The Zephyr: Manned Martian Aircraft

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ABSTRACT

This research was undertaken with the objective of designing a manned aircraft capable of flight in the Martian atmosphere. The resulting design, the "Zephyr", has a maximum payload of two space-suited astronauts and a range of 200 nautical miles, making it suitable for base-to-base transport, mesoscale scientific surveys, and search-and-rescue coordination, in anticipation of permanent human bases on Mars from which the aircraft can be deployed. The design was evaluated in terms of structure, stability, performance, and aeroelasticity. In addition, a low Reynolds number propeller was developed to meet thrust specifications. It has been concluded that a feasible design was achieved.

RÉSUMÉ

L'objectif de cette recherche était de concevoir un avion équipé pouvant voler dans l'atmosphère martien. Le "Zéphyr", la conception finale, a une charge maximale de deux astronautes en scaphandre, et une portée de 200 milles nautiques. En anticipant des postes permanents humains sur Mars, cet avion est considéré approprié au transport entre bases, les sondages scientifiques à mi-échelle, et la coordination des expéditions de secours. La conception a été évaluée au niveau de sa structure, sa stabilité, son exécution, et son aéroélasticité. De plus, une hélice à bas nombre de Reynolds a été développée pour répondre aux normes de poussée. Il a été conclu que le Zéphyr est réalisable.

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LIST OF SYMBOLS

C_{bhp}	Specific fuel consumption
C_{Do}	Parasitic drag coefficient
C_D	Total drag coefficient
C_L	Lift coefficient
$C_{l\beta}$	Roll non-dimensional stability derivative
$C_{n\beta}$	Yaw non-dimensional stability derivative
D	Drag
E	Endurance
g	Acceleration due to gravity on Mars
h	Altitude
K	Induced drag coefficient
L	Lift
M_{max}	Maximum level flight mach number
$N_{1/2}$	Number of cycles to half-amplitude
P	Power
r_{fuel}	Rate of fuel burn
R	Range
$t_{1/2}$	Time to half amplitude
T	Thrust or Period
V	Velocity
V_{max}	Maximum level flight speed
V_{mp}	Minimum power speed
V_{mt}	Minimum thrust speed
V_s	Stall speed
V_v	Vertical speed
V_x	Maximum climb angle speed
V_y	Maximum rate of climb speed
W_f	Final weight
W_{fuel}	Fuel weight
W_i	Initial weight
η_p	Propeller efficiency
λ	Stability-axes equation solutions

INTRODUCTION

Within the next hundred years it is conceivable that the exploration and subsequent colonization of Mars will take place. As time progresses, colonies will become larger and further apart, and the need for transportation will grow. The cheapest inter-base mode of transport will be trains, able to move large masses efficiently. For a time rockets could be used, but this would become prohibitively expensive. For fast versatile transport, aircraft will be needed.

Scientific surveys and reconnaissance can be conducted on the macroscale by satellite and on the microscale by ground rover. Many designs are available already for unmanned aircraft capable of flight in the Martian atmosphere to provide for directed mesoscale exploration. These aircraft will be valuable for increasing knowledge; however, there is always something to be gained by a manned aircraft survey.

Finally, as more area comes under the influence of human habitation, the potential for accidents increases. A helicopter or other hovering craft would be the ideal rescue vehicle, but this is not feasible in the low density Martian atmosphere. An airplane, however, could coordinate search-and-rescue operations directing ground transport.

The purpose of this research was then to prove that an aircraft could be designed capable of flight in the Martian atmosphere which could fulfill the roles of: 1) providing quick versatile inter-base transport; 2) conducting mesoscale scientific surveys and reconnaissance; 3) participating in search-and-rescue operations.

