Conceptual aerodynamic design of a Martian airplane

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Abstract
A stable tailless blended wing Martian airplane configuration with minimum folds for packing inside an aeroshell has been designed. The airplane weighing 120 kg in Mars is designed to fly at a cruise Mach number of 0.6 at an altitude of 1.5 km. A swept wing is designed with modified ESA40 reflex airfoil and winglets. Elevons and small all-movable twin vertical tails have been used for aerodynamic controls. The CFD studies indicate that the airplane nearly fulfills the design requirements. The trim lift coefficient is 0.5 and the achieved range is 461 km with an endurance of 53 minutes.

1. Introduction

A conventional approach to explore Mars includes the orbiting satellites or rovers moving on the Martian surface. On the other hand, a Mars airplane (Fig. 1) is a new approach to explore Mars [1]. It provides higher resolution than the orbiting satellites and larger spatial coverage than the rovers [1]. It offers an additional advantage of manoeuvring to specific locations of interest over other airborne platforms/ systems (e.g., balloons and airships). Design of a Marian airplane is challenging from aerodynamic design and engineering viewpoint. In this paper, a Martian airplane is designed to meet the specific requirements of performance and packaging. The effect of propulsive jet on the aerodynamic characteristics of the airplane is not assessed in the present study.

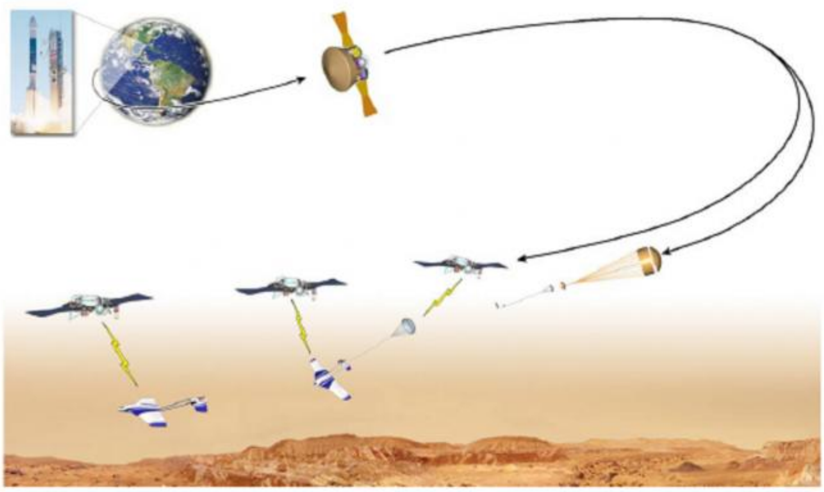

Figure 1: Typical Mars mission profile from launch to atmospheric flight in MARS (AIAA 2003-6610)

2. Design requirements

The design requirements of the Mars airplane is unique and differs from typical aircraft design requirements. For example, there is no take-off or landing requirements. But the range and payload requirements still exist for this science mission. Based on the typical earlier Mars mission profile, the following design requirements are set for the conceptual aerodynamic design of Mars airplane. (i) To design a stable tailless airplane for the cruise altitude of 1.5

km and Mach number of 0.6, (ii) To stow the airplane within 3.5 m diameter Viking aeroshell derivative, (iii) To minimize the number of folding events, (iv) To achieve a flight range of approximately 500 km, (v) To provide sufficient longitudinal and directional stability, (vi) To cruise with minimal use of trimming surfaces with on-board propulsion system.

3. Aircraft configuration design

3.1 Airfoil design

Thin Martian atmosphere [2] requires the airfoil with high maximum lift coefficient. In the present airfoil design, the requirement of lift coefficient at cruise with stable pitching moment coefficient was set as a design objective. The required design lift coefficient estimated for the 120 kg mass of airplane is 0.42. Hence, the selected airfoil must meet this design lift coefficient requirement. Moment reference centre (mrc) of the airplane was fixed at 50% of the airplane length. The objective is to get a stable pitching moment curve, which must trim (i.e pitching moment coefficient=0) at certain angle of attack where the lift coefficient is sufficient enough for cruise condition. Cruising near the trim angle of attack requires minimal use of trimming surfaces; therefore it meets one of the conceptual design requirements.

