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Studies on an aerial propellant transfer space plane (APTSP)

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Abstract

This paper presents a study of a fully reusable earth-to-orbit launch vehicle concept with horizontal take-off and landing, employing a turbojet engine for low speed, and a rocket for high-speed acceleration and space operations. This concept uses existing technology to the maximum possible extent, thereby reducing development time, cost and effort. It uses the experience in aerial filling of military aircrafts for propellant filling at an altitude of 13 km at a flight speed of $M = 0.85$. Aerial filling of propellant reduces the take-off weight significantly thereby minimizing the structural weight of the vehicle. The vehicle takes off horizontally and uses turbojet engines till the end of the propellant filling operation. The rocket engines provide thrust for the next phase till the injection of a satellite at LEO. A sensitivity analysis of the mission with respect to rocket engine specific impulse and overall vehicle structural factor is also presented in this paper. A conceptual design of space plane with a payload capability of 10 ton to LEO is carried out. The study shows that the realization of an aerial propellant transfer space plane is possible with limited development of new technology thus reducing the demands on the finances required for achieving the objectives.

1. Introduction

Today, most operational launch vehicles are expendable. The sole exception is the US space shuttle, which is partially re-used. As the many technologies needed to deliver a payload into space have matured, ideas about the utilization of space seem more realizable and consequently, the variety and magnitudes of payloads have increased. This increasing requirement and the obvious benefits of substantially lower costs have driven worldwide searches for a reusable launch vehicle (RLV) [1]. Several concepts have been proposed but none have translated into an operational vehicle. A common obstacle in each concept appears to be an excessive optimism about the levels of performance of essential technologies: for example, anticipating substantial improvements by way of new materials. Until such levels are achieved, no mission can complete. In this paper, we present analysis of a concept derived from the ‘Black Horse’ of the British Space Agency and using available levels of performance in essential technologies.

The Black Horse concept consists of a vehicle that takes off from standard airport runways and climbs to an altitude of 8 km using rocket engines. At this altitude, it is refueled from a tanker. Then it climbs to

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Nomenclature

C_l	lift coefficient
C_d	drag coefficient
I_{sp}	specific impulse in s
m_f	mass of fuel (kg)
m_i	initial mass at rocket ignition (kg)
m_o	final mass at rocket burn out (kg)
m_{ox}	mass of oxidizer (kg)
m_{pl}	payload mass (kg)
m_R	mixture ratio
m_{st}	total structural mass (kg)
M	mach number
n	deceleration
V_{ox}	volume of oxidizer (m^3)
Q	propellant transfer rate (l/min)
sf	structural factor
T_{cr}	cruise time (s)
ρ_{ox}	density of oxidizer (kg/m^3)
ΔV	incremental velocity (m/s)
σ	atmospheric density ratio
ρ_o	density of air
r_o	radius of earth

deliver a payload into LEO and returns. The fuel is a mixture of hydrogen peroxide and kerosene. The design assumes a structural factor of 8.2%. We consider this level to be too optimistic for the near future. At currently available levels of structural factor of about 15%, the I_{sp} required dictates that a LOX/LH2 engine be used. Our analysis assumes these starting points and develops accordingly.

Another design, which is similar to the Black Horse in that it makes use of aerial fuelling, is the Pioneer Rocketplane [2]. This concept is more immediately realizable because it is also designed around currently available levels of technology. An essential difference is that there is an expended upper stage. So a LOX/kerosene engine is sufficient to carry the rocketplane to an altitude of 70 miles to release the upper stage and return. Both LOX and kerosene are loaded from tanker aircraft. The proposal presented here considers a fully reusable vehicle and so requires a LOX/LH2 engine. Hydrogen handling on the ground must be addressed carefully, but a single transfer (LOX) from tanker aircraft will suffice.

Other RLV concepts have also considered solutions for LOX/LH2 engines. And, a widely reported proposal is to use the LACE concept. LACE calls for a supersonic cruise during which atmospheric air is collected, liquefied, and purified. Although, LOX is required for the mission to complete, it does not seem essential that the same vehicle must prepare the LOX, carrying all the equipment necessary for the LACE process [3]. The extended, powered, supersonic cruise phase is also a drag on performance. Aerial propellant transfer might seem unusual, but is a widely practiced operation. At present there is no experience with aerial transfer of LOX, but we cannot see a fundamental difficulty in developing this technology [4].