CONFIGURATION

Sketches of the Zephyr appear in **Figure 1**. The aircraft is a span-loaded two-crew biplane with the two fuselages joined by a horizontal stabilator. Two vertical tails are mounted as

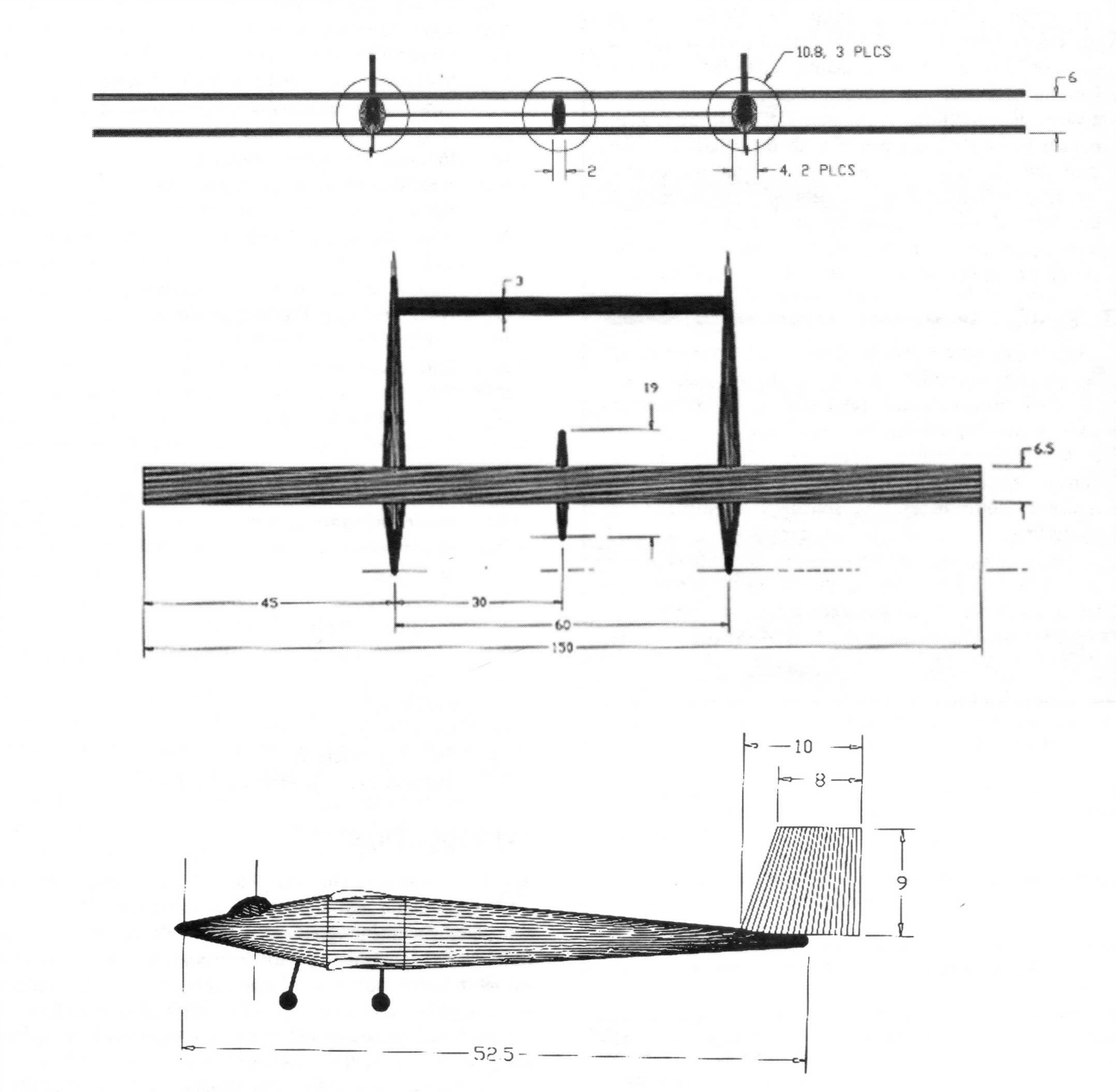

Figure 1.
Front, top and side view of the Zephyr (all dimensions in feet).

stabilators on each fuselage and roll control is provided by spoilerons on the upper wings. Each fuselage holds a crew station, power cell and fuel, and electric motor and propeller. A pod, centrally mounted on the wings carries another motor and propeller for a total of three propellers.