The airfoil analysis was carried out using public domain software XFOIL Version 6.96 [3]. The aerodynamic analysis at Mach number 0.6 and Reynolds number of 0.17 million for existing low Reynolds number and flying wing airfoils [4] showed that all the airfoils except ESA40 (reflex airfoil) do not meet the design objectives, specifically the stable trim requirement at positive angle of attack.

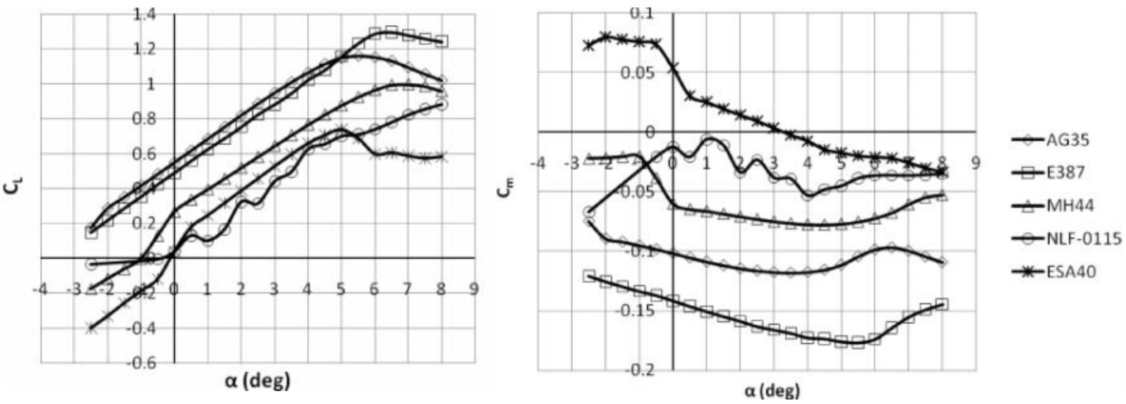

Figure 2: Lift and pitching moment coefficient of several low Reynolds number and flying wing airfoils

For the tailless aircraft, airfoils with small or negligible pitching moment are recommended in general. ESA40 reflex airfoil has stable pitching moment curve and positive pitching moment coefficient at zero degree angle of attack (α) (shown in Fig. 2) and trims at α= 3.3°. The maximum lift coefficient of ESA40 is 0.73, which is comparatively lesser than other airfoils. The reduced lift coefficient is due to the reflex nature of the ESA40 airfoil and it has to be increased further. Hence the ESA40 was selected as a baseline airfoil and carried forward for modification to increase its maximum value of lift coefficient. The ESA40 and modified ESA40 airfoils are shown in Fig. 3. The lift and pitching moment coefficient of ESA40 and modified ESA40 airfoils are shown in Fig. 4. Lift coefficient of modified ESA40 aerofoil is higher without compromising stability and natural trim requirement.