The essential analysis of this concept does not distinguish between manned and unmanned missions. The re-entry calculations have assumed it to be a manned mission when fixing a deceleration limit, and the detailed weight estimate has also allowed for a manned operation. There would be a change in the effective payload, but a more important effect is the need to develop an autonomous LOX transfer technology. This is not in keeping with one of the prime principles of this concept that as far as possible currently available technologies would be used.

2. Configuration design

Typical expectations of an RLV concept are [5]:

(1) Around 100 reuses for a stage and 50 reuses for an engine.
(2) A reasonable development time of about 5 years.
(3) Cost effectiveness (one-tenth of the present cost).
(4) Service life of 10 years.
(5) High launch frequency ($>$ 10/year).
(6) Mission flexibility.
(7) Safe abort capability.
(8) High reliability.

However, at this concept stage, the design begins with the requirement: 10 ton payload in 400 km LEO, with a payload fraction of 2%.

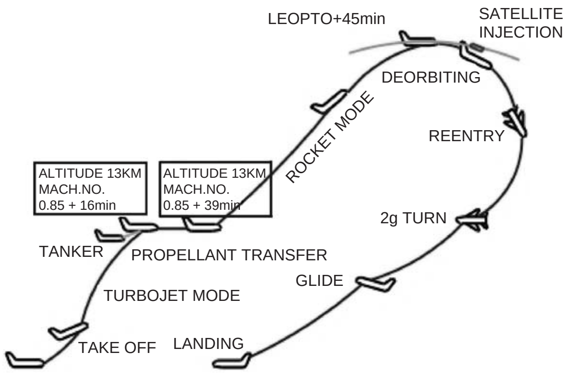

Fig. 1. Mission profile of APT spaceplane.

2.1. Mission profile

Fig. 1 shows the mission profile of APT space plane. Total mission includes the following operations:

(1) Horizontal take-off using conventional jet engines.
(2) Climb to 13 km altitude.
(3) Rendezvous with tanker.
(4) Cruise at constant speed ($M = 0.85$) at 13 km altitude.
(5) Filling of oxidizer from the tanker during cruise.
(6) Separation of tanker.
(7) Ignition of rocket engines at 13 km altitude.
(8) Reach the orbit at required velocity.
(9) Satellite injection.
(10) De-orbiting.
(11) Re-entry into Earth's atmosphere.
(12) Glide and land in powerless mode.

2.2. Rocket mode

The design methodology used in this work consists of two stages. In the first stage, the structural factor of the upper stage, and the payload mass, the mass parameters of the upper stage are calculated from a known incremental velocity using the standard formulae for flight in vacuum. In the second stage, the trajectory is calculated using the effects of gravity and drag (within the atmosphere). This parameter is used as input for the lower stages to estimate the mass parameters of the entire vehicle.

The parameters for rocket mode operation were calculated using equations given in [6]. Rocket engines are ignited at an altitude of 13 km at a Mach number, $M = 0.85$, after the total quantity of oxidizer required for rocket operation is filled from the tanker aircraft. The general rocket equation for incremental velocity is [4]

$$\Delta V = I_{sp} \ln\left(\frac{m_i}{m_o}\right). \tag{1}$$

The total mass is the sum of structural mass and payload mass $m_o = m_{st} + m_{pl}$. Structural factor of a rocket engine is $sf = m_{st}/(m_i - m_{pl})$. It follows that

$$\Delta V = I_{sp} \ln \frac{m_i}{sf(m_i - m_{pl}) + m_{pl}}. \tag{2}$$

Knowing the structural factor, payload mass, and specific impulse of the rocket engine used, the initial mass of rocket mode can be found out using Eq. (2). Mass of propellant required for rocket mode is $m_{prop} = m_i - m_{st} - m_{pl}$; mass of oxidizer is $m_{ox} = m_{prop} m_R/(m_R + 1)$; mass of fuel is $m_f = m_{prop}/(m_R + 1)$.

Cruise time for oxidizer filling (T_{cr}) is obtained by taking the propellant transfer rate Q, to be 113 l/s, which is the current transfer rate for military aircraft. Cruise time, $T_{cr} = m_{ox}/\rho_{ox} Q$. Table 1 lists the major vehicle parameters for a 10-ton payload with a rocket engine of specific impulse 4600 Ns/kg (using LOX/LH2) and a rocket mode structural factor of 15%. Also listed are the corresponding parameters for a much smaller payload of 700 kg.

