JOURNAL OF PROPULSION AND POWER
Vol. 10, No. 4, July–Aug. 1994

Air Liquefaction and Enrichment System Propulsion in Reusable Launch Vehicles

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A concept is shown for a fully reusable, Earth-to-orbit launch vehicle with horizontal takeoff and landing, employing an air-turborocket for low speed and a rocket for high-speed acceleration, both using liquid hydrogen for fuel. The turborocket employs a modified liquid air cycle to supply the oxidizer. The rocket uses 90% pure liquid oxygen as its oxidizer that is collected from the atmosphere, separated, and stored during operation of the turborocket from about Mach 2 to 5 or 6. The takeoff weight and the thrust required at takeoff are markedly reduced by collecting the rocket oxidizer in-flight. This article shows an approach and the corresponding technology needs for using air liquefaction and enrichment system propulsion in a single-stage-to-orbit (SSTO) vehicle. Reducing the trajectory altitude at the end of collection reduces the wing area and increases payload. The use of state-of-the-art materials, such as graphite polyimide, in a direct substitution for aluminum or aluminum-lithium alloy, is critical to meet the structure weight objective for SSTO. Configurations that utilize "waverider" aerodynamics show great promise to reduce the vehicle weight.

Nomenclature

E = energy
F = force
g = gravitational constant
I = impulse
I_{sp} = specific impulse
M = Mach number
m = mass
$\mathring{m}$ = nozzle exhaust mass flow rate
P = power or pressure
P_{t0} = freestream stagnation pressure
P_{t2} = diffuser exit stagnation pressure
q = flight dynamic pressure
U_{ts} = ultimate tensile stress
V = velocity
V_e = nozzle exit velocity
V_0 = initial velocity
W = weight (or mass)
W_0 = vehicle initial mass
W_1 = vehicle orbital mass
ΔV = velocity increment
ρ = density

Introduction

GROWTH in the use of outer space is critically dependent on the cost of getting there. There is no lack of missions waiting to be accomplished. Also, with each passing year, new and exciting missions are proposed. However, with the cost of several thousand dollars per pound to deliver material to low Earth orbit, many ideas for the utilization of space are too costly to be practical. This article explores the prospects of a practical single-stage-to-orbit (SSTO) system with horizontal takeoff (HTOL) and landing to significantly reduce the cost of Earth to low orbit transportation.

In previous studies it was shown that an air liquefaction and enrichment system (ALES) is an attractive propulsion system for a low-cost Earth-to-orbit transportation system.[1,2] The study concluded that a fully reusable two-stage-to-orbit (TSTO) HTOL launch system is feasible with 1990 technology and materials. The gross take-off weight (GTOW) of the ALES TSTO system, capable of delivering 10,000 lb (4500 kg) of payload to a 100-n.mi. (185-km) polar orbit, was found to be about 550,000 lb (250,000 kg); about half the GTOW of a non-ALES TSTO vehicle.

In the same study, however, it was also concluded that an ALES SSTO system may not be practical with 1990 design experience and materials technology, and current ALES propulsion system design, operations and assumed mass properties. However, the SSTO may be feasible with *either* aggressive technology and advanced materials, possibly derived from the National AeroSpace Plane (NASP), *or* advanced vehicle design concepts such as blended body-wing or waverider vehicles.

In this study, the study was further extended to define a low-cost transportation system with an initial operating capability (IOC) in the 2005–2010 time period. The IOC time period, 2005–2010, is selected with the following rationale:

1) Current and additional expendable rockets will be available at about today's cost for at least 20 more years.
2) The current Shuttle system, modestly improved, will continue to be the manned reusable system until another emerges with significantly lower operating cost.
3) Development of a replacement system is only desirable if large operating economy can be expected.
4) Because of the large cost of developing and demonstrating a major new system, extensive small-scale tests and evaluation should precede program commitment, and therefore, a significant time from the present to IOC is expected.
5) The technologies of vehicle health management, lightweight reusable structures, air turborockets, and air collection and separation need to be further developed.

As corollary objectives to obtaining low operating cost, the integrated vehicle-propulsion system should exhibit: 1) high overall effective specific impulse, 2) quick and inexpensive turnaround between missions, 3) minimum ground operations and facility requirements, and 4) feasible development and test facilities that adequately simulate all flight conditions. Derived from these and economic considerations, the final characteristic objectives include capability to deliver useful

Presented as Paper 93-2025 at the AIAA/SAE/ASME/ASEE 29th Joint Propulsion Conference and Exhibit, Monterey, CA, June 28–30, 1993; received Aug. 14, 1993; revision received Oct. 28, 1993; accepted for publication Nov. 11, 1993.

