Concept definition and design of a single-stage-to-orbit launch vehicle-hyperplane

Raghavan Gopalaswami
Retired from Givernment of India in 1988

CRITICAL FACTORS IN CONCEPTUAL DESIGN AND TECHNO-ECONOMICS OF REUSUABLE SPACEPLANES

R.GOPALASWAMI
53, Vayupuri, Secunderabad- 500094, India.

Comparing advanced reusable spaceplane concepts and programmes, Buffo identified seven attributes to describe a true spaceplane. This paper adds two more attributes to clearly distinguish spaceplanes from rocketplanes. One of them is the hydrogen fuel fraction (HFF), the ratio of the mass of liquid hydrogen fuel to the take-off mass. Excellent correlation between HFF and spaceplane techno-economic performance is seen from regression trends arising from parametric analysis. A modified rocket equation, termed spaceplane equation, shows how high HFF, scalability and multi-stage rocket performance are feasible in single stage to orbit (SSTO) spaceplane by the combined use of fuel efficient airbreathing engines and in-flight LOX addition (FLOX). A LACE-FLOX aerocryogenic engine cycle with air liquefaction (using LACE engines) integrated with vortex/higee liquid oxygen separators may well provide both high HFF and a single combined-cycle engine from earth-to-orbit. The global status of aerocryogenic technologies is also provided.

Keywords: Hydrogen fuel fraction, aerocryogenic, LACE-FLOX, spaceplane equation, scalability

1. INTRODUCTION

Reusable Launch Vehicles (RLVs) were first seriously studied with several feasible projects proposed in the 1960s. However, the advent of the rocket-based "Space Shuttle" concept and programme put an end to airbreathing design concepts.The first Space Shuttle failure in 1986 challenged mankind's expanded presence in space. This led to a variety of airbreathing RLV designs from many countries. While most of them remained as design concepts, a few like NASP moved ahead with funded programmes. A few years later the NASP programme was closed down and the future of direct, airbreathing ascent from earth to orbit in a single stage was in doubt. But, new RLV design concepts based on rocket propulsion emerged. To date, however, none appear to be a replacement for the Shuttle. This paper reviews the evolution of RLV design concepts and suggests a method for comparing their techno-economic performance.

2. THE CONCEPT OF SPACEPLANES IN 1980S

Analysis of several RLV mass properties reveals why only a certain class of RLVs (characterized by Buffo [1] as "True Spaceplanes") meet the requirements of revenue earning space applications like space solar power and space tourism. This paper identifies a critical conceptual design parameter, the "hydrogen fuel fraction (HFF) at take-off", for a comparative assessment of performance and cost of emerging RLV design concepts.

Buffo characterized "True Spaceplanes" as

i. Horizontal take-off and landing systems.
ii. Using conventional or slightly modified aircraft runways.
iii. Reusable
iv. Being Single-Stage-To-Orbit (SSTO) or Two-Stage-To-Orbit (TSTO) concepts with airbreathing propulsion.
v. Using advanced materials.
vi. Re-entering the atmosphere and landing with or without propulsion power.
vii. Being either manned or unmanned.

Using these seven criteria, Buffo compared spaceplane concepts from USA (NASP), UK (Hotol), Germany (Saenger), Japan (Hope), France (Hermes), Soviet Union (Buran) and India (Hyperplane). It is to be noted that Hope, Hermes and Buran are not fully reusable. Figure 1 illustrates the Avatar/ Hyperplane reusable spaceplane concept with In-Flight LOX Collection, from India. The non-orbital hypersonic transatmospheric spaceplane (RLV) technology demonstrator (HTV) of the Hyperplane concept of family of scalable spaceplanes is illustrated in Fig. 2.

Reviewing the progress in spaceplane conceptual designs during the last 20 years, it is seen that except for the design concepts from UK (Skylon) and India (Avatar), no other spaceplane design concept was sustainable to date. This could be explained by the addition of two more attributes that may be added to Buffo's seven criteria:

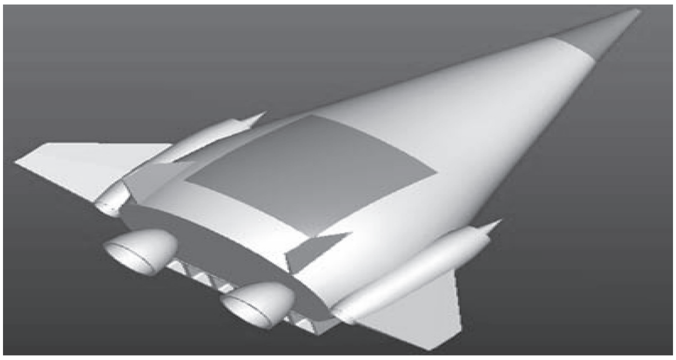
a. All spaceplane engine and airframe technologies must be fully ground-testable.
b. Spaceplanes should have relatively high HFF at take-off, essential for high techno-economic performance.

The latter of the two above is at present hypothesis only, as experimental systems have not been flown yet. Nevertheless, it is shown to be a useful parameter in this study, to evaluate preliminary RLV system design concepts.

The NASP programme was cancelled because its design demanded an airbreathing scramjet engine to operate at Mach

Symbols

g_o	acceleration due to earth's gravity, 9.81 m/s^2.
I_s	Vehicle mission average specific impulse, (s)
I_{sA}	Vehicle mission average specific impulse air breathing engines, (s)
I_{sR}	Vehicle mission average specific impulse rocket engines, (s)
k_L	Velocity loss factor
M_o	Mass of space vehicle at take-off (t)
M_A	Mass of space vehicle at end of air breathing phase (t)
M_E	Mass of space vehicle in orbit (t)
M_{PL}	Mass of payload in orbit (t)
M_s	Mass of (empty) structure at take-off (t)
R	Vehicle mass ratio (overall) = M_o/M_E
R_R	Vehicle mass ratio (Start rocket phase) = M_A/M_E
$R^* = R_1 \times R_2 \times R_3 \ldots R_n$	Mass ratio of multi-stage rocket with 'n' stages i =1, 2, 3…n
R_A	Vehicle mass ratio (End airbreathing phase) = M_A/M_0= Mass addition ratio
SLC	Specific Launch Cost ($/Kg in LEO)
$\beta = I_{Sa}/I_{sR}$	Ratio of endo/exo atmospheric mission average specific impulse
$\varepsilon = (R_i)/R$	Mass ratio multiplier factor (Multi-stage Rocket)
$\zeta = (R_A)^\beta$	Mass ratio multiplier factor (single-stage-to-orbit spaceplane)
V_I	Vehicle ideal orbital velocity (m/s)
V_E	effective exhaust Velocity (V_E = g Isp)
ΔV	Maximum change of velocity of the vehicle in drag free, field-free space
V_A	Vehicle Velocity at end of the endo-(airbreathing) phase (m/s)
V_0	Vehicle delivered orbital velocity (m/s)
$\Delta V_A = V_A - 0$	Vehicle velocity increment from take-off to end air breathing phase (m/s)
$\Delta V_R = V_0 - V_A$	Vehicle velocity increment end-air breathing phase to end rocket phase (m/s)



HYPERPLANE/AVATAR
(India)

Fig. 1 Reusable Spaceplane with In-Flight LOX Collection.

20 and at heights up to 40 Kms, extreme conditions that could not be verified in ground tests. Saenger, Hope, Hermes, and Buran dropped out of consideration possibly due to budgetary constraints and political decisions and it may be pointed out that their propulsion design concepts ended up (in retrospect) in relatively low HFF (8-15%) at take-off.

