Initial Design of a Lightweight Mars Aircraft Mission

Lochie Ferrier

This paper outlines the design of a CubeSat based mission for Mars observation using a lightweight unpowered fixed-wing aircraft. The mission is comprised of a 3 unit support stage and a small deployable aircraft. During atmospheric entry, the support stage is deployed from a large primary spacecraft. The support stage then releases a parachute and descends independently providing communications support for a small self-assembling aircraft that is flown autonomously above the Martian surface for an extended period of time. The aircraft transmits surface imagery and data on Martian atmospheric during its flight. After a 10 minute flight, the support stage and aircraft are destroyed on impact. This paper discusses simulations and analysis performed to refine and validate the design. Communications, aerodynamics and deployment challenges are also discussed.

I. Introduction

Recently, there has been a large focus placed on independent interplanetary CubeSat missions with projects such as NanoSail-D and INSPIRE (Interplanetary NanoSpacecraftPathfinder In Relevant Environment) making significant progress [5, 6]. However, relatively little research attention has been placed on the potential of CubeSats as secondary payloads on larger interplanetary missions. The idea of a Martian aircraft using this deployment method is explored in this paper. Fixed wing flight in the Martian atmosphere is not a new concept, as many projects such as NASA ARES have examined its potential in the past [7]. Most studies have featured the aircraft as the primary payload, resulting in the need for long range flight in order to perform science worthy of the launch opportunity. Scientific instruments at the time of the study were also usually too large to be accommodated by smaller aircraft. However, as a result of recent advances in space systems technology, it may not be necessary to use aircraft of this scale to meet valuable objectives. It may be wise to first solve the problem of Martian flight using a smaller craft. This paper outlines an initial mission design and discusses some of the challenges involved in performing lightweight Martian flight. An explanation of mission simulation and analysis is also provided.

II. Mission Objectives

To guide mission design, several approximate objectives were developed, with a focus on science value and technology demonstration potential.

1. Test controlled unpowered fixed wing autonomous flight over a 10 minute timespan in the Martian atmosphere.
2. Gather and transmit high spatial resolution imagery of the surface or other lightweight science data over a large area at a low altitude.
3. Test the suitability of CubeSats as a supplement to Mars exploration missions.

III. Design Methodology

An iterative approach was used to translate the mission objectives into a realistic hardware and mission design for the Martian environment. The 1U container for the aircraft and the primary spacecraft entry sequence were primary constraints throughout. Commercially available technologies were used as reference designs for components such as the parachute and onboard computers for the support stage. Models were constructed using hobbyist materials and CAD software to investigate various aircraft and support stage configurations. The performance of these designs was then evaluated using CFD techniques and the computer simulation written specifically for the mission. The design was altered according to the results of these simulations and the process repeated. Several applications of this iterative process led to the design presented in this paper.

IV. Mission Hardware

A. Support Stage
To decrease the amount of modification required for the primary craft, the mission is based around an independent support stage. The support stage hosts the aircraft and provides communication support during its flight. The support
stage also features a parachute so that the aircraft can be released at a descent velocity less than 50 m/s. The initial design for the support stage measures 30x10x10cm (3U) and has a total mass of 1.5kg with the aircraft loaded. The 3U CubeSat standard was chosen for the support stage, as it allowed sufficient volume for a lightweight aircraft, but did not have the mass penalties of larger 6U and 12U standards [15]. The 3U standard has also been heavily developed and is currently well supported, unlike the larger proposed standards. To ensure that the design was realistic, CubeSat and model rocketry COTS components were used as reference designs for support stage components. As shown in Figure 1 and Table 1, the support stage is comprised of three central components, each of which is roughly 1U in volume. A 72 inch toroidal parachute is stowed at one end of the support stage to slow the descent. The parachute is modelled on the Iris Ultra 72 inch Compact Parachute offered by Fruity Chutes Consumer Aerospace Solutions [8]. The central component of the support stage houses the electronics to support the aircraft during flight and control deployment of mission elements. A 10 Whr EPS based on a Clyde Space unit provides sufficient power for all electronic systems [9]. A custom-designed interface board provides interface electronics for deployment systems and sensors such as an IMU for feedback. An onboard processor modelled on the NanoMind A712D CubeSat computer coordinates the mission autonomously, triggering events such as parachute and aircraft release [10]. The largest electronics component is the Iris X-band radio, a deep-space transponder in development by NASA JPL for interplanetary CubeSats [4]. The transponder is compatible with the Deep Space Network and will be deep space tested as part of the NASA INSPIRE mission [5]. The final 1U of support stage volume is used to store the aircraft in its folded configuration. The design of the deployment mechanisms for the parachute and aircraft are not discussed in this paper due to their mechanical complexity, however 0.25U of volume and 0.15kg of mass were reserved in the support stage design for their future inclusion.