The initial design sizing was driven by the Martian environment (Pressure ~ 6 mbars, Density ~ 3 X 10^{-5} slug/ft^3, Temp. ~ 200 K, g ~ 12.3 ft/s^2), the payload requirement of two space-suited astronauts (160 kg, 134 Mars pounds), the range requirement of 200 nm, and a stall speed of no higher than 100 knots (for reasonable take-off lengths). Using historical data from powered sailplanes and extrapolated hydrogen-oxygen fuel cell specifications, the initial sizing resulted in a weight of about 2000 Mars pounds and wing area of about 2000 ft^2.

Looking more deeply into the structural design of the wing, it became clear early on that to support such a huge wing area, novel ideas were needed. Returning to a biplane design meant the wings were automatically close to half the length and therefore more manoeuverable and structurally lighter. The tradeoff was increased drag. Structural requirements and thus weight were further reduced significantly by span-loading the payload.

The wings, horizontal stabilator, and vertical tails were a structural-composite design consisting of graphite spars, graphite ribs, and kevlar epoxy skin reinforced in the shear web by graphite (see **Figure 2**). The wing aerofoil is a Selig 1223 high-lift low Reynolds number design[1]. The fuselage and central prop mount contain graphite tubes as longerons and truss pieces in a standard configuration. Kevlar/epoxy composite again forms the skin.

The choice of a connecting horizontal stabilator between the two fuselages was made because copious negative pitch moment was required to balance the large moment from the high-lift Selig main-wing airfoils. A stabilator was chosen as opposed to elevator/stabilizer combination because of ease of design and construction. The same logic was applied to the vertical stabilators which were sized to provide control in an engine-out scenario.

The chosen power plant was a hydrogen-oxygen fuel cell very similar to the power cell used for electricity on the space shuttle. The specifications for this cell were taken from Perez-Davis and Faymon[2] which extrapolated to post-2000 technology. Comparatively, a hydrazine engine could have supplied more power per weight but would have been impractical since fuel would have to be shipped from Earth. Electrolysed ground ice could provide the fuel for the Zephyr *in situ*.

A propeller was designed with an optimization program based on momentum theory incorporating the Martian atmosphere. (The program and design are described later in this paper.) The optimum propeller had ten blades with a diameter 5.9 feet, three of these propellers were required to provide the neccessary thrust to keep the aircraft airborne near the stall. Thus, the basic design of a biplane with two fuselages span-loaded and an additional central pod for an additional propeller took shape.

For a complete description of the Zephyr see Morrissette[3].

Structure

Structural Analysis

Structural analysis was performed for a 3g loading of the main wing, fuselage, and horizontal stabilator. In addition, the main wing back bending at cruise speed and 1g, along with the vertical tail bending at maximum-lift control-surface loading, and the main wing bending for a single gear at 2g landing was examined. The structural analysis showed that the design was robust enough to handle full control deflections, 3g loads, and a 2g landing. A safety margin of 50% was assigned to the loading so that the flight envelope was set to 2g. Because of symmetry, the aircraft could structurally resist the same negative loads, but the asymmetric airfoil would make this situation highly improbable.

A +/-2g flight envelope was deemed sufficient since the aircraft cannot experience high load factors without a quick loss of speed due to a power deficit. As well, the short period mode of 4 seconds, and seasonally calm Martian atmosphere, ensure that quick loading of the structure cannot take place. Therefore, the structure was considered to be safe and robust enough for the specified mission parameters.

Stability

Longitudinal Static Stability

The stick-fixed static margin was calculated to be 5.5% for conditions of full fuel and full payload at cruise, improving to 6% at landing speed. This means that the aircraft has average static pitch stability. A trim analysis showed a requirement for 5 degrees of stabilator deflection for trim at cruise. This was built in as the stabilator incidence angle, with the stabilator capable of +/− 10 degrees of deflection about this angle, guaranteeing full pitch control.