Thus, the surviving concepts are the Hotol [2]/Skylon [3, 4] and the Hyperplane/Avatar as they meet all the nine attributes of true spaceplanes. Skylon C2 (345 tonnes take-off weight/15 tonnes payload weight) with 25% HFF is a larger version of Skylon C1 (275/12), with a take-off capable undercarriage; while the 198-tonne Hotol was the original version that had a specially designed detachable trolley for fully fuelled take-off and light-weight undercarriage for landing.

Following the "Hyperplane" concept [5, 6] the 25-tonne take-off weight Avatar conceptual design emerged by extensive numerical simulations [7-10] as potentially the smallest feasible orbital spaceplane in the "Hyperplane" class i.e. airbreathing spaceplanes with in-flight air collection, liquefaction and LOX separation in hypersonic flight regime. These design concepts reported 56-60% HFF with new technologies for in-flight generation and on-board storage of liquid oxygen (FLOX system). The distinctive feature of Hyperplane/Avatar concepts was that no liquid oxygen was carried on-board at take-off.

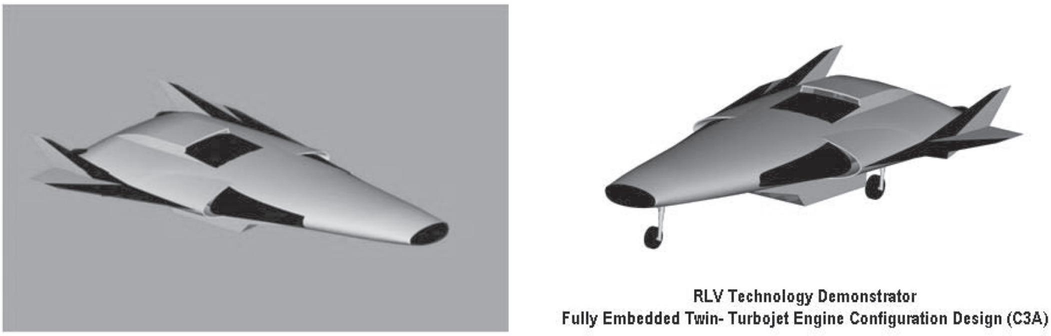
3. THE CONCEPTS OF RLVS IN 1990S

Throughout the 1990s, several spaceplane designs emerged in the US as smaller private companies entered the fray. The term 'spaceplane' was replaced by the term RLV. These concepts, however abandoned airbreathing propulsion concepts in favour of ground/air launched conventional rockets, where all the oxygen needed for propulsion in space is carried at launch. The reduction in cost of access to space was envisaged by reusing single stage vehicles or the two stages of conventional rockets, by parachute recovery of each stage.

Among the more prominent RLV concepts in the US were the X-33, the Roton, Astroliner, Pathfinder, Spacecruiser, Space Access SA-1, Kistler K1, Argus Maglifter and NASA's SLI-C1 and C2; and in UK, the Spacecab. The question arises: are these rocket based RLV systems the only options for low cost access to space for emerging space markets? What about a fresh look at the concepts of 1980s, the "True Spaceplanes"?

It is to be noted that spaceplanes are useful for space transportation into Low Earth Orbits (LEO, 300-500 Kms) only. But, telecommunication and space solar power satellites require space transportation to orbital heights of up to 36,000kms. Major space agencies currently believe that for such deep space missions they need to continue deployment of large, vertically stacked space rockets.

However, concepts related to reusable space transportation vehicles envisage deep space transportation infrastructure con-



Configuration Tentative

•Take-off Mass: 3.5 to 5.0 tonnes
• Horizontal Take-off & Landing
•Fully Reusable
•Demonstrates Reusability & Maintainability
•Flying Test Bed for Airbreathing Engine (Turbojet /LACE/Scramjet)
•Demonstrates aerocryogenic technologies in Flight (Heat Exchangers & LOX Separators)
•Atmospheric Flight Regime Identical to Hyperplane/Avatar Spaceplane.
•Non-orbital
• Dual Fuel (Hydrogen/Aviation Kerosene)

Fig. 2 The Hyperplane/Avatar transatmospheric hypersonic flight RLV technology demonstrator.

sisting of a space station node in LEO to which space traffic would use reusable spaceplanes. Payloads from the spaceplanes would be transferred to space station based multi-stage LOX/ LH_2 space rockets that return to the space station for reuse. The LOX/LH_2 propellants employed and spare parts/equipment for the multi-stage rockets would also be brought up from earth in the first instance. But closer to earth, what is optimal for revenue earning space missions like space tourism and space solar power?

3.1 Importance of Hydrogen Fuel Fraction

The hydrogen Fuel Fraction (HFF) is defined as the ratio of the mass of liquid hydrogen fuel to the take-off mass of a reusable launch vehicle. It is necessary to see why the HFF for airbreathing spaceplanes is an important attribute and why it has not figured so far in the perceptions of spaceplane and rocketplane designers. For aeroplane designers, fuel fraction is an important design parameter that they wish to maximize, whereas a rocket engineer wants to maximize the propellant fraction that includes oxidizer and fuel.

The spaceplane is basically an aeroplane with a capability to attain orbital height and speed. Jones and Donaldson reported [11] airbreathing aircraft that ascend directly from earth to orbit need at least 56% of take-off weight to be hydrogen fuel, failing which enough kinetic and potential energies will not be available for it to be placed in the earth orbit. In addition to this fuel mass fraction, the aeroplane needs to have a distinct relationship between its propulsive efficiency and the thrust to drag ratio. However, this important finding was overlooked by the aerospace vehicle design community.

Aerospaceplane engineers see this differently. They look at a hypersonic aeroplane as if it were a 100-tonne conventional rocket that would have a payload at best 2 tonnes. Good structural design could ensure an empty structure weight of 13 tonnes while the remaining 85 tonnes would be for propellant. This 85 tonnes would consist of 25% hydrogen fuel (21 tonnes) and 75% oxygen/oxidizer (64 tonnes).

Consider an aerospaceplane that carries no oxygen on board. Its take off weight would be 100-64=36 tonnes, but still containing 21 tonnes, or 58% (21/36) of take-off mass of hydrogen. Jones and Donaldson's requirement of 56% would be met. This was the basis of the Hyperplane/Avatar design concepts. Not carrying LOX on-board at take-off but carrying out air liquefaction and oxygen separation while flying in a Mach 3.5 to Mach 8 regime would result in high payload fractions, varying from 5% to 10% reported for Hyperplane/Avatar (Table 1). Other spaceplane concepts, such as Skylon consider deep air pre-cooling stainless steel heat exchanger technologies without in-flight air and oxygen liquefaction. Hydrogen/air rocket engines designed on similar principles [12-15] are reported to achieve higher specific impulse (about 3500s). These are called Liquid Air Cycle Engines (LACE). Such LACEs using refractory metal heat exchangers are said to enable the engines operate up to Mach 7/8 and steep, optimized trajectory from earth to 80km altitude, after which the main engines are shut down and the vehicle coasts on a ballistic trajectory to a 300 km LEO.

The feasibility of high payload fractions need not be viewed with scepticism and disbelief. Avery and Dugger [16] reported the difficulties of sub-sonic combustion at very high speeds. This led to investigation of new types of engines which can extend the speed of airbreathing vehicles to orbital velocity and uses air collection, condensation and fractionation while the vehicle is flying in the Mach 5-8 range on ramjet power with liquid hydrogen fuel. The oxygen enriched liquid obtained and stored on-board is then burned with the hydrogen fuel in the rocket engine phase to accelerate the vehicle to orbital speeds. The success of such a system depends on development of lightweight air collection and fractionation equipment. Extensive numerical simulation of trajectories and mass fractions of a family of reusable spaceplanes based on the "Hyperplane" concept as a function of launch mass was carried out and the results are shown in Fig. 3. It can be seen from the trend lines that for the Hyperplane-type of spaceplane, the payload fraction increases from about 1% to 10% as a function of launch mass as did the payload-to-structure ratio.