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payload fractions to a low Earth orbit in a reusable single stage with horizontal takeoff and landing. These characteristics can only be achieved with a propulsion system possessing both high overall specific impulse and acceptable specific weight.

Contemporary chemical rocket technology [e.g., Space Shuttle main engine (SSME)] approaches the maximum practical specific impulse attainable for chemical reactants. This precludes large improvement in the performance of vehicle systems using conventional chemical rocket propulsion. We have assessed the capabilities and assessed the development status of candidate classes of propulsion systems, and one of the most promising is selected for more depth of study. Because of the paramount importance of propulsion performance to the solution of this problem, this discussion is primarily propulsion oriented.

Propulsion System Concepts

The candidate propulsion concepts can be placed into two categories: 1) systems that employ propellants with specific energy greater than that available from LOx-hydrogen combustion, or 2) systems that use mass augmentation from the atmosphere.

These two categories follow from the basic energy and momentum equations defining propulsion performance. The energy equations

$$E = \tfrac{1}{2}mV_e^2 \quad \text{(for a rocket)} \tag{1}$$

or

$$E = \tfrac{1}{2}m(V_e^2 - V_0^2) \quad \text{(for an airbreather)}$$

indicate that as the exhaust velocity is increased, energy must be supplied as the square of the exhaust velocity. The momentum, or thrust equations

$$F = \dot{m}V_e \quad \text{(for a rocket)} \tag{2}$$

or

$$F = \dot{m}(V_e - V_0) \quad \text{(for an airbreather)}$$

state that thrust increases to only the first power of exhaust velocity V_e.

Since the thrusting mass rate $\dot{m}$ occurs to the first power in both energy and momentum equations, it can be inferred that, in an energy limited system, the way to improve energy efficiency is to increase the amount of mass flow per unit of energy expended. An outstanding example of this is the trend from turbojets to turbofans, then to higher bypass turbofans, and recently, renewed interest in turbopropeller designs.

The advantage of mass augmentation tends to decrease for increasing vehicle ΔV. Figure 1 shows a comparison of the energy efficiency and specific impulse for a system using hydrogen with heat of combustion of 51,000 Btu/lb (118,600 kj/kg) as the energy source. The energy efficiency is the fraction of the heat of combustion utilized to increase the vehicle ΔV. Note that at relatively low ΔV, the energy efficiency of a rocket, even at very high specific impulse, is very small, showing a 4% efficiency for 5000 ft/s (1520 m/s) of ΔV at 3000 s (29,400 m/s) I_{sp}. Note also the opposite effect, that at high-stage ΔV, modest specific impulse indicates more efficient use of the heat of combustion, showing about a 10% efficiency for 30,000 ft/s (9140 m/s) of ΔV at 500 s (4900 m/s) I_{sp}.

The advanced energy-augmented propulsion concepts, such as metastable helium, solar heated plasmas, perforated solar sails, and antiproton annihilation are not expected to be available in the specified IOC time window. Those that could meet the IOC (fluoride propellants, tripropellants, or nuclear systems) have too little long-range performance potential or have severe safety problems, or both.[3]

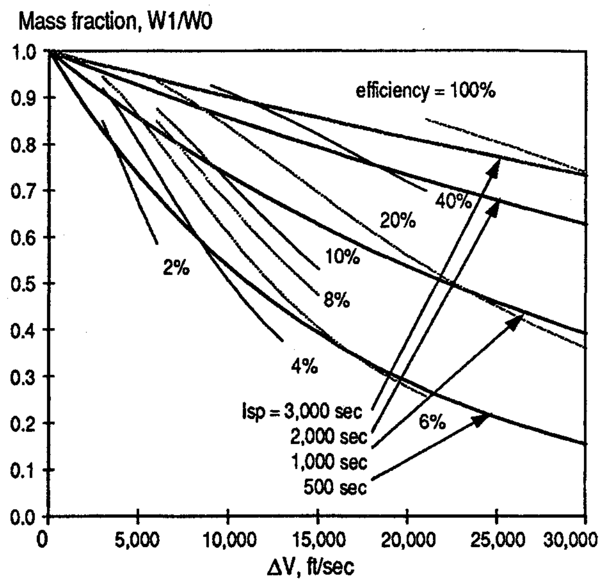

Fig. 1 Energy efficiency of a rocket is small at low ΔV. But at high ΔV, modest specific impulse indicates more efficient use of energy.