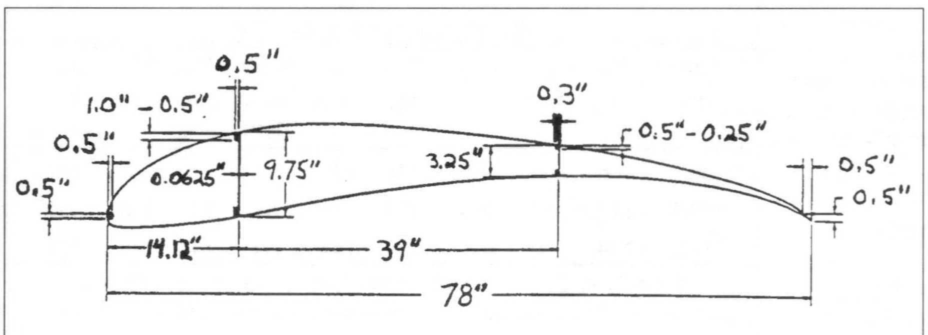

Figure 2.
Representative wing cross-section.

Lateral Static Stability

Though the static margin can be calculated for yaw and roll stability, the cg is usually placed to ensure pitch stability, and tail sizings are used to ensure lateral stability. The yaw and roll stability derivatives were calculated to be:

$C_{n\beta} = 0.0228$ 1/rad
$C_{l\beta} = -0.00358$ 1/rad

With the conventions defined in **Figure 3**, a positive value for $C_{n\beta}$ is stable and a negative value for $C_{l\beta}$ is stable, therefore the aircraft has static lateral stability. The leading contributors to the yaw stability were the vertical tails. The vertical tails were the only contributors to roll stability because the wing vertical positions were symmetric with respect to the vertical location of the cg and cancelled each other out.

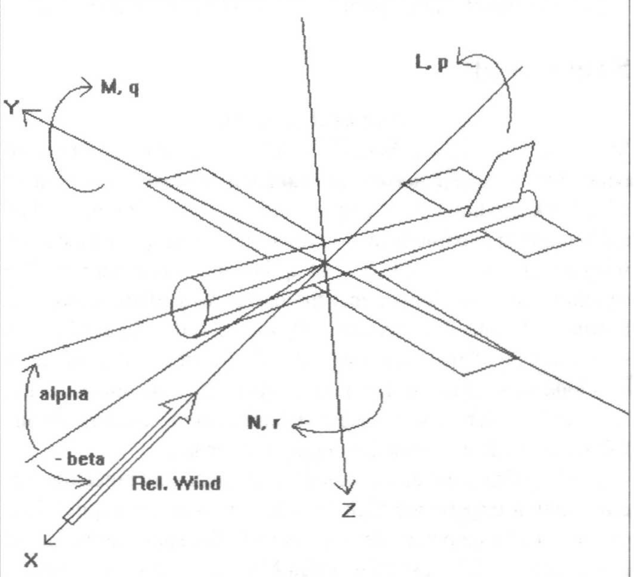

Figure 3.
Stability axes coordinate system.

The calculation to determine if it was possible to trim the aircraft with a non-functioning outboard engine was performed. This calculation showed that the aircraft could be yaw trimmed with 14° of vertical tail deflection at landing speed.

In order to take-off or land with a crosswind, an aircraft must be capable of sideslip. Calculations for this aircraft showed that at take-off or landing speed, the aircraft could maintain a 9.5° sideslip angle with full vertical tail deflection. This corresponded to a maximum cross-wind component of 16.5% take-off speed or 18 knots.

Longitudinal Dynamic Stability

Eigenvalue solutions to the longitudinal linear non-dimensional stability-axes equations gave the time-to-half amplitude, number of cycles to half amplitude, and period for each mode were:

Short-Period	**Phugoid**
$t_{1/2} = 4.08$ s	$t_{1/2} = 370$ s
$N_{1/2} = 0.0250$	$N_{1/2} = 3.71$
$T = 16.0$ s	$T = 99.8$ s

The time-to-half amplitude of the short-period mode indicates a slow response to changes in angle of attack. The period is long by comparison, however, so that the short period mode will damp out before any oscillations occur.