![Figure 1. CAD model of 3U support stage with side panels removed. Individual electronic components are not shown on circuit boards, and the parachute container shape is illustrative. Deployment mechanisms for the parachute and aircraft are not shown.](image)

<table>
<thead>
<tr>
<th>Component Name</th>
<th>Allotted Mass (kg)</th>
<th>Allotted Volume (CubeSat Volume Unit of 10cm³)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Toroidal Parachute</td>
<td>0.2</td>
<td>1</td>
</tr>
<tr>
<td>Interface Board</td>
<td>0.07</td>
<td>0.1</td>
</tr>
<tr>
<td>10 Whr EPS with Battery</td>
<td>0.17</td>
<td>0.15</td>
</tr>
<tr>
<td>Onboard Processor</td>
<td>0.06</td>
<td>0.1</td>
</tr>
<tr>
<td>Iris X-band Radio</td>
<td>0.4</td>
<td>0.4</td>
</tr>
<tr>
<td>Aircraft</td>
<td>0.15</td>
<td>1</td>
</tr>
<tr>
<td>Deployment Mechanisms</td>
<td>0.15</td>
<td>0.25</td>
</tr>
<tr>
<td>3U Structure</td>
<td>0.3</td>
<td>N/A</td>
</tr>
<tr>
<td><strong>Sum</strong></td>
<td><strong>1.5</strong></td>
<td><strong>3.0</strong></td>
</tr>
</tbody>
</table>
B. Aircraft

The aircraft was designed with an emphasis placed on folding geometry, payload capacity and flight performance. It measures 39 cm in length, 53 cm in wingspan and has a payload capacity of 75 grams in a 288 cubic centimetre volume. In its folded configuration, the aircraft is stored in a 1U CubeSat volume. The scale of the aircraft compared to the support stage can be seen in Figure 2, demonstrating the packing efficiency of the design. The model of the aircraft presented in Figure 2 was used for determining folding geometry. To understand the aerodynamic performance of the aircraft, a separate model was constructed with a pointed nose, Aquila SM gliding aerofoil wing and NACA 0009 symmetrical aerofoil tail and elevator surfaces. This model is shown in Figure 3 with simulated airflow generated by the XFLR5 low Reynolds number CFD program [16]. Using XFLR5, the aircraft was calculated to have a drag coefficient $C_D$ of 0.010, lift coefficient $C_L$ of 0.402 and a maximum lift to drag ratio of 41.042. This allowed for prediction of the aircraft’s flight performance in the low-density Martian atmosphere, a flight environment that is not easily replicated using conventional wind tunnels or flight testing. The results of this aerodynamic analysis and its usage in a realistic mission simulation are outlined further in Section VI.

![Figure 2. Size comparison of deployed aircraft and support stage. The folded aircraft is shown inside the support stage.](image)

![Figure 3. Aerodynamics model of the aircraft with simulated airflow using XFLR5 CFD modelling software [16].](image)
To execute a controlled flight and collect meaningful science data, the aircraft carries flight sensors and two cameras. Alongside imaging data, data can be extracted from the flight control sensors, as flight path disturbances can be attributed to atmospheric effects. The sensor payload consists of:

- A surface facing high resolution camera for surface imaging and possibly navigation using efficient computer vision techniques.
- An IMU unit to determine aircraft orientation and flight path.
- An altimeter to determine aircraft altitude and flight path.
- A horizon facing low resolution camera for backup orientation determination and imaging.

C. Communications

Due to the large distance from Mars to Earth and the mass and volume constraints of the mission, delivering reliable communications is a challenging task. To increase the amount of science data that can be recovered, information is only transmitted from the mission to Earth, with no uplink capacity. To ensure that communications are maintained as the aircraft maneuvers, the communication between the aircraft and the Earth is divided into two stages. A small UHF radio on the aircraft transmits science and telemetry data to the support stage which features a similar radio as part of the Iris X-band system [5]. This data is then filtered and transmitted over the Deep Space Network using the Iris X-band radio on board the support stage. Due to the limited 3-4 W output of the Iris system using 18 W of DC power, the signal from the Iris X-band radio would likely require further relay through an orbiting Mars spacecraft [4]. This area of mission design requires further analysis to determine the viability of long distance communications in the dynamic flight environment.