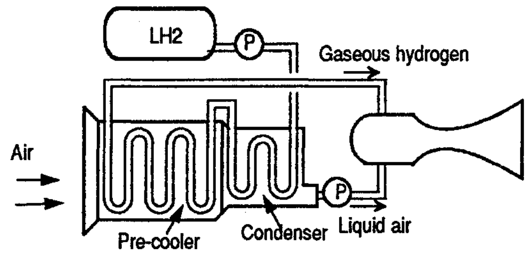

Fig. 2 Basic LACE uses cooling capacity of liquid hydrogen to liquefy incoming air.

Therefore, the study was concentrated on the mass-augmented propulsion system concepts. There are basically five mass augmentation schemes: 1) various ejector rockets and ejector ramjets; 2) turbomachinery pumped systems (turbojets, turbofans, turborockets, etc.); 3) liquid air cycle engines including cryojets; 4) ramjets/scramjets; and 5) air collection and enrichment.

The ejector ramjet and the air turborocket both utilize internal oxygen, and consequently, they are hybrid airbreathing-rocket systems. They have much lower specific impulse at low speed than a true airbreathing engine such as a turbojet. They may be attractive, however, because of their high thrust-to-weight ratio which favorably impacts payload. When the transition speed from airbreathing to rockets is raised, say from Mach 3 to 8, use of the high-specific impulse airbreathing engines over a larger speed range will favorably impact the orbit-to-takeoff vehicle weight ratio. This accounts for much of the interest in supersonic combustion ramjets (scramjet). However, scramjet performance is uncertain above about Mach 15, and the uncertainty increases with increasing speed. Rocket propulsion would then be required above this speed.

The liquid air cycle engine (LACE) in Fig. 2 was conceived in the late 1950s as an innovative means to replace the stored oxidizer for a rocket engine by using the large cooling capacity of liquid hydrogen to liquefy atmospheric air. Because the power required to pump liquid oxidizer is a small fraction of that required to compress ambient air, this airbreathing concept showed potential superiority over turbojets as well as rocket engines. Analysis indicated the heat exchanger size and weight were sufficiently small to justify this promise, and engine concepts emerged with specific impulse twice that of rockets and less specific weight than turbojets.

Later, the concept of combining a ramjet with LACE emerged. At speeds suitable for a hydrogen-fueled ramjet, the liquid air produced in the LACE heat exchanger is stored in tanks for later use in a rocket engine. This idea promised to greatly increase the speed capability of a single-stage vehicle. A short time later, it was shown that separating and storing mostly the oxygen of the liquefied air might extend the speed all the way to orbit. This air collection and enrichment (ACE) concept is shown in Fig. 3.

The development of the ACE system as pursued vigorously in the early 1960s with work on LACE and ramjet engines, heat exchangers, separators, para-ortho hydrogen catalysis, and vehicle systems studies. These projects were eventually terminated because of decisions to emphasize development of rockets in the Apollo and Shuttle programs, and emerging interest in scramjets. Data resulting from these studies and experiments are a valuable asset today. An excellent review of the development of airborne air separators and heat exchangers is presented recently with extensive bibliograpy on these subjects.[4]

SSTO with ALES Propulsion

Of the five mass-augmented systems that were compared, only a system employing air collection and enrichment or a scramjet appeared to have the capability to attain orbit in a single stage. Also, a collection and enrichment system employing a turborocket accelerator possesses valuable symbiotic characteristics that enhance the performance of each. In 1986, a system employing a LACE augmented turborocket engine for takeoff and low-speed acceleration was combined with an air collection and enrichment system.[2] To distinguish the new air liquefaction and enrichment system, it was termed ALES.

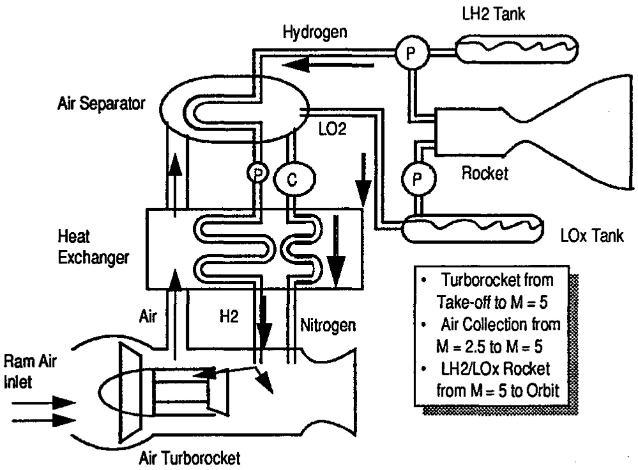

Fig. 3 ACE concept separates and stores mostly the oxygen of the liquefied air. The propulsion system can be used all the way to orbit by using the stored oxidizer in the rockets during trans-atmospheric flight.