The phugoid mode characteristics were typical of larger aircraft. With a period of 100 seconds and a time-to-half amplitude of 6 minutes, the mode is long-lasting but slowly oscillating and thus easily controlled.

In summary, the Zephyr displays longitudinal dynamic characteristics much like a larger aircraft on Earth. It is longitudinally dynamically stable and should present no problems to the pilot.

Performance

The Martian reference atmosphere values used for the performance calculations were taken from Neunteufel[4]. It should be noted that these were average values which vary considerably with season and latitude. Also, "sea level" or 0 altitude on Mars is defined as an average topograpic height, so there are considerable tracts of land that may be as much as 6000′ below "sea level".

Drag Polar

The drag polar was represented by:

$$C_D = C_{Do} + K \bullet C_L^2 \tag{1}$$

where: $C_{Do} = 0.0225$
$K = 0.0241$.

Performance Characteristics

Table 1 shows various performance characteristics as a function of altitude. Algorithms for determining these values were taken from Raymer[5].

Time and Fuel to Climb to 3000 ft.

With the climbing differential equation:

$$dt = \frac{dh}{V_v} \tag{2}$$

Table 1.
Performance characteristics.

Altitude (ft)	V_s (kts)	V_{max} (kts)	M_{max}	V_{mt} (kts)	V_{mp} (kts)	Shaft Power (HP)	V_y (kts)	V_x (kts)	Sink Rate (ft/min)
0	106	204	0.48	158	120	57	120	94	656
6560	117	213	0.50	173	131	63	131	110	718
13120	128	220	0.52	189	144	69	144	125	788

From left to right, the values presented are: Stall Speed, Maximum Level Flight Speed, Associated Maximum Level Mach, Minimum Thrust Speed, Minimum Power Speed, Associated Minimum Shaft Power, Best Rate of Climb Speed, Best Angle of Climb Speed, and Gliding Sink Rate (if flown at V_{mt}, glide ratio is 21.4 : 1).

it was assumed that the vertical speed changed in a linear fashion:

$$V_v = V_{v_i} - a(h_f - h_i) \tag{3}$$

where the constant α was determined from the rates of climb at two altitudes, namely 0 ft and 3000 ft.

$$a = \frac{V_{v_f} - V_{v_i}}{h_f - h_i} \tag{4}$$

In this way, the solution to Equation 2 is:

$$\Delta t = \frac{1}{a} \ln\left(\frac{V_{v_i}}{V_{v_f}}\right) \tag{5}$$

from which one obtains the change in fuel weight as being:

$$\Delta W_{fuel} = r_{fuel}\,\Delta t \tag{6}$$

where r_{fuel} is the rate of fuel burn, assumed to be 25.1 lb/hr at full power (Mars pounds)[2].

The maximum climb speeds at 0 ft and 3000 ft were 448 and 422 ft/min respectively giving a value for a of -1.49×10^{-4} 1/s. Thus the time to climb to 3000 feet was 6.90 minutes and the weight of fuel burned was 2.89 pounds. Again, note that these are Mars pounds.

Sustained Turn Rate

Using an algorithm from Raymer[5], the turn rate at 3000 ft was calculated for maximum thrust at different flight speeds. This gave the sustained turn-rate envelope in **Figure 4**. Attempting to turn beyond the sustained turn- rate envelope is possible, but will result in a loss of speed. The constraints on such a manoeuvre were the stall limit ($C_L > C_{L\ max} = 2.1$) and the structural limit of 2 gs, which are both shown in **Figure 4**. Note that the corner speed which occurs at the intersection of the stall and structural limit represents the fastest possible instantaneous turn rate.