D. Primary Spacecraft

The choice of primary spacecraft would likely be fixed if the mission was implemented, due to the low amount of spacecraft traffic to Mars atmospheric entry. The Mars Science Laboratory (MSL) Entry Descent and Landing (EDL) flight tested architecture was chosen as the reference primary spacecraft for this initial design[1]. The availability of literature, future missions, low entry velocity and significant payload capacity were all factors in selecting this spacecraft as a reference design. It is envisaged that through collaboration with the primary payload designers, the 1.5 kg total mission mass could be accommodated within the primary spacecraft entry payload capacity of approximately 3300 kilograms [2]. A CubeSat compatible deployment capsule such as a P-Pod would need be mounted to the aeroshell in a manner that would allow deployment of the support stage without interference with the primary mission [3].

V. Mission Sequence

The mission performance information in this section such as altitudes and velocities refers to the simulated scenario with aircraft aerodynamics discussed in VI. Figure 4 provides an overview of the mission sequence with this performance information annotated.

A. Support Stage Deployment

To this date, CubeSats have only been deployed in LEO under minimal acceleration. For this scenario, a container with a single small spring and electronically operated door is sufficient[3]. At the support stage deployment altitude of 11,000 m, the primary spacecraft descent velocity is approximately 282 m/s. The support stage must be reliably ejected with high velocity in order to minimise the impact threat to the 21.5 m primary spacecraft parachute and main spacecraft body. To produce this velocity, the CubeSat deployment module would likely need to be modified or redesigned with more powerful springs and a robust deployment door.

B. Initialization

After deployment from the primary spacecraft, the support stage initializes its electronic systems, which have been isolated from power using an ideal diode since integration with the primary spacecraft on Earth [9]. The initialization process is started using a separation switch triggered by the release of the support stage from the deployment tube. Four seconds were allocated for initialization, as the descent velocity of the support stage is still over 282 m/s at
Figure 4. Illustration of mission sequence with components and trajectories not to scale. Note that the horizontal movement of the support stage is for illustration purposes only.

Table 2. Initialization Stages

<table>
<thead>
<tr>
<th>Number</th>
<th>Stage</th>
<th>Time (s)</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>Activate power distribution system and assess battery levels.</td>
<td>0.2</td>
</tr>
<tr>
<td>2</td>
<td>Boot onboard processor and self-test for radiation damage during transit from Earth.</td>
<td>2</td>
</tr>
<tr>
<td>3</td>
<td>Run program to test interface board.</td>
<td>0.4</td>
</tr>
<tr>
<td>4</td>
<td>Record data from IMU.</td>
<td>1</td>
</tr>
<tr>
<td>5</td>
<td>Use IMU data to make parachute ejection decision.</td>
<td>0.2</td>
</tr>
</tbody>
</table>
this point and altitude must be conserved to prolong aircraft glide duration. The initialization process consists of 5 sequential stages, outlined in Table 2.

The allotted time for the boot of the onboard processor is likely to present a software and hardware engineering challenge. With appropriate design, possibly utilizing a fast boot procedure that does not load all programs, the required time frame may be achieved. If the boot time cannot be achieved independently, another option is to use a circuit on the primary spacecraft to initialize the support stage before deployment. This should be avoided however, as it could result in a need for primary spacecraft redesign. As the altitude loss with a 4 second initialization timeframe is 1165 metres, minimizing the initialization timeframe is critical to meeting mission objectives.

C. Parachute Deployment

After the initialization period, the support stage has a descent velocity of 296 m/s. It is necessary to decelerate the support stage as deployment of the aircraft at this velocity could cause fatal structural damage to the airframe. The 1.82 meter toroidal parachute is deployed for this purpose. With a drag coefficient of 2.2, the parachute immediately decelerates the support stage to 18 m/s despite the low atmospheric density of 0.0066 kg/m$^3$ at 9835 metres above the surface [14, 13]. After parachute deployment, the descent velocity is low enough to eject the aircraft. However, the shock of deployment would likely cause the support stage to oscillate underneath the parachute. To allow this motion to subside before aircraft deployment, a 3 second delay is implemented, after which time the support stage is 9781 metres above the surface.