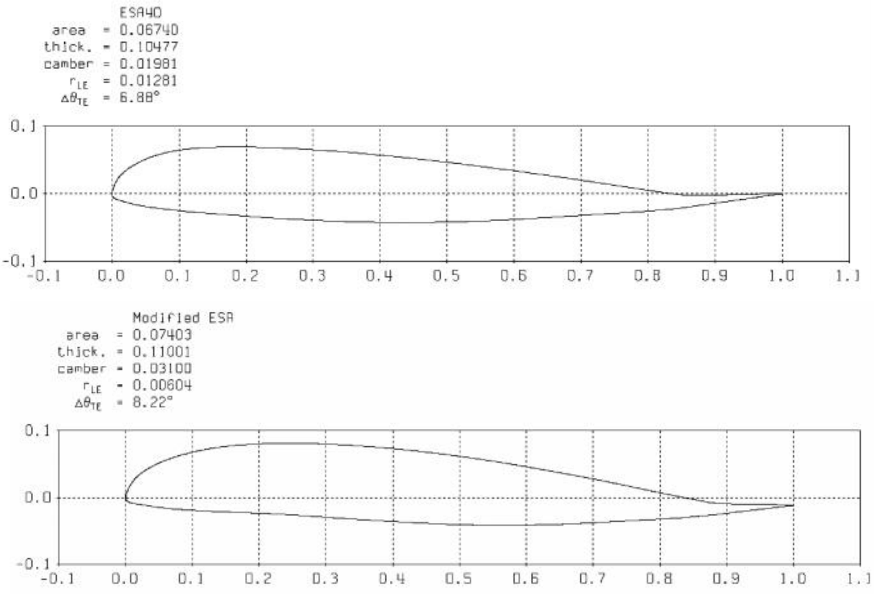

Figure 3: ESA40 and modified ESA40 airfoil shapes

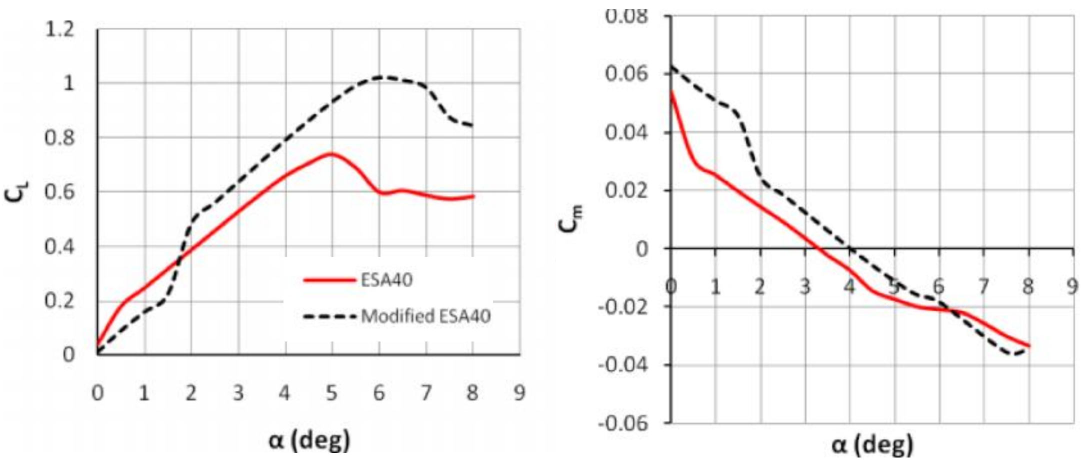

Figure 4: Lift and pitching moment coefficient of ESA and modified ESA airfoils

3.2 Wing, winglet and elevon design

The aeroshell packaging is one of the important constraints which restricts the airplane size and hence the wing area. The trailing edge of the wing is kept straight (i.e. no sweep) and the wing dihedral and wing twist were not considered in the conceptual design. The leading edge of the wing angle of 21.8° was arrived at by assuming the root chord of 2 m and a taper ratio of 0.5. The wing folding lines were set at 1.27 m from the wing center line considering packaging requirements. The geometric parameters of the reference wing are shown in Table 1. The low wing aspect ratio of 3.3 can be effectively increased by the addition of winglets because it can potentially increase the effective span and its planform efficiency. The winglet mainly reduces the wing-induced drag caused by the tip vortex and also provides yaw stability. The geometric parameters of the single winglet are given in Table 1. For pitch and roll control, two outboard elevons (one on each wing) are used. Since, the present airplane is configured for flying at stable natural trim angle of attack, the control surface deflection will be needed only for minor attitude corrections. The total elevon area is 5.3% of reference area and the chord is 16% of mean aerodynamic chord and the elevons are placed on outboard wing to avoid wing-folding lines.