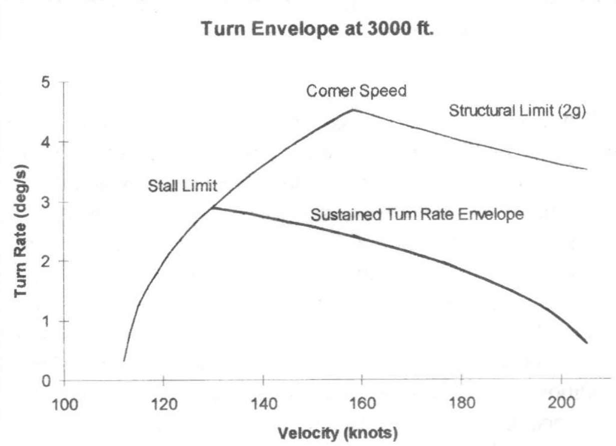

Figure 4.
Turn envelope at 3000 ft.

The maximum sustained turn rate is the greatest limitation of this aircraft. With available power, the maximum g that could be pulled in a turn was 1.36, corresponding to a bank of about 43°. This gave a turn rate of 2.89 deg/s, or a full 360° turn in 124 seconds with a turn radius of 4280 feet (or 29 wing spans). This improved slightly at "sea level" to 3.09 deg/s, or a 360° turn in 116 seconds with a turn radius of 4040 feet (or 27 wing spans).

Service Ceiling

The service ceiling is defined as the altitude at which the aircraft has a maximum climb rate of 100 ft/min. Assuming a velocity that is always at V_{mp}, the service ceiling was found to be 17,500 feet, taking into account a decrease of propeller efficiency to 60%.

V-n Diagram

The V-n diagram describes the operating envelope of the aircraft. The envelope is bounded on top and bottom by the structural limits of the aircraft (+/− 2 g) and the maximum C_L (2.1 positive lift, 1.0 negative lift) of the wing. The never exceed speed, V_{ne}, is limited by propeller characteristics. The V-n diagram appears in **Figure 5**. Note that V_{mp}, V_{mt}, and V_{max} were calculated for the cruise height of 3000 feet.

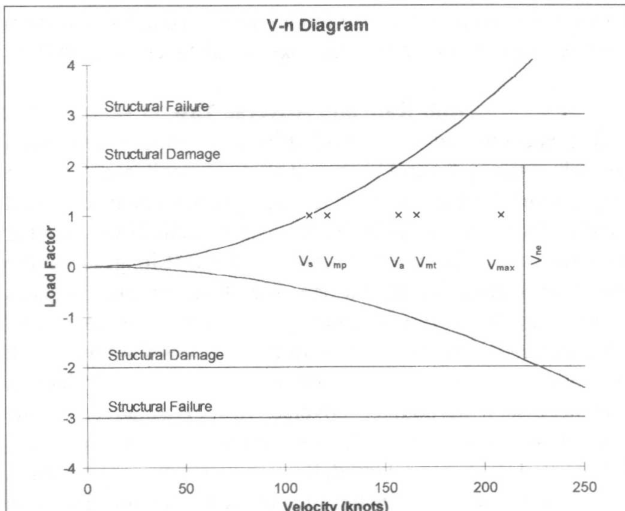

Figure 5.
V-n Diagram.

Take-Off Distance

The take-off distance was estimated by breaking up the manoeuvre into three segments: the ground roll, the transition, and the climb to obstacle height according to an algorithm from Raymer[5]. Assumptions made for the calculation were: 1) during the roll, C_L equalled 1.3[a]; 2) the thrust throughout the roll was calculated from max power at a velocity of 70% stall speed[b]; 3) the coefficient of friction was assumed to be 0.03, a value consistent with a dry concrete or asphalt runway; 4) during the transition, the aircraft assumes the maximum climb angle for the speed of 1.1 V_s by rotating at 1.1g[c]; 5) for obstacle clearance, the aircraft

[a] The C_L of the wing at 0 angle of attack.
[b] Recall $T = P/V = \infty$ at 0 velocity.
[c] 1.0g plus the centripetal acceleration.

was assumed to maintain the climb at 1.1 V_s until an altitude of over 50′ was achieved. The result was that for no wind and concrete runway conditions, the Zephyr required a minimum of approximately 20,000 ft (3.8 miles) of runway to take-off, and then a further 6100 ft to clear a 50 ft obstacle. This performance can be improved with the addition of a JATO for additional thrust; however, this length of runway is not unreasonable for an established base and would make the aircraft independent of external kit.