The technical attractiveness of this concept is due to its very high airbreathing performance at low Mach numbers, and use of a uniquely fueled rocket at high speed where rocket propulsion is appropriate. The ALES subsystem provides oxidizer at high pressure for the turborocket, and the turborocket compressor delivers air to the heat exchanger at much higher pressure at takeoff and low-speed acceleration than a pure LACE. The ALES concept of air collection and enrichment was selected for further evaluation and system study.

For takeoff and initial acceleration, the system operates as shown in Fig. 4. The afterburning turborocket is the main airflow path in the system and generates all of the thrust. The power to drive the fan is generated by a LACE as follows:

1) Air is bled from the exit of the turborocket compressor and then cooled and condensed by the cold hydrogen fuel flowing to the engine.

2) The liquid air is pumped and ducted to the turborocket high-pressure combustion chamber. The cooling capacity of hydrogen limits about 5 kg of air condensation for each kg of hydrogen.

3) In the high-pressure combustion chamber, the liquefied air is burned with the hydrogen from the heat exchanger and these combustion products power the fan turbine. These combustion products are hydrogen-rich by a factor of 6.

4) This hydrogen-rich stream is then introduced into the turbofan afterburner, and burned with more air so that the exhaust from the nozzle is a stoichiometric mixture.

This arrangement of fan and turbine is very light compared with the usual turbine engine because all of the fan pressure ratio is available for thrust and none is required to drive the turbine as in a turbojet. Furthermore, because the air to be pumped is liquid, only a fraction of the amount of power is required by this pump compared with the compressor power required if it were a gas.

From about Mach 2.5 to 5, the system operates in the air collection and enrichment mode as indicated in Fig. 5. In this

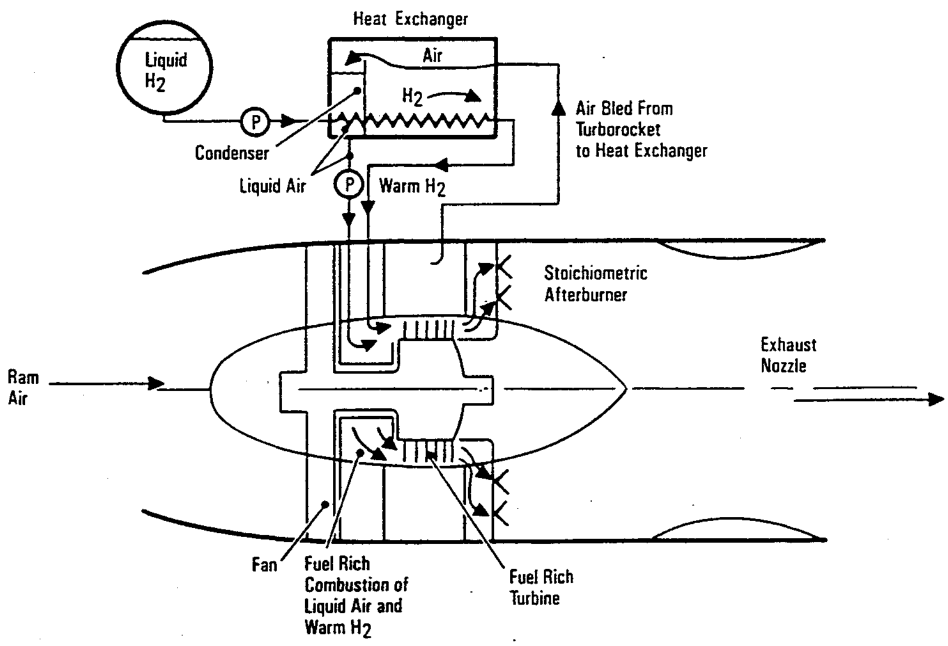

Fig. 4 LACE augmented turborocket engine for takeoff and low-speed acceleration was combined with an air collection and enrichment system.