Table 1: Wing and winglet geometric parameters

	Wing	**Winglet**
Span (m)	5	0.5
Root chord (m)	2	1
Tip chord (m)	1	0.5
Wing sweep angle (deg)	21.8º	45º
Wing reference area (m^2)	7.5	0.375
Wetted area (m^2)	13.6	0.61
Mean aerodynamic chord (m)	1.556	0.778
Taper ratio	0.5	0.5
Aspect ratio	3.3	0.75
Airfoil (reflex)	Modified ESA40	NACA 0010

3.3 Vertical tail design

The use of winglets for directional control of airplane increases the structural and control complexity. To eliminate this, two centrally mounted small all-moving vertical tails were designed and placed over the fuselage. The small all-moving vertical tails can be used in case of minimum control requirement. The tails have been carefully sized to fit within the aero shell. The geometric parameters of the vertical tail (single) are given in Table 2.

Table 2: Vertical tail geometric parameters

Span (m)	0.188
Root chord (m)	0.406
Tip chord (m)	0.259
Leading edge sweep (deg.)	53.83º
Area (m^2)	0.059
Wetted area (m^2)	0.089
Airfoil	NACA 0010

3.4 Fuselage design

In case of Mars airplane, the fuselage provides room for all the payloads, equipments, propellant tanks and thrusters. The use of conventional fuselage will lead to wing-fuselage blending issues and increased base drag. Therefore the fuselage cross-section was derived from the wing airfoil shape and it produces less drag compared to conventional fuselage. The fuselage also generates lift in addition to the wing lift. With the designed fuselage shown in Figure 5, two cylindrical propellant tanks of about 24 cm diameter and a cylindrical barrel length of 32 cm can be placed near airplane centre of gravity (c.g) to avoid large c.g movement due to propellant usage throughout the mission. In order to accommodate the propellant tanks near the airplane c.g the fuselage maximum airfoil thickness has been moved close to the airplane centre of gravity. In the aft portion of the fuselage 0.3 m length has been cut for thrusters mounting. The geometric parameters of fuselage are given in Table 3.

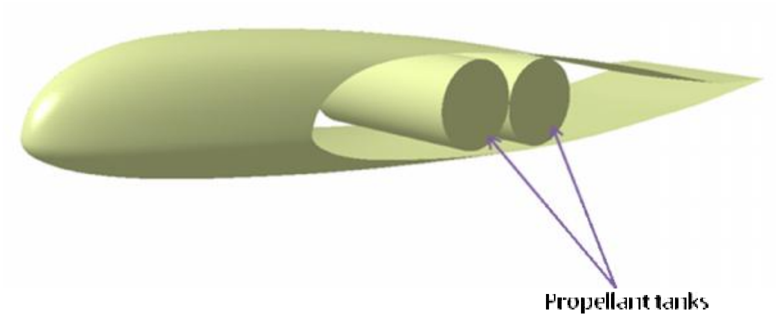

Figure 5: View of fuselage indicating the placement of propellant tanks

Table 3: Fuselage geometric parameters

Maximum thickness (m)	0.37
Length (m)	2.5
Wetted area (m^2)	2.34

3.5 Aeroshell packaging

CATIAV5 was effectively used to study the stowage of configuration inside 3.5m diameter Viking aeroshell. The airplane was placed upside down inside the aeroshell such that the down folded winglet portion of the airplane wing faces the canister part of the aeroshell. The perspective view of the airplane and the stowed airplane inside the aeroshell is shown in Figure 6.