An independent calculation based on a method taken from PP. 236 to 242 of Jones[6] gave a take-off distance of approximately 25,000 ft, verifying the previous value.

Landing Distance

The landing-distance calculation was much like the take-off calculation in reverse, with the three segments of approach, flare, and ground roll. The approach was assumed to take place at 1.3 V_s with no thrust, giving a 2.7 degree glide slope. The flare was modelled like the transition for the take-off with the difference being that 1.2 g was used in the flare and the average velocity was 1.15 V_s. The result was that the total required distance to land at an airfield over a 50 foot obstacle was 13,260 ft. A runway without the obstacle still required 12,660 ft. This was less than the take-off requirement, so that the minimum runway length required was the take-off distance of 20,000 ft.

Roll Rate and Adverse Yaw

Roll is controlled on the Zephyr by four spoilerons mounted on each of the upper and lower wings. Each spoileron is 10 feet long, mounted outboard from the fuselages, and can be extended 5 inches. The spoileron effectiveness was modelled by estimating the reduction in C_L at ten stations, each 1.0 ft long, along the spoileron. Decreasing C_L on one side of the aircraft produces adverse yaw because induced drag is reduced. This was somewhat mitigated by the drag action of the spoilerons which provided proverse yaw; however, some vertical tail deflection was necessary to counter adverse yaw. This adverse yaw moment was added to the lateral trim analysis, where it was found that 11° of vertical tail deflection was required to counter the adverse yaw from 5 inches of spoileron actuation. This was within control limits[d].

To determine the time to roll to 30 degrees of bank, a simple model was used where the roll moment imposed by the spoilerons was opposed by the aerodynamic reaction and the moment of inertia about the x axis. It was found that with a full spoileron deflection of 5 inches, it took 11.7 s to roll to 30 degrees of bank.

Maximum Range and Endurance

The maximum range is obtained by flying the aircraft at maximum L/D, as given from the equation (P. 458 of Raymer[5]):

$$R = \frac{\eta_p}{C_{bhp}} \frac{L}{D} \ln\left(\frac{W_i}{W_f}\right) \tag{9}$$

[d]Note that there is not enough control from the vertical tails to counter this adverse yaw in an engine-out scenario. However, in that case the aircraft will only experience less than one degree of side-slip.

Maximum endurance is achieved by flying the aircraft at minimum power and is given by the equation (P. 459 of Raymer[5]):

$$E = \frac{\eta_p}{C_{bhp}\, V_{mp}} \frac{L}{D} \ln\left(\frac{W_i}{W_f}\right) \tag{10}$$

For both of these equations, W_i and W_f represent the initial and final weight of the aircraft, the difference being fuel burned. In the case of the Zephyr, the total fuel weight was 54 pounds (6 pounds H_2, 48 pounds O_2). Again, note that all weights quoted in pounds are Mars pound, not Earth pounds.

The specific fuel consumption, C_{bhp}, was calculated by noting from Perez-Davis and Feyman[2] that a 96 HP fuel cell operating at full power consumes 65.9 lbs/hr of fuel. It is very important to note that lb are Earth pounds and must remain in that form since one horse-power is 550 ft-lbs/s where the pounds are Earth pounds. The C_{bhp} was therefore 0.68 lbs/hr*HP or 3.44 $\times 10^{-7}$ 1/ft.

Using this fuel consumption and an assumed cruising altitude of 3000 feet, the maximum range and endurance were calculated to be 194 nautical miles and 1.54 hours respectively.

Aeroelasticity

Divergence

A discrete-element model was constructed to examine the divergence properties of the outer wings (the sections of the wings that are outboard from the fuselages). The assumption was made that if the unsupported outer wing was incapable of divergence, then the supported inner section would be incapable as well.