2.2.1. Structural factor

Table 2 lists structural factors of several launch vehicles from around the world. The Black Horse concept, which is similar to the present proposal, assumes a structural factor of about 8.2%. It seems too low a figure requiring much advancement beyond presently available levels of technology. Considering the additional weight of the air-breathing engine and sub-systems, thermal protection system for re-entry, and propellant for de-orbiting, a structural factor of 15% has been assumed for the present design. This value is in the range between rocket engine stages that have a structural factor between 10% and 12% and

[4] Nomenclature is given in Appendix at end of paper.

Table 1
Major vehicle parameters

Payload (ton)	10.00	0.70
Total structural mass (ton)	30.57	2.14
Propellant mass (rocket mode)	173.23	12.08
LOX (ton)	148.48	
LH2 (ton)	24.75	10.35
		1.73
Total mass for rocket mode (ton)	213.80	14.97
Mass of kerosene for air breathing mode (ton)	13.90	0.70
Time of cruise (s)	1380	96
Time of rocket mode flight (s)	365	323
Max thrust requirement for air-breathing engine (at the end of cruise) (kN)	250.54	16.40

Table 2
Structural factors of launch vehicles

Country	Vehicle	Strap-on booster	Per stage (Stage 1, 2, 3, 4)
Japan	HI	0.17[a]	0.15, 0.17, 0.16
	HII	0.16[a]	0.12, 0.15
Russia	Energia		0.10, 0.09
	Zenith		0.10, 0.10, 0.10
	Proton		0.10, 0.09, 0.16, 0.15
Europe	Ariane 4	0.25[a], 0.10[b]	0.07, 0.08, 0.11
USA	Delta		0.05, 0.13, 0.14
	Pegasus/Taurus		0.13, 0.11, 0.20
	Scout	0.14[a]	0.14, 0.23, 0.21, 0.15
India	PSLV/GSLV		0.15 overall

[a]Solid propellant.
[b]Liquid propellant.

short-duration flying, short-winged aircrafts that have structural factors between 12% and 17%. It is believed that the structural factor of 15% is achievable with the current status of composite material technology. Fig. 2 shows the variation of mass of the vehicle

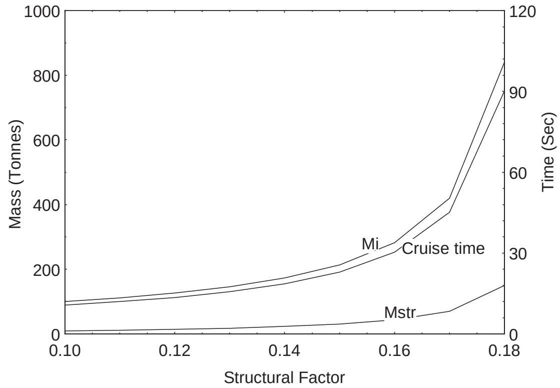

Fig. 2. Variation of vehicle parameters with overall structural factor of rocket mode.

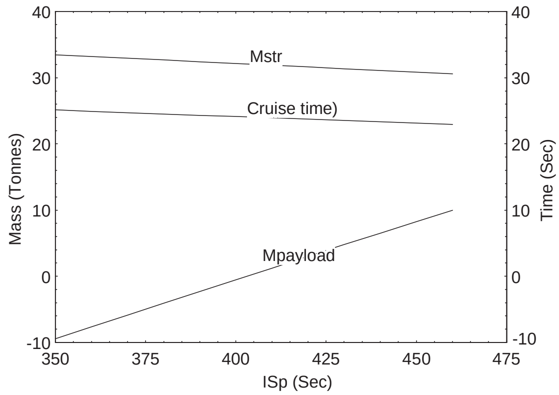

Fig. 3. Variation of vehicle parameters with rocket engine specific impulse.

(at the time of rocket ignition), cruise time for LOX transfer (T_{cr}) and total structural mass (m_{st}), with structural factor (sf).

2.2.2. Selection of propellant

Fig. 3 shows the variation of payload, cruise time and structural mass with specific impulse for the take-off weight of 213.8 ton. A non-cryogenic propellant combination with a specific impulse of 3500 Ns/kg gives a negative payload. Hence a LOX/LH2 propellant combination with a specific impulse of 4600 Ns/kg was selected.