3.2 Structural and Cost Effectiveness Parameters

The parameters that characterise rocket based RLVs, airbreathing/rocket based spaceplanes and expendable rockets are the payload fraction (payload mass/launch mass), propellant fraction (propellant mass/ launch mass) and structure mass fraction (structure mass/launch mass). These three basic parameters are then used to derive parameters as relevant to an aerospacecraft:

3.2.1 Structural Effectiveness Factor (SEF)

Structural Effectiveness Factor (SEF) is defined as the ratio of payload fraction to structure fraction. For any given vehicle:

$$SEF = (M_{PL}/M_o) / (Ms/M_o) = (M_{PL})/(Ms) \qquad (1)$$

A good RLV/Spaceplane design would maximize the SEF. Values of SEF for various spaceplane design concepts are presented in Table 1 and Figs. 4 & 5. It can be seen that higher the HFF, higher is the SEF. This shows the importance of HFF as a critical factor in preliminary concept design.

3.2.2 Technology Effectiveness Index (TEI)

Technology Effectiveness Index (TEI) is defined as the SEF for unit mass of RLV at take-off. For any given vehicle, with an operational life cycle f_L, the TEI may be written as

$$TEI = f_L \times (SEF)/(M_o) \qquad (2)$$

In this analysis, since all the spaceplane concepts examined are normalised to the same operational life cycle, this factor f_L may be ignored.

3.2.3 Cost Effectiveness Index.

The TEI is an element of the cost-effectiveness of a vehicle. For any given vehicle, Cost Effectiveness Index (CEI) = (SLC) x (TEI) $_{where}$ SLC is the specific launch cost.

3.2.4 Relative Technical Cost of Access to Space (RTCAS)

Relative Technical Cost of Access to Space (RTCAS) is obtained from the CEI by normalizing SSTO RLVs to estimated Skylon C2 as reference vehicle at a SLC_{Ref} of \$1300/Kg in LEO [17] (current prices). TSTO RLVs are normalized with respect to Saengar as reference vehicle at a SLC_{Ref} of \$3063/kg in LEO [18, 19].

$$RTCAS = (SLC)_{Ref} \times (TEI)_{Ref}/(TEI) \qquad (3)$$

SEF, TEI and RTCAS values are computed for each vehicle and shown in Table 1 and Figs. 6 & 7. A clear trend of increasing SEF and decreasing RTCAS is seen with high regression coefficients and thus the importance of HFF being used as a measure of comparative evaluation of different spaceplane/ RLV designs is established.

4. MODIFYING THE TSIOLKOVSKY ROCKET EQUATION FOR SPACEPLANES WITH FLOX MASS ADDITION

The Tsiolkovsky (or Ideal) rocket equation relates the maximum change of speed of the rocket (ΔV) to the effective exhaust velocity (V_E) and the initial (M_0) and final (M_E) masses of a rocket or a reaction engine. It considers rocket as a device that can apply acceleration (thrust) to it by expelling part of its mass at high speed in the opposite direction. For any such manoeuver (or flight involving a number of such manoeuvers)

$$\Delta V = V_E \, Ln \, M_o/M_E \qquad (4)$$

The term M_o/M_E =R is called the rocket mass ratio.

The Ideal Rocket Equation is strictly valid only for a constant effective exhaust velocity/specific impulse and in the absence of external forces such as atmospheric drag or gravity

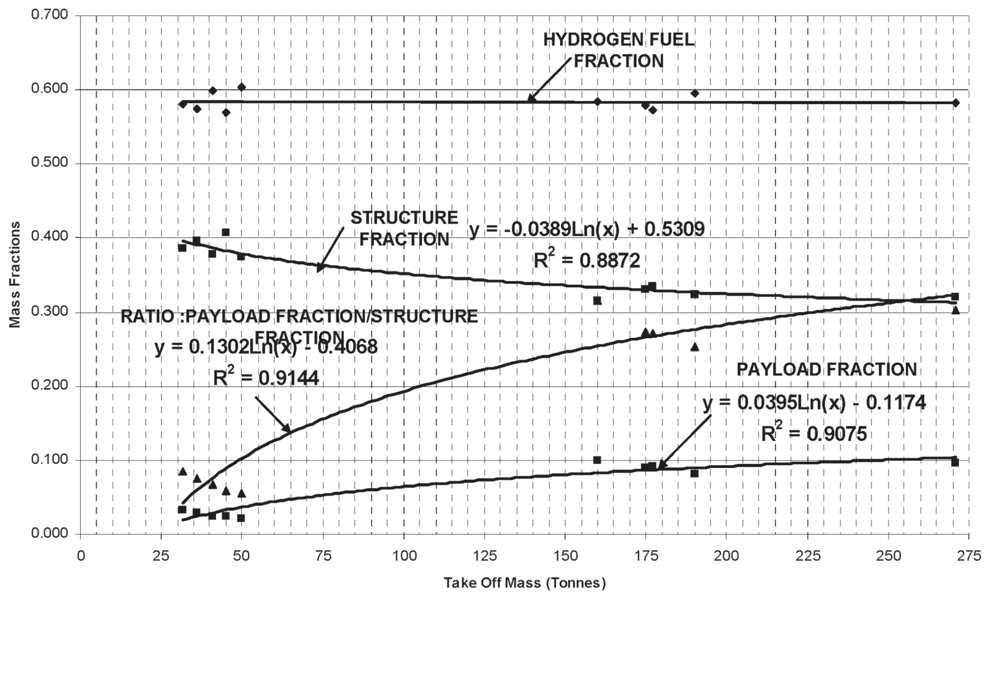


Fig. 3 Numerical simulation of reusable airbreathing spaceplanes with in-flight mass addition.

TABLE 1: *Mass Properties and Parameters of Reusable Launch Vehicles.*

Sl. No	Reusable Launch Vehicle and Country of Origin	Type	Launch Weight (Tonnes) (A)	Hydrogen Fuel Mass Fraction	Structure Mass Fraction (B)	Payload Mass Fraction (C)	SEF =(C/B) x10^{-3} (D)	TEI Technology Effectiveness Index (E) =D/A x 100	RTCAS ($/Kg in LEO) (F) $= (SLC)_{Ref}$ x $(TEI)_{Ref/}$ $(TEI)_V$
1	Japan	TSTO	450	0.12	0.258	0.022	85	18.95	5228
2	Kistler K1 (USA)	TSTO	382	0.133	0.187	0.013	70	18.20	5443
3	Spacecab (UK)	TSTO	400	0.148	0.39	0.01	26	6.41	15453
4	Sanger (Germany)	TSTO	329	0.34	0.47	0.05	106	32.34 $=(TEI)_{Ref}$	3063 $=(SLC)_{Ref}$
5	StarRaker (USA)	SSTO (Heavy Lift)	2279	0.28	0.155	0.044	284	12.46	8643
6	X-33 (USA)	SSTO (Heavy Lift)	730	0.146	0.1	0.025	250	34.25	3144
7	Japan (Mitsubishi)	SSTO (Heavy Lift)	350	0.58	0.25	0.057	228	65.14	1653
8	Skylon C2 (UK)	SSTO (Heavy Lift)	345	0.28	0.154	0.044	286	82.82 $=(TEI)_{Ref}$	1300 $=(SLC)_{Ref}$
9	Skylon C1 (UK)	SSTO (Heavy Lift)	275	0.245	0.154	0.044	286	103.90	1036
10	Hyperplane 271 (India)	SSTO (Heavy Lift)	271	0.58	0.321	0.097	302	111.51	966
11	Hotol UK (Heavy Lift)	SSTO 198	0.246	0.173	0.036	208	105.10	1024	
12	NASP (136) (USA)	SSTO (Heavy Lift)	136	0.56	0.33	0.11	333	245.10	439
13	Avatar	SSTO (Small)	25	0.6	0.35	0.05	143	571.43	188
14	Black Horse (USA)	SSTO (Small)	22	0.14	0.34	0.04	117.65	534.76	201
15	GSLV (India)	Expendable	402	0.14	0.136	0.034	250	62.19	Expendable
16	Ariane (France)	Expendable	467	0.15	0.06	0.02	333	71.3	Expendable
17	Space Shuttle (USA)	Partially Reusable	2000	0.1	0.376	0.014	37	1.85	Partially Reusable

and decreasing mass of the rocket. It cannot be used for mass addition occurring in flight that is feasible due to advanced technologies and system concepts.