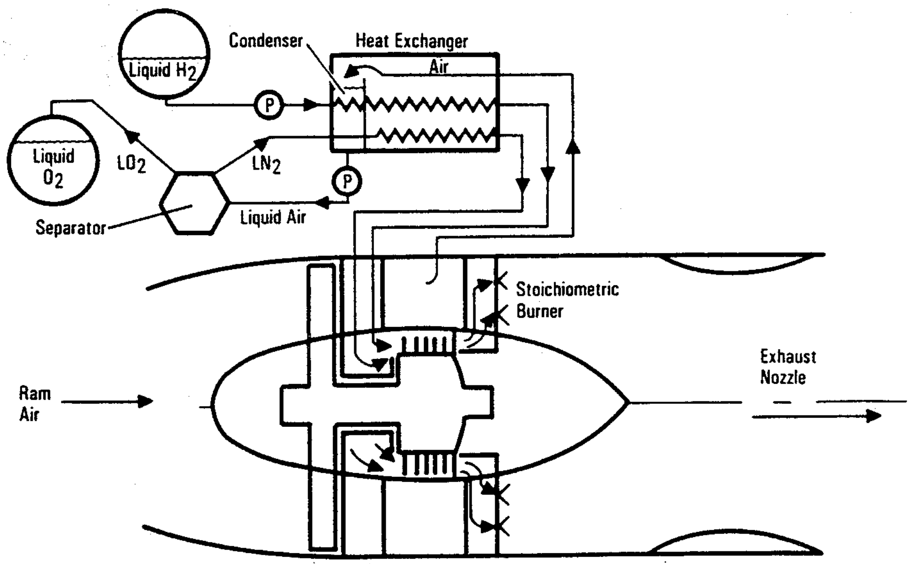

Fig. 5 Liquid oxygen product from the separator (about 90% LO_2) is stored in tanks for later use in a rocket engine and the nitrogen (about 98% nitrogen) is used as a coolant and expanded through the turborocket turbine to generate power.

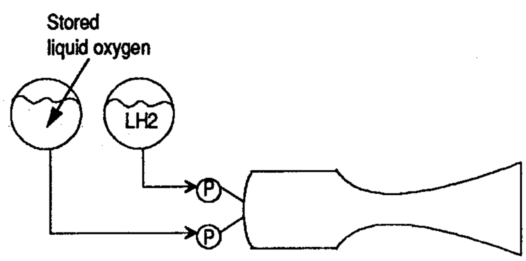

Fig. 6 High-speed rocket operation uses stored oxidizer from Mach 5 to orbit.

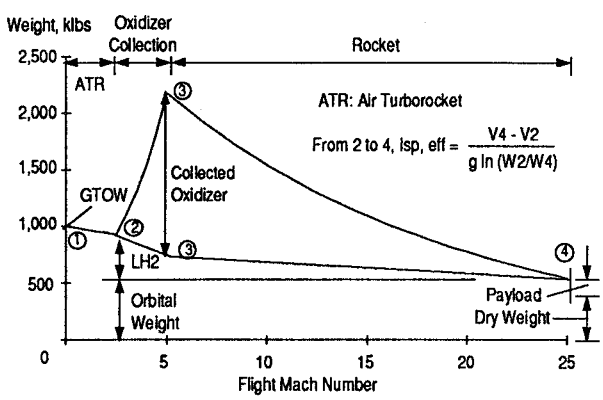

Fig. 7 Vehicle weight increases from Mach 2.5 to 5 to about twice the takeoff gross weight. When sufficient oxidizer has been collected, rocket engines are used to accelerate from about Mach 5 to orbit.

speed regime, the turborocket operates more like a ramjet (i.e., the fan pressure ratio is small) so that the fan turbine power can be reduced. The liquid air or saturated air vapor was routed to a rotating distillation-type oxygen separator. The liquid oxygen product from this separator (about 90% LO_2) is stored in tanks for later use in a rocket engine, and the waste nitrogen saturated vapor (about 98% nitrogen) is used as a coolant in the heat exchanger.

The warm nitrogen and hydrogen emerging from the heat exchanger are expanded through the turborocket turbine to provide power to the fan. The hydrogen-rich fan turbine exhaust is then burned in the turborocket airflow in the afterburner at an appropriate fuel-to-air ratio. It is expected that at Mach 5, this mixture ratio will be stoichiometric with respect to the hydrogen and oxygen present.

With sufficient oxygen collected and stored at Mach 5, the rocket acceleration is initiated. In this mode, the stored liquid oxygen is burned with the remaining liquid hydrogen in high-pressure, high-impulse rocket engines as depicted in Fig. 6.

Figure 7 shows a weight history for single-stage-to-orbit operation. Fuel is consumed along the line 1-2-3′-4. Oxygen is collected at a rate five times that of hydrogen consumption along line 2-3 so that the vehicle weight increases from Mach 2.5 to 5 to about twice the takeoff gross weight. When sufficient oxidizer has been collected, rocket engines are used to accelerate from about Mach 5 to orbit.