Table 3
Rocket engine specifications

Thrust (ton)	82
Chamber pressure (atm)	200
Specific impulse (Ns/kg)	4600
Propellants	LOX/LH2
Mixture ratio	6
Area ratio	100
Cycle	Staged combustion cycle
No. of engines	3

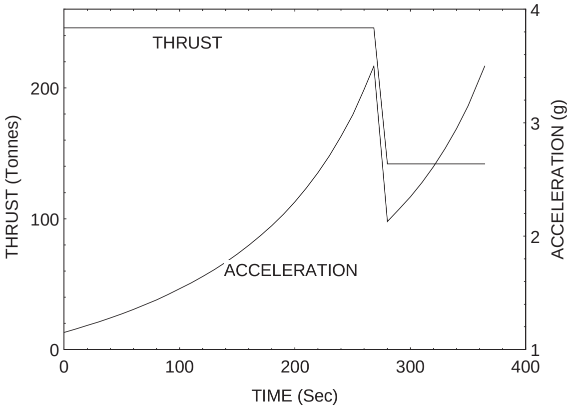

Fig. 4. Thrust and acceleration history of rocket engine.

2.2.3. Rocket engine

A high chamber pressure cryogenic engine using staged combustion cycle was selected for powering the upper stage [7]. To reduce the overall vehicle envelope, a three-engine cluster was selected. The specifications of the engine are given in Table 3.

The initial acceleration of the vehicle at rocket ignition is limited to 1.15 g due to structural considerations. The maximum acceleration of the vehicle is limited to 3.5 g due to crew considerations. The rocket engine has to be throttled down to meet this criterion. Fig. 4 shows the variation of thrust and acceleration profiles with respect to time during rocket mode operation.

Table 4
Wing parameters

Wing area (m^2)	230
Aspect ratio	4
C_{lmax}	1.1
Wing loading (max) kN/m^2	9.12
Airfoil	Supercritical airfoil with t/c-9%
Wing volume (m^3)	78.72

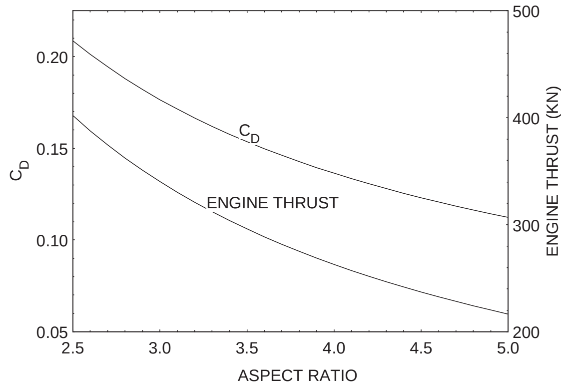

Fig. 5. Variation of C_d and air breathing engine thrust with wing aspect ratio.

2.3. Wing

The maximum weight of the vehicle is at the point of completion of propellant transfer. The wing area was estimated based on this weight. The angle of attack increases during propellant filling to increase the lift co-efficient. A large value of $C_{lmax}(=1.1)$ was chosen to minimize wing weight. The wing area was estimated from C_{lmax}. A highly swept wing planform is required to reduce the heating effects during atmospheric re-entry. The wing aspect ratio was fixed based on the thrust requirement during air breathing mode.

The major design parameters of the wing were calculated using standard procedures [8,9], and are given in Table 4. The variation of C_d and engine thrust with aspect ratio is shown in Fig. 5. The variation of wing area and engine thrust with C_{lmax} is given in Fig. 6.

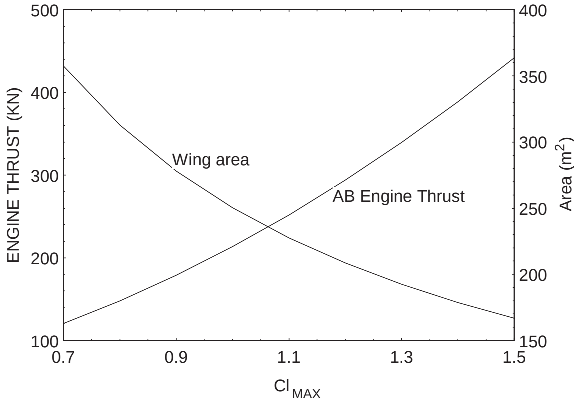

Fig. 6. Variation of wing area and afterburning engine thrust with C_{lmax}.