D. Aircraft Deployment

The aircraft is folded in a tight configuration before deployment due to the volume constraints of the support stage. The mechanism to eject the aircraft from the support stage was not considered in this design, due to the limited design timeframe. However, the folding mechanism presented here has been designed to support a spring driving the rear of the aircraft out of the support stage. The unfolding process has been designed to use tape springs on the hinges of the airframe to unfold the aircraft after deployment. This method has not yet been tested for structural strength, mass or reliability. However, the geometry of the design has been tested as shown in Figure 5. Using this model, it was realised that the wingtips should feature a low-friction coating or material, as they can catch on the storage container and disrupt deployment. The unfolding sequence begins when the aircraft is approximately two thirds out of the support stage. At this point, the wings spring open as the aircraft is pushed out and the tail begins to unfold. Once the aircraft

![Figure 5](image-url)

Figure 5. The aircraft unfolding procedure shown using a scale aircraft model constructed from foam and a 1U container. As the airframe is not spring loaded, when the unfolding process is complete, the wings and tail are not fully extended as they would be with a spring loaded model.
has cleared the support stage, the tail fully extends and the two sections of the tailplane spring into place. It is predicted that this process would take place in less than 2 seconds after which time the aircraft can begin its autonomous flight.

E. Aircraft Flight

Before the aircraft can begin its mission, it must accelerate to its cruise velocity by diving directly towards the surface. The required glide airspeed for the aircraft at its release altitude is approximately 58 m/s as described in Section VI. The time for the aircraft to dive and pull out at an acceleration of 5 Martian Gs is 15 seconds. Once the aircraft has recovered from this dive and is in stable flight, it is 9216 metres above the surface. Given that the average sink rate of the aircraft is 12.6 m/s, the aircraft has approximately 732 seconds or 12 minutes and 12 seconds of usable flight time before it impacts the surface. This is comfortably above the requirement of 10 minutes of flight time and translates to a straight ground track flight path length of 35.136 km. The primary constraint limiting the flight time is the descent velocity of the support stage, as it provides communications support to the aircraft and flight time is not meaningful without communications relay for science data. During its 10 minute flight, the aircraft is autonomously controlled, using an onboard altimeter and IMU. Two onboard cameras may also be used to assist with navigation depending on the power of the onboard processing unit. It is likely that the location of the aircraft’s flight would be fixed by the requirements of the primary spacecraft. Although, if some input was allowed, there are some desirable destinations. The precise control of an aircraft mission allows for exploration of difficult to access areas on Mars such as Valles Marineris, a 7km deep area of canyons located just below the equator of Mars. The area is well suited to a lightweight aircraft mission, with many flat areas around the canyons for a primary spacecraft to land and rich potential for aircraft flight. The canyons are of great scientific value, with fault areas, chaotic terrain and evidence of glacial activity [12]. The canyon’s depth may also increase flight duration significantly. Figure 6 shows a possible 35 km straight line flight path over the main tract of Valles Marineris for illustration purposes. However, to minimise the communications distance between the aircraft and the support stage during the flight, it would be necessary to turn the aircraft during the flight, making a straight path difficult to execute. A more accurate evaluation of aircraft flight potential will be possible once an in-depth analysis of communications performance has been completed.

![Figure 6. Illustration of a 35 km flight path (marked by the yellow line) over the Valles Marineris main tract using publicly available Mars Orbiter Laser Altimeter data from Google.](image)

F. End of Mission

After 732 seconds of aircraft glide time, the support stage impacts the surface. The support stage would likely be destroyed by the high impact velocity, resulting in a loss of communications. At the time of support stage impact, the aircraft is 2106 m above the surface as shown in Figure 10.

VI. Mission Simulation

A computer simulation was developed to calculate mission performance from XFLR5 CFD analysis and other design information. The mission was simulated in a sequential manner and the motion of the support stage and aircraft were modelled seperately.
A. Simulation Process