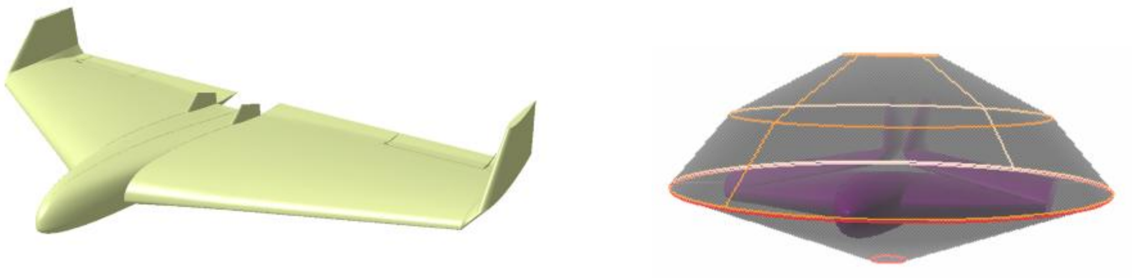

Figure 6: Perspective view of designed configuration and the configuration stowed inside the aeroshell

3.6 Bi-propellant thrusters

The Mars airplane requires 42 N of thrust to maintain flight. Two fully developed and qualified Aerojet R-6C thrusters (22N) were selected for the conceptual design. The total 44N thrust provided by the thrusters includes the margin of thrust to accommodate climbs, accelerations and turbulent conditions.

4. Numerical simulations for design verification

The aerodynamic characterization of the vehicle was ascertained after a number of CFD simulations using in-house Cartesian grid based RANS Navier Stokes solver employing k-ε turbulence model with wall function approach. Though, the Reynolds number is low in the Martian atmosphere for the given condition, turbulent flow simulation was carried out as a first step and laminar simulation will be carried out in future.

Figure 7 shows the Mach palette and pressure palette over the designed airplane at M=0.6 and angle of attack 6 deg. High flow expansion takes place near the leading edge of the wing and winglets. This leads to a high suction pressure of around –2 on the leeward side.