Shepherd [20] modified the rocket equation to take into account variable external forces of gravity and drag, but with constant exhaust velocity. Since g varies with altitude, and drag varies with speed and Reynolds number, Shepherd simplified and rearranged the rocket equation by assuming zero drag, constant gravity and constant flight path angle.

Bayer [21] went a step further in an attempt to apply the ideal rocket equation more effectively for winged spaceplanes with air breathing engines and with variable specific impulse (hence variable exhaust velocity). Bayer's final derivation assumed the form of a propulsive equation for variable specific impulses.

These modified rocket equations are useful only when mass decreases due to propellant consumption. A further modification of the ideal rocket equation is necessary for spaceplanes that increase their mass in flight. This derivation is given in the

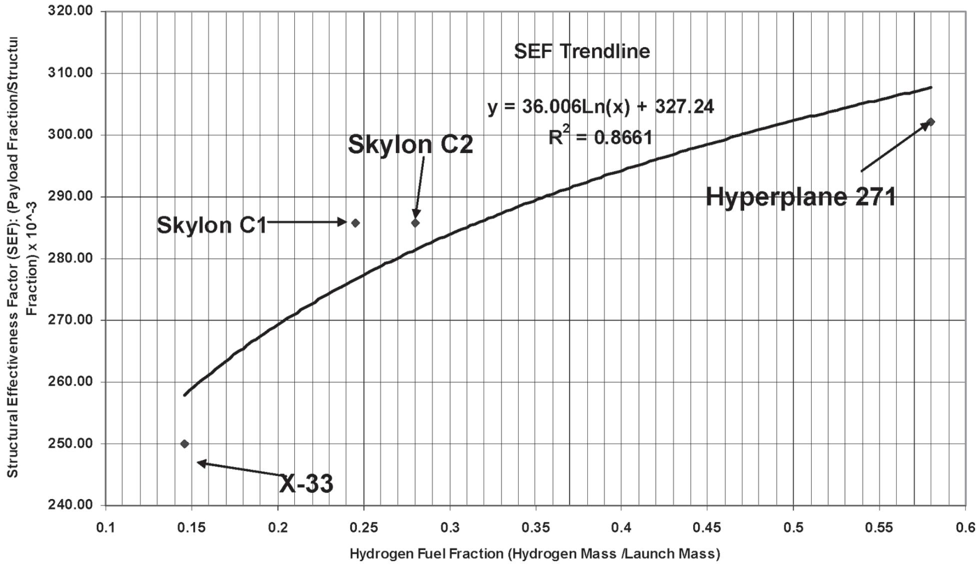


Fig. 4 Structural effectiveness factor of heavy lift SSTO launch vehicles.

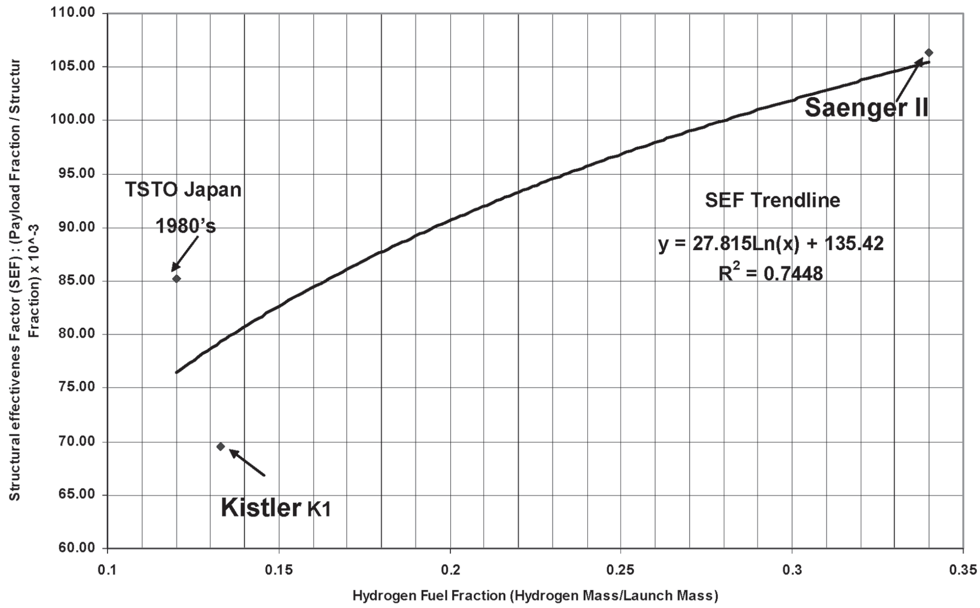


Fig. 5 Structural effectiveness of TSTO vehicles.

Note at the end of the paper. It introduces a 'spaceplane mass ratio multiplier factor' [22], for analytical treatment of spaceplanes with mass addition in flight.

4.1 Application of Spaceplane Equation

Airbreathing spaceplanes in the last part of the 20th century emerged with the following alternate concepts to avoid carrying LOX as an inert mass onboard from take-off when flight to orbit could more effectively make use of the earth's atmosphere. These were:

In-flight re-fuelling at subsonic speeds and altitudes of 15-20 kms e.g. the Black Horse Spaceplane [23] that uses a rocket engine with hydrogen peroxide/aviation kerosene as propellants. It takes-off with just sufficient oxidizer only till aerial tanker rendezvous. High density of oxidizer and very high oxidizer-to-fuel mixture ratio enabled compact spaceplane design, but the penalty is in the use of low efficiency propulsion systems.

Air-breathing rocket engines up to hypersonic speeds (Mach 5) and 28kms altitude followed by pure LOX/hydrogen rocket engine: the Skylon takes off with 25% hydrogen fraction and 55% liquid oxygen fraction on board. Fuel efficiency is enhanced several fold but flight in atmosphere is restricted to Mach 5. In this case, the real mass of the vehicle does not increase in flight, but an equivalent oxygen mass addition may be estimated from the air mass flow rate through the engine while in airbreathing mode of ascent

Airbreathing hydrogen fuelled LACE/turbojet/turboramjet engines with high fuel efficiency up to Mach 7/8 and 30kms altitude, followed by LOX/hydrogen rocket engine. An example is the Hyperplane/Avatar concept for take-off with 58-60% hydrogen fraction and zero liquid oxygen on board and entire LOX mass is added in high speed flight.