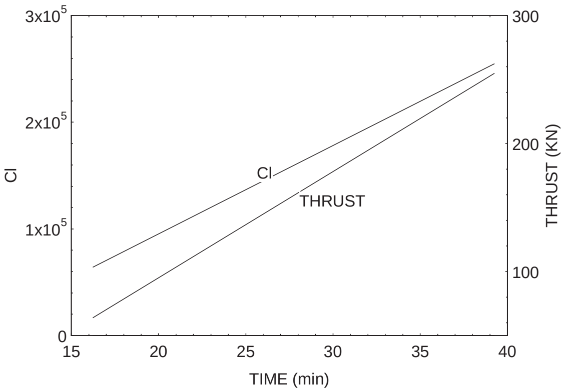

Fig. 7. Variation of lift coefficient and engine thrust during cruise.

During propellant filling, the angle of attack progressively increases which increases the lift coefficient corresponding to the increase in vehicle mass. The variation of C_l and thrust during cruise is given in Fig. 7.

2.4. Air breathing engine

An after-burning turbojet engine can be used for during atmospheric flight. Since the air-breathing engine is used during atmospheric flight only, which is about 30 min, the weight of the engine is the major consideration rather than specific fuel consumption. Hence an afterburning turbojet engine with a high thrust-to-weight ratio is needed. The overall mass

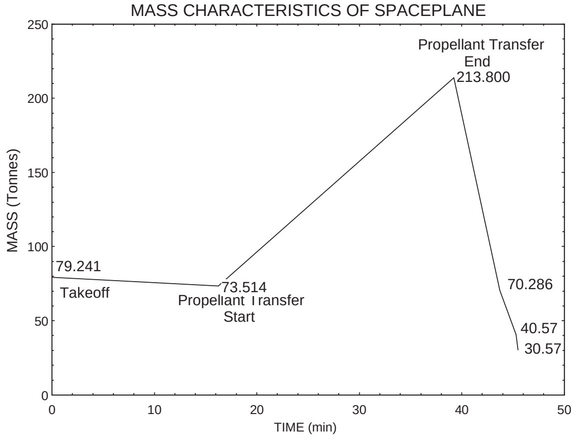

Fig. 8. Mass characteristics of spaceplane.

parameters of the vehicle from take-off to landing are shown in Fig. 8.

2.5. Weight budget

An approximate weight estimate was done using empirical relations and equations from [8,10] and is shown in Table 5. The structural weight limit can be realized using advanced materials like carbon–carbon composites, which are now-a-days commonly used in modern aircrafts and rockets.

3. Aerial re-fuelling

The space plane will take on LOX in flight from a specially modified cargo aircraft, enabling it to perform the launch mission without the heavy structure needed to carry LOX at take-off. The tanker aircraft will have a set of commercial LOX tanks on a cargo deck feeding a transfer line extending through one of the hatches. Since only the hatch door needs to be modified to pass the line, no airworthiness-affecting modifications are needed to the tanker. This is a considerable saving over other concepts that use an aircraft to carry their entire orbital vehicle, requiring a considerable structural strengthening of the aircraft. Two tankers that have been used extensively are KC-25 (BOEING 747 tanker version) which can carry about 200 ton of propellant and KC-10 (DC-10 tanker version), which can carry about 150 ton

Table 5
Weight budget

Component	Weight (kg)
Rocket engine	3500
LOX Tank	1889
LH2 Tank	5166
Turbo jet engines	4500
Wing	7080
Landing gear: main landing gear	2640
Nose wheel	558
Total	3198
Crew cabin: instruments	400
Floor	38
Seat and accessories	212
Total	650
Thermal protection system	650
Life support system	347
Tail planes	1250
Reaction control system	180
Fuselage: payload bay skin	358
Stringers and longerons	1226
Frame	847
Total weight of fuselage frame	2016
Nose weight	144
Total	2160
Gross structural weight	30,570
Payload	10,000
Kerosene	16,800
LOX	148,483
LH2	24,747
Total weight	230,600

of propellant. Since the propellant mass needed is around 150 ton, new tanker aircraft need not be designed.

4. Re-entry and landing

A powerless gliding is done to land on the Earth's surface. In a successful entry, the structure, the equipment, and any living occupants must tolerate the maximum deceleration and the heat transfer to the vehicle. In addition, at the end of the entry phase, the vehicle must be in a proper position with an appropriate velocity for a predicted touchdown for a landing at a desired location with a lifting re-entry. The vehicle must remain in an entry corridor defined by undershoot and overshoot boundaries. If the vehicle strays into the undershoot area, positive lift can be used to return to the corridor. Negative lift can be used to return to the corridor from the overshoot area [11]. A lifting-re-entry trajectory is the equilibrium glide, which is a relatively flat glide in which the gravitational force is balanced by the combination of the lift and centrifugal forces.