This propulsion system was further defined and the SSTO system performance evaluated.[1,2] The primary result was that a 10^6-lb (450,000-kg) GTOW vehicle has the capability to deliver 451,400 lb (205,000 kg) to a 100-n.mi. (180-km) polar orbit. However, the inert weight, including auxiliary propulsion propellant, residuals, and reserves are estimated to be 515,250 lb (234,000 kg), so that the payload capability is minus 63,800 lb (29,000 kg).[1] In reviewing that study, several modifications have surfaced that, when incorporated, yield attractive payload performance improvements.

Effect of Aerodynamic Improvements

One of the unique characteristics of a vehicle with an ALES propulsion system is that the vehicle weight increases as it accelerates during the oxidizer collection period. This unique characteristic demands different vehicle design criteria than conventional Earth-to-orbit vehicles. Since the ALES SSTO vehicle weight is the maximum at the end of oxidizer collection, not at the takeoff as conventional SSTO vehicles, the vehicle aerodynamic performance at the end of collection is one of the most critical design parameters. A circular cross-sectional vehicle concept may be structurally efficient, but the hypersonic aerodynamic performance is poor.[1] Alternative vehicle concepts such as blended body-wing or waverider configuration were not considered in the study because design technology of such vehicle concepts were not mature enough to consider for a 1990 IOC.

Numerous studies, however, indicated that by using advanced vehicle configurations the hypersonic lift-to-drag ratio (L/D) can be significantly improved over the conventional circular cross-sectional configurations. The design and engineering technology of those advanced configurations appears significantly mature to have an IOC of 2005–2010. Weights of the ALES SSTO using the advanced vehicle concepts were evaluated using 1990 materials to deliver a 10,000-lb (4,500-

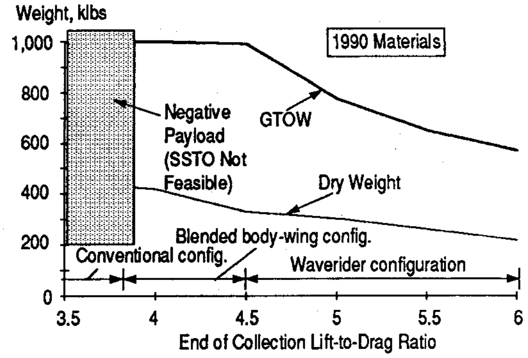

Fig. 8 Improved end-of-collection L/D reduces GTOW and dry weights of ALES SSTO.

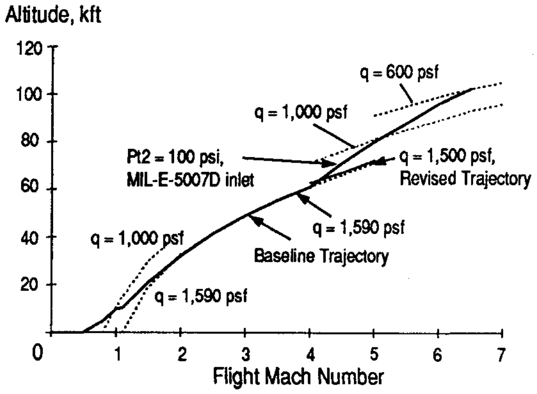

Fig. 9 Ascent flight trajectory was modified to improve payload performance.

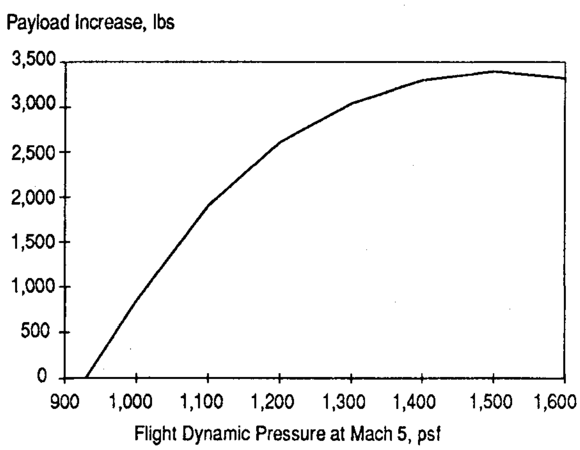

Fig. 10 When the Mach 5 dynamic pressure is increased to 1500 psf (0.72 bars), the payload is increased by 3400 lb (1550 kg).

kg) payload to a 100-n.mi. (185-km) polar orbit, and are shown in Fig. 8.