4.1.1 Parametric Mapping of Spaceplane Designs

Novel spaceplane design and technology domains emerge from parametric mapping using the spaceplane equation that could synergize more effective design and development of advanced space transportation systems. For example, this equation enables scaling aerocryogenic spaceplanes by numerical simulation for a 'first cut" appreciation of different concepts. Typical results are shown in Fig. 8 and have high values of regression

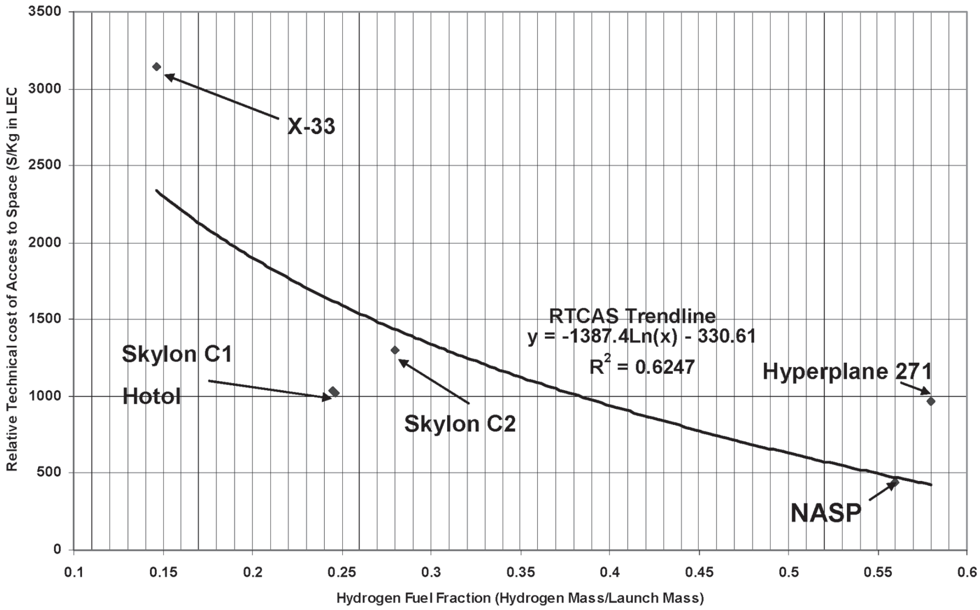


Fig. 6 Relative technical cost of access to space (SSTO vehicles)

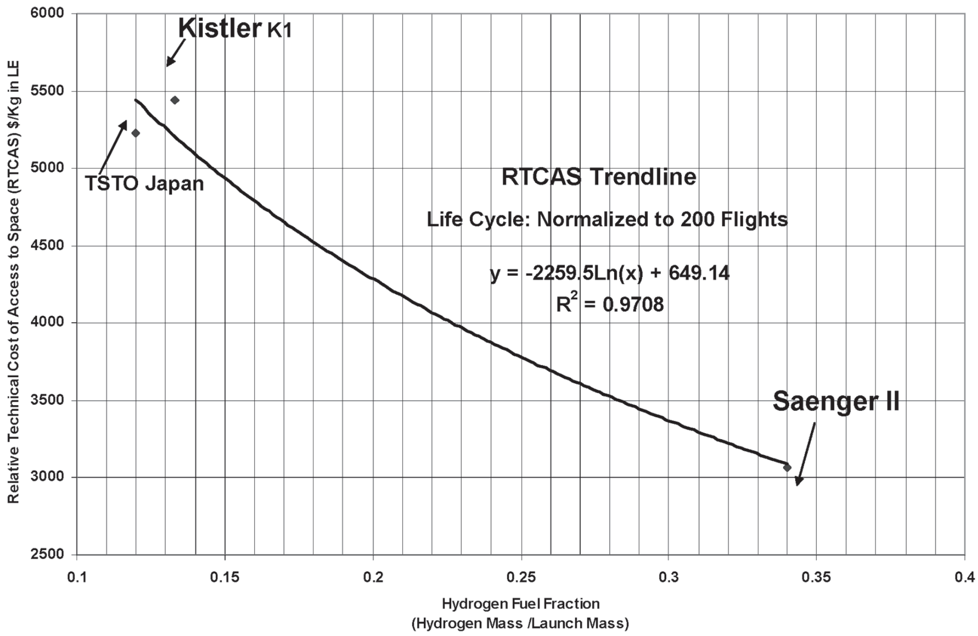


Fig. 7 Relative technical cost of access to space (TSTO vehicles).

coefficient (>98%). The costs and risks to develop and test heavy lift aerospace transportation systems are indeed high. In the light of scalability of spaceplanes with aerocryogenic engines, it may be worthwhile for such conceptual design perspectives as scaling a family of geometrically similar spaceplanes be examined for future missions.

4.2 Cost of Access to Space

The cost of access to space (as a function of payload mass flow to orbit in tonnes per year) demanded by various classes of revenue earning "mass missions" has been reported by Ashford [24] in the form of a space transportation demand curve. Data extracted from his curve are placed in Table 2. Ashford's estimation of cost of access to space is about \$100-200/kg for payload mass flows ranging from about 1000 to 50,000 tonnes per year. Mankins reported space solar power with a cost of access to space of \$200/kg in LEO [25].

In assessing the cost of access to space, a clear distinction is required between the mission cost and technical cost. The mission cost includes specific operational factors like life-cycle and flight rates, and economic factors like amortization of R&D investments spread over a total production run, traffic levels and fleet size, as these are extremely important considerations for a specific engineering and investment proposal. The technical cost is drawn from purely design factors.

In this paper, the basic economic analysis principle of "other factors remaining the same" is adopted while considering mission-related costs of access to space. However, the life-cycle of spaceplanes is both a technical factor as well as an economic factor, unlike fleet size, flight rate, development investment amortization etc. Hence the life-cycle factor could be considered in the estimations of relative technical cost.

Among SSTO reusable spaceplanes studied, the Skylon C2 assumes a life-cycle of 200 flights and a specific launch cost of \$1300 per kg in LEO based on fixed and variable costs spread over vehicle life cycle of 200 flights [3, 17].In the case of the Saenger TSTO, the first stage is designed as a passenger air-

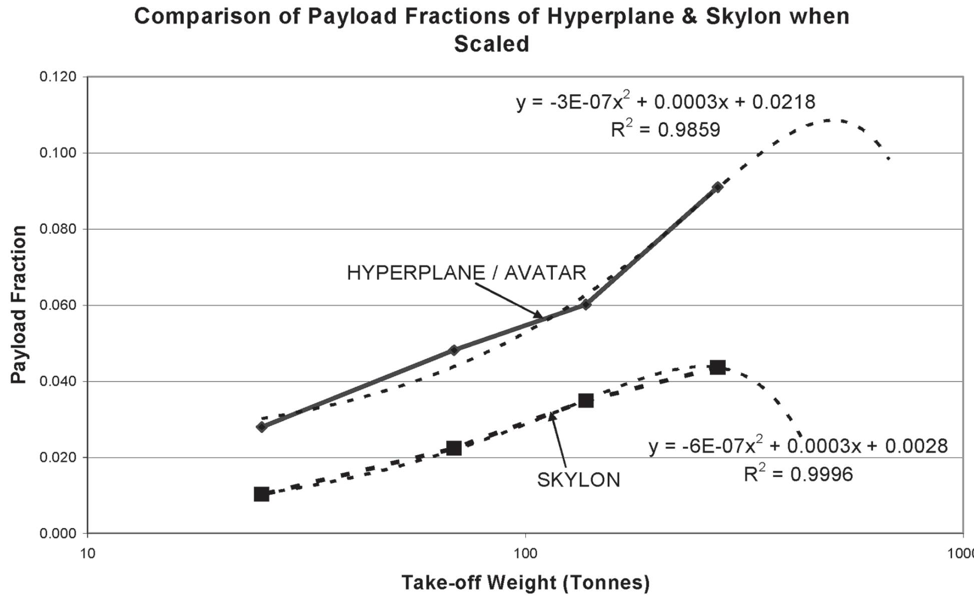


Fig. 8 Typical result of numerical analysis using the spaceplane equation

craft, and the second stage alone is a reusable spaceplane. The passenger aircraft will experience some 20,000 take-off and landings and 50,000 hours flight time whereas the second stage space launch vehicle will be exposed to only 150-200 launches and 500-1000 hours flight time [19] with a specific launch cost of $1867/kg in LEO at 1985 prices (or $3063/kg current prices at 2% inflation rate over 25 years).