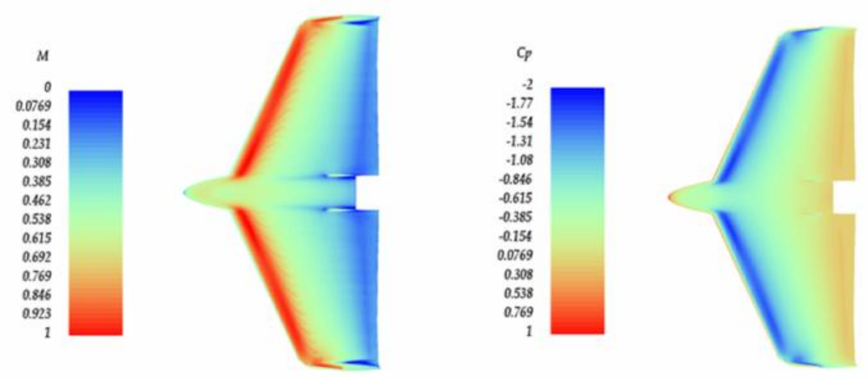

Figure 7 : Mach and pressure coefficient distribution over the configuration

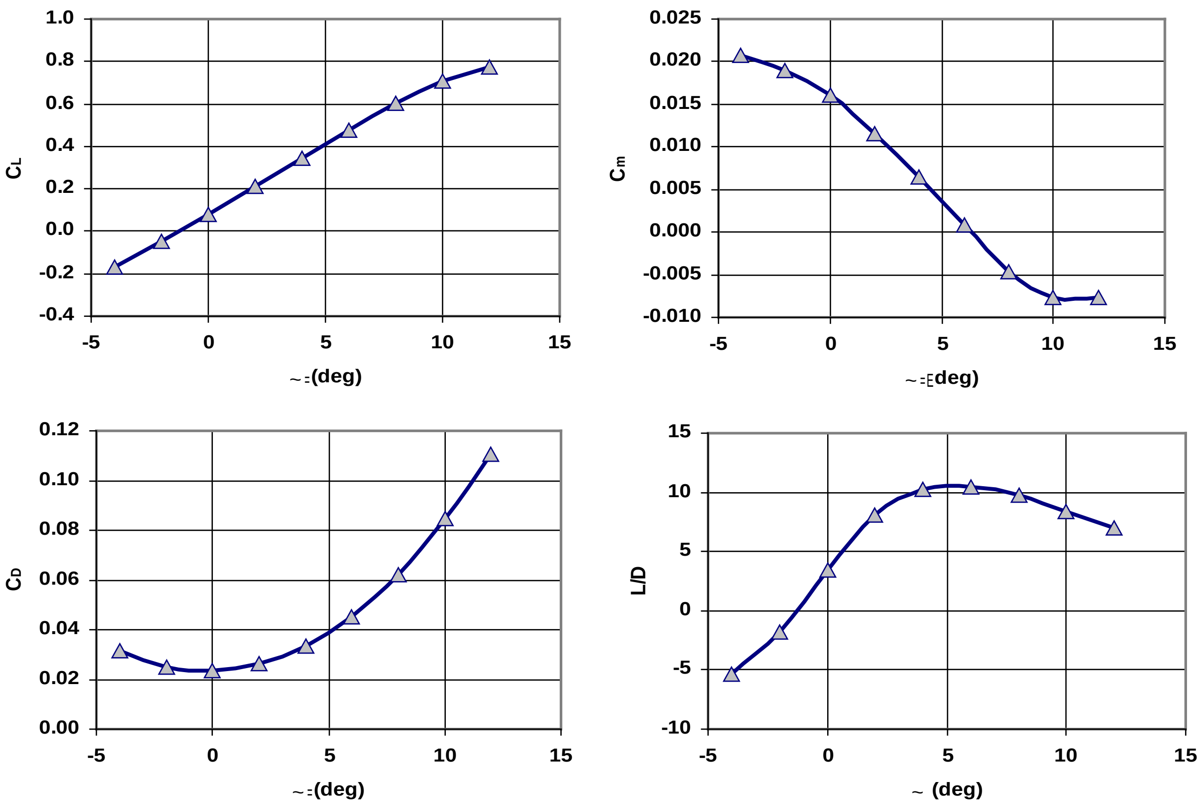

Figure 8: Computed aerodynamic coefficients of the designed martian airplane

Figure 8 shows the aerodynamic characteristics of the vehicle at M=0.6 and various angles of attack. The lift coefficient plot indicates that slope is constant upto ~8 deg. At higher angles of attack upto 12 deg, the lift coefficient slope reduces slightly. Pitching moment coefficient at angle of attack 0 deg is positive (pitch-up). The curve shows negative slope, indicating static stability. The natural trim angle of attack is 6.3 deg. Parabolic nature of the drag coefficient is visible in the plot. Minimum drag coefficient is 235 counts. Skin friction drag coefficient has been computed theoretically and added to pressure drag predictions. Lift to drag ratio increases with angle of attack, reaches a maximum of 10.5 at 6 deg and decreases with further increase in angle of attack.

Figure 9 shows the effect on elevon deflection on pitching moment coeffcient at various angles of attack. The plot indicates controllability of the vehicle with elevon deflection of ±10 deg.

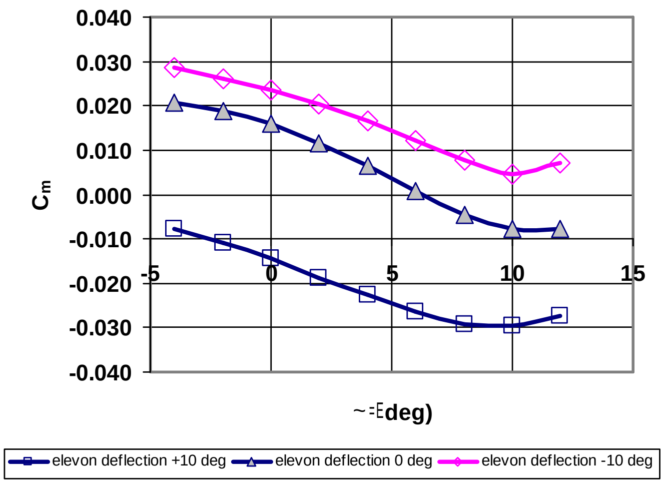

Figure 9: Longitudinal controllability of the designed configuration

Figure 10 shows the variation of ($C_{n\beta}$) directional stability derivative and dihedral effect ($C_{l\beta}$) at various angles of attack. The plot indicates that the vehicle is directionally stable (positive $C_{n\beta}$) and has dihedral effect (negative $C_{l\beta}$).

