Here, the force on the "THS" is downward which means it generates negative lift. Using Newton's 1st Law, the summation of forces affecting the aircraft is:

$$F_L - W - F_T = 0 \quad (28)$$

Balancing the moments about the aircraft *CG* yields:

$$\sum M_{CG} = 2.45F_L - (24.548 + 2.45)F_T = 0 \quad (29)$$

By solving equations 28 and 29, the forces affecting the aircraft are found as:

W = 2.244 MN
F_T = 223.96 kN
F_L = 2.468 MN

X. PROPOSED DESIGN ANALYSIS

In the proposed aircraft design, the distribution of forces affecting the aircraft is shown in Figure 16.

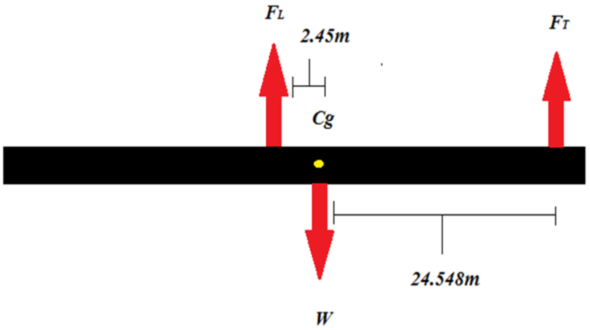

Figure 16: Aircraft forces in the proposed design

As shown in Figure 16, the air force on the THS (FT) is upward, which means it generates positive lift. In this configuration, the center of lift which is the point where FL affects the aircraft has been moved forward 2.54 meters. Therefore, the summation of forces affecting the aircraft is:

$$F_L + F_T - W = 0 \quad (30)$$

The summation of moments about the CG is:

$$\sum M_{CG} = 24.548F_T - 2.548F_L = 0 \quad (31)$$

Solving the above two equations leads to the following values:

W = 2.726 MN
F_T = 256.17 kN
F_L = 2.47 MN

The rest of flight conditions are the same as before, that is, the aircraft is cruising in a straight and level flight with the following conditions:

Altitude: 35,000 feet
Mass: 230 ton
Speed: Mach 0.8 (979.2 km/hr)
Fuel consumption: 6,498 kg/h,

There are two lifting forces; the wings (F_L) and the THS (F_T). Thus, the total lift force will be:

$$F_L^* = F_L + F_T = W = 2.726\ MN \quad (32)$$

The increase in the weight carrying capability is equal to:

$$W_{Proposed\ design} - W_{Current\ design} = 482\ kN \quad (33)$$

Since the aircraft mass (m) is equal to $\left(\frac{W}{g}\right)$, the corresponding mass increase is m = 49,134 kg. This means that the proposed design enables the aircraft to carry much more weight than the existing design, and without increasing fuel consumption. The percentage of this added weight relative to the original weight can be calculated as 49,134/230,000 = 21.4%. This increase can be used to carry more payload. If more payload is not desired, it can be used to carry more fuel, which enables extended range of operation. If extended range is not desired, the same weight can be kept while the proposed design will demand less lifting force; From equation 27, this will enable lower coefficient of lift (which means more freedom in designing the shape of the wing), lower speed (at higher levels, because you always can fly slower at lower altitudes, which means less fuel is burned), smaller wing surface area (which can save cost), or a combination of all these factors. It should be emphasized that all the above benefits are enabled without increasing fuel consumption.

XI. CONCLUSION & FUTURE WORK

Current fixed-wing aircraft design involves downward load on the THS. The idea of reversing the direction of this load results in increasing the total lift. This idea can be materialized by shifting the wings location forward. To investigate the proposed design, an airplane model based on longitudinal dynamics is used. The stability of the new design is compared to the current design, for both open-loop and closed-loop systems. The proposed design where the load on the THS becomes upward instead of downward results in improving flight efficiency; that is, we can choose between increasing payload, extending the range, reducing the thrust, or using a smaller wing, or any combination of these benefits. In all these cases, there is an operational advantage. This advantage is translated to cost savings or higher revenues. However, the proposed design renders the aircraft unstable. To address this issue, active controller such as PID or PD can be used to stabilize the aircraft using the elevator. To avoid any increased risk on passengers in case of controller malfunction, redundant controllers can be added. Further developments on this work may be achieved by trying other types of controllers to improve the stability characteristics.

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