In this analysis, all spaceplane comparisons are normalized to a life cycle of 200 launches, and Saenger/Skylon specific launch costs for TSTO/SSTO respectively. Values of RTCAS for various spaceplane designs grouped as heavy lift TSTO, heavy lift SSTO are presented in Figs. 7 & 8 and for the two small SSTO spaceplanes in Table 1.

To illustrate the difference between mission and technical costs, an example is provided: The StarRaker [26] is a 'super-heavy' lift SSTO with a launch mass of 2279 tonnes (Payload 100 tonnes). The StarRaker is thus not only a different class of launch vehicle because of its extremely large size but also due to altogether different mission profiles. Its estimated RTCAS is $7576 per kg, that is far greater than all other design concepts in the heavy lift SSTO class, unlike the Skylon C2 with launch mass of 275 tonnes (12 tonnes payload) with a RTCAS of $1300 per kg. Yet, both RLVs have almost the same HFF and same SEF. But since mission characteristics are completely different (StarRaker operations calling for 584,000 tonnes payload/year and Skylon 2400 tonnes/year, the mission specific cost of StarRaker is quoted at $33 per kg in LEO (current prices) as against Skylon's $1300 per/kg in LEO. Hence the need for distinction to be drawn between *mission* cost and *technical* cost.

5. AEROCRYOGENIC TECHNOLOGIES: KEY TO SAFE & AFFORDABLE SPACEFLIGHT

5.1 Enhancing Engine Fuel Efficiency

Air cooling and air liquefaction technologies with compact, lightweight, cryogenic heat exchangers are components of LACE engines. They are useful to enhance rocket engine fuel efficiency. Air cooling (in the case of the Skylon engine) gives rise to engine specific impulse in the range of 2500s [27,28] that operate till Mach 5; addition of air liquefaction enhances specific impulse further up to 3500s [12-15] that operate up to Mach 8. Since some part of LOX mass needed for exo-atmospheric flight of the vehicle is avoided by pre-cooled airbreathing engines, the take-off mass of the vehicle for a given payload is reduced and the HFF at take-off is increased to a smaller extent (from 20-25%) since a large quantity of LOX for exoatmospheric flight is still required on board at take-off.

5.2 Step-Jump for High Hydrogen Fuel Fraction

FLOX aerocryogenic systems enable a quantum-jump in enhancement of HFF (56-60%). The cryogenic nitrogen released after LOX separation further enhances the cooling capacity

TABLE 2: *Cost of Access to Space for Revenue Earning Space Missions.*

Mission	Application Areas	Payload Mass Flow (Tonnes/Year)	Mission Cost of Access to Space ($/kg)
Information Missions	Communications, Meteorology, Navigation, Earth Resources	100	20,000 -2000
Mass Missions (Near Term)	Space Tourism Space Manufacture	1000-100,000	2000 – 200
Mass Missions (Far Term)	Space Solar Power Space Colonization Mining in Space	100,000 to 10 Million	200-10

available from liquid hydrogen alone. The issues of liquefaction (momentum-loss) drag, purity of LOX after separation process and overall weight of the aerocryogenic system need to be addressed comprehensively at the stage of spaceplane flight performance and flight path design for which multi-variable optimization techniques are essential.

A fair amount of design and experimental work has been carried out in India [29-35] including numerical simulation of trajectories for airbreathing SSTO vehicles with FLOX systems; and system preliminary modelling, design and heat and mass balance optimization for a reusable hypersonic transatmospheric flight test vehicle to demonstrate in-flight oxygen liquefaction at 1kg/s with on-board storage. Detailed aero-thermo-kinematic optimization of a complete in-flight air liquefaction and oxygen separation system to minimize vehicle hydrogen fuel consumption in hypersonic flight has also been studied. Detailed in-flight performance FLOX process modelling studies have been carried out. For FLOX systems preliminary design, the specific weights, specific volume and specific frontal areas of in-flight air liquefaction and oxygen separation systems have also been reported. The term “FLOX” was also adopted by Russia (CIAM) [36] and a concept was presented using FLOX for Saenger.

5.3 Air Cooling

Remarkable progress in design, development and production of stainless steel and inconel cryogenic heat exchangers for air cooling up to a speed of Mach 5, temperatures up to 1300K i.e. the precooler, has been accomplished in the UK through the Sabre engine and Skylon vehicle concept [3,4, 27, 28]. Major problems like frosting of the heat exchanger during ascent flight in the atmosphere have also been successfully addressed. This trail-blazing, two-decade long commitment to aerocryogenic systems is continuing with increasing interest and support from the UK government and ESA.

5.3.1 Air Liquefaction Technologies

Complex aerocryogenic process flow systems and technologies like hydrogen tank-return for cooled-air condenser stage, air compressor and air spray to improve efficiency are required to maximize fuel efficiency. Mitsubishi has made some progress in this technology for speeds up to Mach 7/8. Further, the LACE engines designed in Japan advocate more complex engineering by use of “slush” hydrogen with tank-return flow path to maximize the heat sink capacity in the liquid hydrogen tanks

5.3.2 LOX separation

Design and preliminary engineering studies were made to develop a flight-rated in-flight LOX collection system for use in a spaceplane hypersonic flight test bed [10]. Considerable theoretical and experimental progress in vortex separator technology has been reported [32]. Some theoretical work on both types of separators has also been carried out [33, 34]. This work has been done with computer-aided multi-variable numerical simulation studies on the flight performance of hypersonic reusable flight vehicles [35]. Substantial work on LOX separators and air collection systems were carried out in Russia [36-38]. Pioneering work on air cooling, air liquefaction and LOX separation technologies and alleviation of performance problems of aerocryogenic technologies in flight, were done in the US [39–42].

5.4 Evolution of Liquid Air Cycle Engines

The LACE propulsion system was first proposed by Marquardt in the early 1960s. The simple LACE engine exploits the low temperature and high specific heat of liquid hydrogen in order to liquefy the captured airstream in a specially designed condenser. Following liquefaction the air is relatively easily pumped up to such high-pressures that it can be fed into a conventional rocket combustion chamber. The main advantage of this approach is that the airbreathing and rocket propulsion systems can be combined with only a single nozzle required for both modes. This results in mass saving and a compact installation with efficient base area utilization. Also, the engine is in principle capable of operation from sea level static conditions up to perhaps Mach 5-6 [27].

Over the last two decades LACE development continued in the UK and Japan at engine technology and experimental level. While JAXA had focused on LACE engines with high air liquefaction ratios using rhenium coated heat exchanger tubes for use up to Mach 9 and more complex LACE cycles using tank-return, air compressor and liquid air spray, the UK Reaction Engines focused on compact light weight stainless steel heat exchangers up to Mach 5 with air cooling. Experimental work in India was carried out on LOX liquefaction from flowing streams of liquid air using vortex separators.