A conventional circular cross-sectional vehicle concept resulted in a maximum hypersonic L/D of about 3.7[1] and resulted in a "negative" payload as shown in Fig. 8. Improved L/D reduced the weights significantly by reducing the airbreathing engine size, which also reduced structure weight. Using a blended body-wing configuration, GTOW of the SSTO can be reduced below 10^6 lb (455,000 kg), which is occasionally referred to as the weight limit for a horizontal takeoff vehicle. Further improvement of the hypersonic L/D reduced GTOW drastically. Using a waverider configuration, GTOW was reduced to 600,000–800,000 lbs (270,000–360,000 kg), which is approximately the weight of a Boeing 747. All of this can be achieved using currently available materials.

Trajectory Effects on Component Weights

The flight trajectory for the previous studies[1,2] was a speed-altitude relationship that maintains a constant flight dynamic pressure of 1590 psf (0.76 bars) up to Mach 4 as shown in Fig. 9. The inlet duct pressure recovery used was that specified in MIL-E-5007D.

The inlet diffuser exit pressure steadily increases along this trajectory to 100 psia (6.9 bars) at Mach 4. To avoid excessive engine internal pressure (and engine weight) the constant dynamic pressure ratio trajectory at Mach 4 was changed to 1 with a constant diffuser exit pressure of 100 psia (6.9 bars) as the speed increases. This reduces the flight dynamic pressure to 930 psf (0.44 bars) at Mach 5. The wing area was increased 45% to accommodate this reduction in flight dynamic pressure to achieve a high L/D at Mach 5 with maximum gross weight.

A better trajectory has since been found that decreases the inert weight. The flight dynamic pressure at Mach 5 was increased by several steps to 1600 psf (0.76 bars). In order to maintain the maximum diffuser exit pressure at 100 psia (6.9 bars), it was assumed that the inlet pressure recovery could be reduced not to exceed this design maximum. The new higher fuel flow at each step was determined for the higher specific fuel consumption and lower L/D.

The collection system weight increased in proportion to the fuel flow to maintain the same collection ratio. As the dynamic pressure increased, the wing area and weight decreased, as did the L/D. The difference between wing weight reduction and collection weight increase is shown in Fig. 10. The difference shows a maximum increase in payload of 3400 lb (1550 kg) when the Mach 5 dynamic pressure is 1500 psf (0.72 bars). The revised trajectory is shown in Fig. 9.

Advanced Materials for Structures

A key to the feasibility of a practical single-stage-to-orbit vehicle is the use of advanced materials in a highly efficient, structurally reliable design. The structure of Ref. 1 was based on weight estimating relationships (WER) developed for the Shuttle that used aluminum as the primary structural material. Table 1 shows the material properties of aluminum (Al), aluminum-lithium (Al-Li), and graphite polyimide (Gr-Pi). The weights for the SSTO vehicle were then reduced by an appropriate factor from the aluminum WER to account for the change in material capability and the estimated state-of-the-art.

The capability ratio (CR) is defined as

$$\mathrm{CR} = \frac{\text{Wt. of mat'l}}{\text{Wt. of Al}} = \frac{\rho,\ \text{mat'l}}{\rho,\ \text{Al}} \times \frac{\text{Uts, Al}}{\text{Uts, mat'l}} \tag{3}$$

The capability ratio is shown on the last column of Table 1. The ratio of the material capability of Gr-Pi to that of Al-Li is 0.22/0.72 or 0.31, indicating an ideal weight savings over Al-Li of 69% when using Gr-Pi (when stiffness is not a factor). However, in the performance estimates of Ref. 1, the IOC was assumed to require 1990 state-of-the-art, so that Gr-Pi weights were only 2.5% less than those estimated for Al-Li. It was estimated that the 1990 state-of-the-art in designing and fabricating parts from Gr-Pi was so immature that a lower weight estimate for this material was not prudent.

The 1990 technology state-of-the-art also dictated that the hydrogen tanks be surrounded by body structure. However, others have indicated that "the volumetric and thermal characteristics of liquid hydrogen necessitates it being stored in the fuselage and the demand for structural efficiency requires that the cryogenic tanks become an integral part of the primary structures."[5] The external hydrogen tank of the Shuttle is a developed example of this philosophy, although it is an expendable, single mission design that is not required to withstand the environment of re-entry. Furthermore, the thermostructural study and development in related airbreathing

Table 1 Material properties comparison

	ρ, lb/in.^{3}, g/cm^3	U_{ts},[a] 10^3 psi, 10^3 bars	Yb,[b] 10^6 psi, 10^6 bars	Max. temp., °F, °C	CR,[c] base = Al
Aluminum	0.1 (2.7)	60 (4.1)	10.5 (0.72)	350 (177)	1
Aluminum lithium	0.093 (2.5)	77 (5.3)	11.1 (0.76)	350 (177)	0.72
Graphite polyimide	0.056 (1.5)	151 (10.4)	20.6 (1.4)	600 (315)	0.22

[a]Ultimate tensile stress. [b]Young's modulus. [c]Capability ratio.