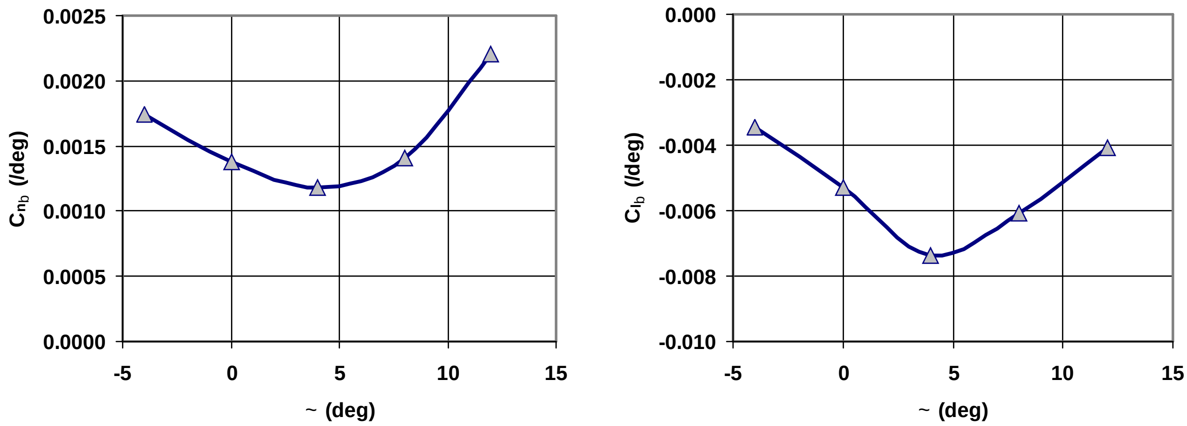

Figure 10: Lateral-directional stability of the designed configuration

5. Performance indices of the designed configuration

From the Cm plot shown in Figure 9, it can be noticed that the airplane trims at α=6.3. Cruising at this trim α requires minimum use of trimming surfaces. In the present study, the airplane cruise angle of attack is set at α=6.3°. The CL and L/D at the trim α is 0.5 and 10.5. The range and endurance of the airplane was calculated using the relations in [5]. Initial weight is 120 kg and weight at the end of the mission is 72 kg. Airplane velocity is 146.8 m/s for a Mach number of 0.6 at 1.5 km altitude. Specific fuel consumption is taken to be 1.7*10-3 per sec. The estimated range and endurance of the designed Martian airplane with a fuel weight of 48 kg is 461 km and 53 minutes.

6. Conclusion

A stable tailless blended wing Martian aircraft configuration with minimum folds (one on each wing) has been arrived at. The designed airplane can be stowed within 3.5 m Viking aeroshell derivative and has been verified using CATIA modeler. The CFD simulations of airplane using Cartesian grid based CFD solver for the cruise conditions indicate that the airplane has longitudinal stability and is longitudinally controllable. The vehicle is also directionally stable and has stable dihedral effect. The airplane trims at angle of attack 6.3° compared to 4° for the airfoil analysis using XFOIL. This difference in trim angle is due to the wing-body-tail combination. Cruising at this trim angle of attack minimizes the use of trimming surfaces. The present conceptual airplane meets the entire design requirement except the range of 500 km; the achieved range is 461 km with an endurance of 53 minutes.

7. Future work

The configuration can be further improved using formal optimisation techniques [6], considering the jet plume effect on aerodynamics.

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