The effectiveness of compact heat exchangers is a function of the cooling capacity available in liquid hydrogen, both by way of enhancing its heat capacity in hydrogen tanks by use of ‘slush hydrogen’ as well as novel design of finned heat exchanger tubes to increase heat transfer rates. The cooling capacity of liquid hydrogen is further enhanced by additional heat exchangers of the FLOX system using cryogenic nitrogen released after lox is separated from the liquid airstream.

This brief review of LACE system and its aerocryogenic technologies places key developments in perspective in the light of the concept of increasing HFF. It can be seen that a LACE-FLOX engine cycle with LACE Heat Exchangers for air liquefaction integrated with LOX vortex/higee separators, both operating from Mach 3.5 to Mach 7/8 would yield spaceplane concept designs with very high payload fractions.

6. CONCLUSIONS

In comparing the spaceplane programmes, Buffo identified seven attributes to describe a spaceplane. Reviewing the critical factors in conceptual designs of spaceplanes that emerged thereafter, this paper shows that two more attributes should be added to clearly distinguish spaceplanes from rocketplanes. One of them is the hydrogen fuel fraction (HFF) at takeoff, while the other is ground testability of components and systems.

Regression equations for trends in parameters of structural effectiveness and relative cost of access to space as a function of HFF for different classes of spaceplanes are studied. High values of regression coefficients are also obtained from HFF trend lines, indicating increasing structural effectiveness and decreasing relative cost of access to space as HFF increases. Similar high regression coefficients are obtained from numerical simulation results from application of a spaceplane equation indicate that spaceplanes with advanced aerocryogenic propulsion systems (such as Skylon and Avatar) are scalable, enabling the least risk-least cost development strategy.

A global status assessment for aerocryogenic technologies that enables high hydrogen fuel fraction at take-off has been provided. International cooperation to develop LACE and FLOX technologies towards an integrated LACE-FLOX spaceplane is suggested, starting with small spaceplanes. A cooperative development strategy could emerge with the capability to provide a combined-cycle engine from earth-to-orbit with air collection and air liquefaction (LACE heat exchangers) integrated with liquid oxygen vortex/higee separators. An advanced LACE – FLOX engine development strategy may be blended gainfully with the Ashford [43] strategy for developing the first orbital spaceplane soon and at low cost and risk using an aviation approach, starting with small, scaled down geometrically similar spaceplanes with LACE-FLOX aerocryogenic propulsion systems

ACKNOWLEDGEMENT

The author is deeply grateful to Dr R.Krishnan, former Director, Gas Turbine Research Establishment Bangalore for sharing his scientific insights and valuable technical observations. He thanks Mr. Keith Gottschalk, eminent RLV historian and academician at the University of Western Cape, South Africa for his global RLV perspectives, moral-boosting reviews and stimulating comments. Sincere thanks are due to Mr. Richard Varvill, Technical Director & Chief Designer, Reaction Engines Ltd, UK, for his patient counsel and frank comments related to hydrogen fuel fraction. Thanks are due to Mr. P.Venugopalan, Director DRDO for according permission to cite work done in DRDL on Hyperplane/Avatar concepts and to Mr. M.Thirumurthi of DRDL for his dedicated help in retrieving archival published work.

REFERENCES

1. M. Buffo "Technical Comparison of Seven Nations Spaceplane Programs", *AIAA Space Programs and Technologies Conference*, September, 1990. AIAA-90-3674-CP.
2. "HOTOL: Europe's New Aerospace Plane: A Concept for the 21st Century", *Chanakya Aerospace*, pp.33-35, May 1985.
3. R. Varvill and A. Bond, "The Skylon Spaceplane: Progress to Realization", *JBIS*, **61**, pp.412-418, 2008.
4. M. Hempsell and R. Longstaff, "Skylon Users Manual", Reaction Engines, Abingdon, UK, Document No. SKY-REL-MA-001, Revision: Version 1, September 2009.
5. R. Gopalaswami, S. Gollakota and P. Venugopalan "A Single-Stage to-Orbit Vehicle Hyperplane", *J. Aeronautical Society of India*, **40**, pp.1-14, 1988.
6. R. Gopalaswami *et al.*, "Concept Definition and Design of a Single-Stage-To-Orbit Launch Vehicle "Hyperplane", *39th International Astronautical Congress*, Bangalore, India, 1988.
7. R. Gopalaswami, S. Gollakota and P. Venugopalan "Avatar: An Aerospacevehicle Technology Demonstrator for Low Cost Access to Space", *National Symposium on Low Cost Access to Space*, Astronautical Society of India, Bangalore, India, September 1996.
8. R. Gopalaswami, "The 'Avatar' Mission: True Pathfinder from Mini-Aerospaceplane to Multi-Mission Hypersonic Mass Transportation", *Proceedings of the International Seminar on Futuristic Aircraft Technologies*, Aeronautical Society of India, December 1996.
9. R. Gopalaswami, Prahlada, Satish Kumar & Y. Satyanarayana, "Spaceplanes with Aerobic Propulsion – Key to Low Cost access to Space", July 2001 and International Aerospace Abstracts (IAA) 2001. AIAA 3699-2001.
10. V.K. Saraswat, P. Venugopalan, Satish Kumar and R. Gopalaswami, "A Multi-role Flight Technology Demonstrator (HTV) for Advanced Air and Space Transportation Systems Development", *International Conference on High Speed Transatmospheric Air and Space Transportation*, Hyderabad, India, June 2007.
11. R.A. Jones and C du P Donaldson, "From Earth to Orbit in a Single Stage", *J. Aerospace America*, **25**, pp.32-34, 1987.
12. H. Hirakose, T. Auki and T. Ito, "A Concept of LACE for Space Plane to Earth Orbit", *J.Hydrogen Energy*, **13**, pp.495-505, 1990.
13. A. Ogawara and T. Nishiwaki, "The Cycle Evaluation of Advanced LACE Performance", *40th International Astronautical Congress*,Malaga, Spain, 7-12 October 1989. Paper No. IAF-89-313.
14. T. Aoki and A. Ogawara, "Study of LACE Cycle for SSTO Space Plane", *39th International Astronautical Congress*, Bangalore, India, 1988. Paper No. IAF-88-252.
15. H. Nouse, M. Minoda, R. Yanagi, T. Tamaki and T. Fujimura, "Conceptual Study of Turbo-Engines for Horizontal take-off and Landing Space Plane", *39th International Astronautical Congress*, Bangalore, India, 1988. Paper No. IAF-88-253.
16. W. H. Avery and D. L. Dugger, Applied Physics Laboratory, John Hopkins University, "Hypersonic Airbreathing Propulsion", *J. Astronautics & Aeronautics*, pp.42-47, 1964.
17. Reaction Engines Ltd. United Kingdom, "Solar Power Satellites and Spaceplanes: The SKYLON Initiative" September 2008.
18. D.E. Koelle and C. Kukzera, "Saenger II – an Advanced Launcher System for Europe", *Acta Astronautica*, **19**, pp.63-72, 1989.
19. Saenger II Spaceplane, www.astronautix.com/lvs/saegerii.htm. (Date Accessed 4 July 2011)
20. D.G. Shepherd, "*Aerospace Propulsion*", American Elsevier, 1972.
21. M. J. Bayer MJ "Analytic Performance Considerations for Lifting Ascent Trajectories of Winged launch Vehicles", *Astra Astronautica*, **54**, pp.713-721, 2004.
22. R. Gopalaswami, "The Spaceplane Equation", *Online Journal of Space Communications*, Issue 16, Solar Power Satellites, 2009, http://spacejournal.ohio.edu/issue16/gopal2.html. (Date Accessed 4 July 2011)
23. M.B. Clapp, "Black Horse: In-Flight Propellant Transfer Spaceplane Design and Testing Considerations", Phillips Laboratory, Kirtland AFB, NM 87117, http://www.islandone.org/Launch/BlackHorse-PropTransfer.html. (Date Accessed 4 July 2011)
24. D.M. Ashford, "Spacecab II: Low Cost Small Shuttle for Britain", *Aerospace*, February 1986.
25. J.C. Mankins, Statement before the Subcommittee on Space and Aeronautics Committee on Science U.S. House of Representatives, 7 September 2000.
26. StarRaker, http://www.astronautix.com/lvs/staraker.htm. (Date Accessed 4 July 2011)
27. R. Varvill and A. Bond, "A Comparison of Propulsion Concepts for SSTO Reusable Launchers", *JBIS*, **56**, pp.108-117, 2003.
28. J. J. Murray, C. H. Hempsell and A. Bond, "An Experimental Precooler for Airbreathing Rocket Engines", *JBIS*, **54**, pp.199-209, 2001.
29. R. Gopalaswami, "Aerothermokinematic Optimization to Minimize Fuel Consumption in Hypersonic Aerobic Flight", *International Conference on High Speed Transatmospheric Air and Space Transportation*, Hyderabad, India, June 2007.
30. V. C. Madhav Rao, "Process Analysis and System Design of an In Flight Oxygen Collection System for a Hypersonic Aerospace Vehicle", Report, Department of Mechanical Engineering, JNTU College of Engineering, Hyderabad, 2002.
31. R. Gopalaswami and K. Srinivasan, "Systems Preliminary Design of an In-flight Air Liquefaction and Oxygen Separation System for a Hypersonic Flight Test Vehicle", *International Conference on High Speed Transatmospheric Air and Space Transportation*, Hyderabad, India, June 2007.
32. S. Jacob *et al.*, "Design & Development of Vortex Tube Separator", IISc, Bangalore, Report 28, May 2002.
33. D.P. Rao, A. Bhowal & N. Verma, "Design of Vortex Separator: Alternate Means of Oxygen Liquefaction", Department of Chemical Engineering, IIT (K), Report March 2001.
34. S. S. Bandhopadyay, "Conceptual Design of In-Flight Liquid Oxygen System – Final Technical Report", Separation Science Laboratory, Engineering Centre, IIT (Kgp.), October 2002.
35. Y. Satyanarayana, R. Srilakshmi and R. Gopalaswami, "Numerical Simulation Techniques for Group Concurrent Design of Hypersonic Transatmospheric Vehicles", *International Conference on High Speed Transatmospheric Air and Space Transportation*, Hyderabad, India, June 2007.
36. V.V. Balepin *et al.*, "Integrated Air Separation and Propulsion System for Aerospaceplane with Atmospheric Collection", SAE Technical Series Paper No. 920974, 1992.
37. V.V. Balepin *et al.*, "Cryogenic Heat Exchange Key Technologies for Precooled Turbojet Engine", SAE Technical Series Paper No. 911183, 22-26 April 1991.