The lifting ballistic coefficient is defined as $\mathrm{LBC} = W/C_\mathrm{d} \times S \times (L/D)$ and the ratio of re-entry velocity to circular-orbit velocity is

$$V/V_\mathrm{cs} = \frac{1}{\sqrt{1 + [\rho_o g_o r_o / 2(\mathrm{LBC})\sigma]}}. \tag{3}$$

It can be seen that if the lifting ballistic coefficient is sufficiently low so that the vehicle slows down at a higher altitude, then the heating load during re-entry is lower. As the ballistic coefficient increases, the vehicle penetrates into the atmosphere before it starts to slow down. An important consideration during the reentry is the deceleration experienced by the vehicle as well as the passengers. Passengers can withstand only up to a maximum of about 1.5 g of deceleration. Deceleration during reentry is defined as

$$n = \frac{-1}{(L/D) + [2(W/C_\mathrm{d}S)\mathrm{e}^{bh}/\rho_o g_o r_o]}. \tag{4}$$

The variation of velocity and deceleration with altitude during re-entry in the equilibrium glide mode was calculated and is shown in Fig. 9.

5. Discussion and concluding remarks

In this paper, a concept for a reusable launch vehicle that can carry 10 ton of payload to LEO or about 3.5 ton to GTO is described. Also calculations were made for a minimum payload of 700 kg and compared

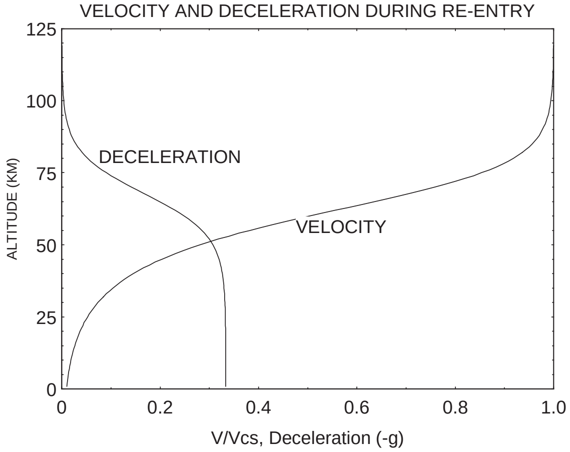

Fig. 9. Velocity and deceleration during re-entry.

with the Black Horse concept. Here, the required incremental velocity is achieved using a LOX/LH2 propellant combination in the rocket mode. The study shows that with a take-off weight of 85 ton and structural factor of 15%, a payload of 10 ton to LEO is possible. In the Black Horse concept the structural factor used is only 8.2%. This requires development of new materials for realization. But with the structural factor at 15%, it is possible to realize the vehicle with the use of available materials. The major advantage of this configuration is that no new technology is required for development of this vehicle. Other advantages of this concept are lowered turn-around time due to minimum launch pad operations and low initial thrust requirement.

The major limitation for a horizontal take-off single stage to orbit (SSTO) launch vehicle using an air-breathing engine for the atmospheric flight is the high takeoff weight due to large and heavy wings. The weight penalty is so high that it is not possible to realize such a vehicle. Here, to reduce take-off weight, we transfer the required oxidiser for rocket mode operations from a tanker aircraft at an altitude and air speed. So, the amount of propellant required for overcoming drag and gravity losses is much less. Vehicle tankage, wings, landing gear, etc. will be lighter. LOX is extremely inexpensive. The experience base for using LOX is current and intact. But compared to other oxidisers, LOX requires high purity. A small impurity tends to burn and cause an evolution of oxygen gas that destroys delicate parts and leads to catastrophic failure. Impurities of all kinds, particularly organics, must be absolutely avoided. Still this vehicle can be realized within a shorter time since existing technologies will suffice.

There remain two critical features of the above study in the realization of this space plane. First, the weight estimate was done with very small margins. The realization of a structural factor of 15% is challenging. A large amount of composite materials have to be used to reduce the structural weight. Secondly, there is no experience in aerial filling of cryogenic propellants. A scheme has to be evaluated for cryogenic propellant transfer and will have to be qualified by conducting a suitable number of trials.

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