The simulation stepped through the mission in the same sequence as outlined above, with 1 second time step resolution. The vertical velocity and altitude of the aircraft and the support stage as well as atmospheric density and timing data were recorded throughout. Vertical velocity was defined as positive in a direction towards the surface. Firstly, the deployment velocity and altitude of the support stage was determined using the MSL EDL mission data [1]. The data point of 11,000 m at a descent velocity of 282 m/s was selected for the support stage deployment. Then, the support stage was released and fell freely for 4 seconds in accordance with the mission design. The support stage and aircraft accelerated together during this time according to the surface gravity of 3.7 m/s/s [11]. The drag on the support stage without the parachute deployed was not modelled as the drag coefficient was determined using CFD software to be insignificant. The parachute deployment was modelled as an instantaneous process, with the descent velocity $v_p$ calculated according to Eq.1 and the atmospheric density $\rho$ in kg/m$^3$ modelled according to Eq.2 [13, 14]. The changes in gravity due to altitude were not modelled, instead the value of 3.7 m/s/s was used for $g$ throughout. In Eq.1, the drag coefficient $C_D$ was 2.2 and diameter $D$ 182cm from the parachute specifications and the mass $m$ was 1.5 kg, the loaded mass of the support stage. The value of $r$ in Eq.2 was 23.4 for altitude $h > 7000$ metres and -31 for $h \leq 7000$ metres. Once the parachute was deployed, a delay of 3 seconds was used to simulate the stabilization of the support stage after the deployment shock. Then the aircraft was released, and its velocity was calculated independently throughout its dive as it gained velocity towards its cruise airspeed. The aircraft mass was subtracted from the support stage mass after release. From the CFD simulation, the maximum lift to drag ratio of the aircraft was 41.042, with a $C_L$ of 0.630 and a $C_D$ of 0.010. This allowed for the modelling of the aircraft’s glide airspeed $v_a$ using Eq.3 with a glide angle 10 degrees below equilibrium glide angle in order to maintain stability despite pointing errors [17]. The aircraft followed a trajectory to pull out of the dive at 18.5 m/s/s after reaching the required airspeed. After the aircraft had performed the pull out maneuver, its vertical velocity was calculated using the glide angle and a timer started to record its total operational flight time. When the aircraft or support stage impacted the surface, the mission was deemed finished and the timer was stopped. The resultant trajectory from this simulation process is shown in Figure 10. This simulation process was used for the final mission design as outlined above. Earlier in the design process a simulation was developed using a similar process without aircraft aerodynamic modelling for the purpose of guiding design. The results of this simulation are shown in Figure 7, Figure 8 and Figure 9.

$$v_p = \sqrt{\frac{8mg}{\pi \rho C_D D^2}}$$

$$\rho = \frac{0.699e^{-0.0000009h}}{0.1921((r - 0.0022h) + 273.1)}$$

$$v_a = \sqrt{\frac{2 \cos(\arctan(\frac{C_D}{C_L}) + \frac{\pi}{18})) W}{\rho C_L S}}$$

B. Limitations

The simulation did not model complex aerodynamic forces such as control surface influences, which despite the low atmospheric density, are likely to significantly influence the mission. Wind and other atmospheric effects were also omitted, which would likely decrease aircraft range and duration performance if included in the simulation.

VII. Conclusions and Suggestions for Future Work

This initial design shows that the concept of an independent lightweight Martian aircraft is promising. However, several areas of the design need further work to determine whether the concept is viable. The primary area of concern is in the communications from the aircraft to the support stage and back to Earth. Future work could focus on verifying that a light, small and powerful transmitter could be produced for the aircraft, as well as analysing the issues with orientating the X-band patch antennas on the support stage to the DSN. Both of these areas could be investigated with minimal cost in a reasonably short timeframe. The second area of concern lies in aerodynamics, as flight in the low density atmosphere remains physically unproven. High altitude test flights and wind tunnel testing could determine whether fixed wing flight is possible at the glide velocity with high inertia and minimal control.

Acknowledgments

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[9] “1U CubeSat EPS 10 Whr Integrated Battery,” Clyde Space Ltd


[16] “XFLR5 Analysis of foils and wings operating at low Reynolds numbers,” Purdue University, 2009

Figure 7. Graph of glide slope, support mass (without aircraft) and mission time of mission designs that result in aircraft impact with support stage altitude greater than 50m.

Figure 8. Graph of glide slope, support mass (without aircraft) and aircraft cruise velocity (airspeed) of mission designs that result in aircraft impact with support stage altitude greater than 50m.
Figure 9. Graph of glide slope, aircraft cruise velocity (airspeed) and mission time of mission designs that result in aircraft impact with support stage altitude greater than 50m.

Figure 10. Graph of altitude and time since support stage deployment from primary craft for mission with simulated aircraft aerodynamics. Aircraft glide time is 732 seconds. Aircraft trajectory is shown in blue, and support stage trajectory is shown in green. Note the slight curve in the support stage altitude over time due to the increase in atmospheric density giving the support stage parachute a greater drag force. Also clearly visible is the aircraft’s dive maneuver to gain velocity, and the 2106 m height difference between the aircraft and the support stage at the time of impact.