38. V.V. Balepin, "Air Collection Systems" in *Developments in High-Speed-Vehicle Propulsion Systems*, *AIAA Progress in Aeronautics & Astronautics*, **165**, pp.385-419, 1996.
39. C.F. Gottzmann *et al.*, "Feasibility Study of a High Capacity Distillation Separator for an air Enrichment System", Union Carbide Corporation ASD-7DR-63-665 Part II, February 1964.
40. Lourdes Q. Maurice, John L. Leingang and Louis R. Carreiro, "Airbreathing Space Boosters Using In-Flight Oxidizer Collection", *28th Joint Propulsion Conference and Exhibit*, Nashville, TN, 6-8 July 1992.
41. W.H. Bond *et al.*, "Prospects for Utilization of Air Liquefaction and Enrichment System (ALES) Propulsion in Fully Reusable Vehicles", *29th Joint Propulsion Conference and Exhibit*, Monterey, CA, 28-30 June 1993. AIAA-93-2025, 1993.
42. J. Vandenkerchove and P. Czysz, "SSTO Performance Assessment with In-Flight LOX Collection", *Acta Astronautica*, **37**, pp.167-178, 1995.
43. D. Ashford, "An aviation approach to space transportation", *The Aeronautical Journal*, **113**, pp.499-515, 2009.

NOTE
DERIVATION OF THE SPACEPLANE EQUATION [22]

The basic rocket equation is

$$\Delta V = V_E \text{ Ln } M_o/M_E \tag{1}$$

The rocket mass ratio $R = M_O/M_E$.

To attain earth orbital velocity of 7800 metres/sec, the ideal mass ratio from (1) $R_I = 5.852$ from (1) above. This is not feasible in a single-stage as practical considerations limit the stage mass ratio to values < 3.5. To increase the mass ratio of a rocket vehicle, several single stages are stacked vertically. For a multi-stage rocket vehicle, the mass ratio factor in the basic rocket equation is modified as

$$R^* = r_1 \text{ x } r_2 \text{ x } r_3 \ldots . r_n \tag{2}$$

Where r_1, r_2 etc. are the mass ratios of each stage.

The rocket equation for a multi-stage rocket can then be written as:

$$\Delta V = g_o .\ I_{sR} \text{ Ln } \{\varepsilon .\ R_I\} \tag{3}$$

where $= R^*/ R_I$ the rocket mass ratio multiplier factor.

The term ε shows that the mass ratio of a single stage (that has practical values from 2.5 to 3.5) is amplified using multi-stage rockets. A three-stage rocket, each stage with R=2.5, the "Mass Ratio Amplifier" $\varepsilon = (2.5)^3/5.85 = 2.67$.

For a spaceplane, ΔV is provided in two distinctly different flight regimes, the endoatmospheric regime (A) where lift and drag losses play a significant role and the exoatmospheric regime in space (C) where gravity losses play a significant role. Hence

$$\Delta V = \Delta V_A + \Delta V_R \tag{4}$$

Since exhaust velocity $V_E = g_o .I_{sp}$, Eqn. (1) can be rewritten as,

$$\Delta V = g_o.I_s.\text{Ln}(M_o/M_E) \tag{5}$$

where go is 9.81 metres/sec^2 and Isp is specific impulse of the propulsion system. Hence, for the airbreathing spaceplane

$$\begin{aligned}\Delta V &= g_o.I_{sA}.\text{Ln}(M_A/M_0) + g_o.I_{sR}.\text{Ln}(M_A/M_E)\\ &= g_o.\ I_{sR}\ \{(I_{sA}/I_{sR}) \text{ Ln } (M_A/M_0) + \text{Ln } (M_A/M_E)\}\end{aligned}$$

Let

$\beta = I_{sA}/I_{sR}$ (airbreathing/rocket engine specific impulse ratio)

$R_A = MA/M_0$ (airbreathing phase mass addition ratio)

$R_R = M_A/M_E$ (rocket phase mass r)

Then,

$$\begin{aligned}\Delta V &= g_o.\ I_{sR}\ \{\beta.\ \text{Ln } (M_A/M_0) + \text{Ln } (M_A/M_E)\}\\ &= g_o.\ I_{sR}\ \{\text{Ln } (R_A)^{\beta} + \text{Ln } (R_R)\}\end{aligned}$$

Taking

$$\zeta = (R_A)^{\beta}$$

We have

$$\Delta V = g_o.\ I_{sR}\ \{\text{Ln } \zeta + \text{Ln } (R_R)\}$$

or,

$$\Delta V = go.\ I_{sR} \text{ Ln } \{\zeta .\ R_R\} \tag{4}$$

The term ζ is termed as spaceplane mass ratio multiplier factor, analogous to ε, rocket mass ratio multiplier factor.

(Received 19 April 2010; 13 December 2010; 11 March 2011)

* * *