The results from the model gave that the lowest divergence-speed eigenvalue solution was far in excess of Mach 1, 1360 knots to be precise. Note that a twenty station model was used and that increasing the number of stations did not change the divergence speed more than 1%. This result indicated that the wing will simply not diverge under any real flight conditions. This is due to a robust wing structure and to the fact that the elastic axis is extremely close to the centre of lift so that the lift does not generate a large twisting moment.

Flutter

While having the elastic axis close to the centre of lift makes divergence highly unlikely, it increases the probability of flutter. Flutter is strongly dependent on the relative positions of the wing cg, centre of lift, and elastic axis, which were on average 31 in, 19.5 in, and 20 in from the leading edge respectively. By Pine's criteria[7], when the elastic axis is aft of the centre of lift, and the wing cg is aft of both, flutter is possible. However a Pine's analysis for flutter estimation showed that the flutter onset speed was 580 knots. Therefore flutter should not occur in any real flight conditions. The high flutter speed is due to the robust design of the wing torsion box and spar caps which resist oscillation.

A more refined analysis, or flight testing, will be required to assess the full three-dimensional flutter response of this aircraft. Pine's criteria, however, has historically been shown to be surprisingly useful for estimating flutter onset, despite the simplicity of the model.

The Propeller

To establish the aerodynamic design of the propeller, an optimization program written by Marcus Basien[8] was used, which was based on the work of Adkins and Liebeck[9]. The theory takes into account low Reynolds number conditions; and compressibility effects were added to the program by Basien, so it was thought to be useful in the Martian regime. The program was modified for the Martian atmosphere: air density, speed of sound and kinematic viscosity. The input to the program was the atmospheric conditions and a set of propeller parameters including flight speed, RPM, blade length, number of blades, and one of either thrust required or power input available. The output was a chord width, angle of twist, C_L, and Mach number at a number of discrete stations along the radius of the blade for a design of maximum efficiency.

The first iteration specified a required thrust of 160 lbs. This resulted in a low RPM (~500) propeller with a radius in excess of 12 ft. Clearly, this presented problems not only with regard to structure, but with respect to ground clearance. In subsequent iterations, the thrust requirement was dropped under the condition that multiple propellers could be used. Finally, a three-propeller configuration was arrived at where each propeller delivered 54 lbs at cruise speed (150 knots).

Parameter searches showed that ten was the optimum blade number. Fewer blades resulted in chord lengths that were too wide and more blades did not improve efficiency.

There was a narrow regime under which the propeller could operate. At large blade radii, higher RPM caused compressibility problems as the blade tips approached Mach 1. At the smaller blade radii, the program attempted to twist the blade near the hub beyond 90° in order to meet the thrust requirement. A blade radius of 5.9 ft and angular velocity equal to 900 RPM were chosen as the optimum values, based on a desire to keep the propeller diameter as small as possible and to allow a safety margin in the RPM.

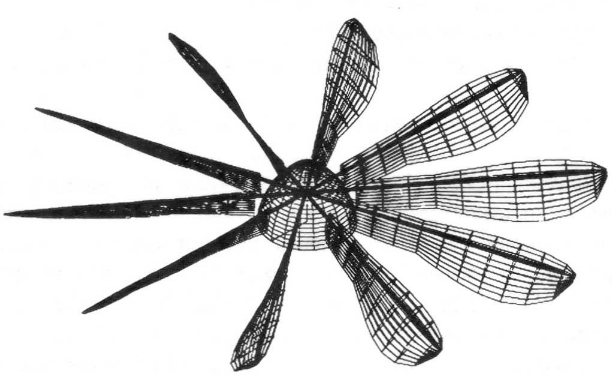

Figure 6.
The Zephyr propeller.

The propeller was optimized for cruise conditions at 150 knots with an efficiency of 70%. Above 200 knots the propeller efficiency decreases markedly. Therefore the never-exceed speed (V_{ne}) of the aircraft was set to 220 knots.

Conclusions

A manned aircraft capable of flight in the Martian atmosphere has been designed using available technology, with the exception of more efficient fuel cell engines predicted to be available in the next decade. Computer modelling has shown that the design meets flight specifications and therefore a manned Martian aircraft is possible for future development.

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