Table 2 ALES SSTO weight summary, 10^6-lb (455,000-kg) GTOW weights in pounds (in kilograms)

System	Ref. 1 Al/Li		Revised with Gr/Pi	
Structure		186,063 (84,570)		110,955 (50,400)
Body	51,689		15,379	
Tail	7,320		4,407	
Wing	61,794		36,340	
Crew module	595		358	
LH_2 tanks	53,500		45,475	
LO_2 tanks	11,760		9,996	
TPS		68,917 (31,300)		50,813 (23,100)
Body	28,074		19,656	
Wing	29,680		20,773	
Tail	2,599		1,820	
Internal	8,564		8,564	
Landing gear		37,100 (16,900)		31,500 (14,300)
Main propulsion		177,068 (80,500)		177,068 (80,500)
A/B engine (including inlet and nozzle)	62,550		62,550	
ALES (including HX and separator)	63,636		63,636	
Rocket (including TVC, feed, etc.)	50,882		50,882	
Auxiliary propulsion		6,572		6,572
Electric power generation and distribution		3,855		3,855
Surface control		3,725		3,725
Avionics ECLSS		3,210		3,210
Personnel and provisions		725		725
Dry weight subtotal		487,235 (221,000)		388,423 (177,000)
Propellant and reserves		576,577 (262,000)		576,577 (262,000)
LH_2 main propellant	548,562		548,562	
LH_2 cooling fluid	445		445	
MPS residual and reserve	9,355		9,355	
APS propellant	18,215		18,215	
Payload		−63,812 (−29,000)		35,000 (16,000)
GTOW		1,000,000 (455,000)		1,000,000 (455,000)

booster program are believed to result in lighter integral cryogenic tankage that can be designed and developed to meet 2005–2010 IOC of this study.

Table 2 summarizes the results of changing material and the IOC on the major component weights and the estimated payload. The structure weights reflect only the change in density of graphite-polyimide compared with aluminum-lithium (0.602), and no advantage was taken of the higher graphite-polyimide strength. However, the assumed body structure surface area was reduced by 50% to account for the effect of integrating the body structure and the hydrogen tank. Because graphite-polyimide has a higher working temperature capability, the thermal protection system can be lighter. The thermal protection system weight shown here is the same as used in the Ref. 1 study for Gr-Pi. The wing weight was reduced 5% before changing material to account for a more favorable trajectory. The landing gear weight was reduced by 15% in the belief that advancements in materials and technology jus-

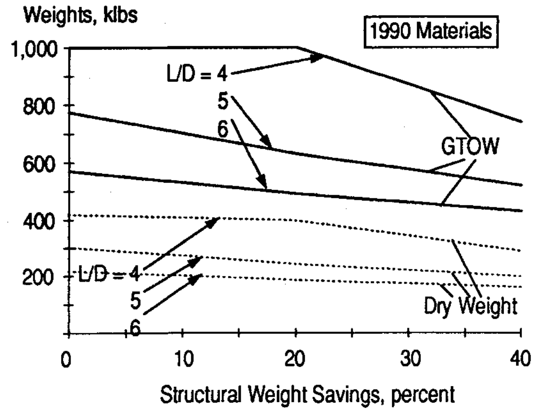

Fig. 11 By using the heat exchangers as a part of load bearing structure, the SSTO dry weight and GTOW can be reduced significantly.

tify this estimate. All other estimates (propulsion, electrical, propellants and reserves, etc.) are the same as for the 1990 estimate.[1]

Innovative Structural Concepts

As an alternative to the integral tanks, it is suggested to use the structure of the air liquefaction system as a part of load bearing structure. Observing the ALES SSTO weight breakdown shown in Ref. 1, the vehicle wing and body structure weight and the ALES weight are about 23 and 13% of the vehicle dry weight, respectively. Since the majority of the ALES weight results from the heat exchangers, by integrating the ALES heat exchangers with the wing-body structure a significant weight saving may be realized.

In Fig. 11, effect of the structure weight savings by the heat exchanger-structure integration is illustrated for the ALES SSTO with 10,000-lb (4500-kg) payload to 100-n.mi. (185-km) polar orbit. If about 20% of the structure weight can be saved by the structure integration, the ALES SSTO could be feasible with the conventional circular cross-sectional configuration (L/D < 4) using currently available materials. The structure integration may be applied for the advanced configurations also to reduce the vehicle weights, but the effects are minor, showing relatively flat curves for L/D of 5 and 6.

Summary and Concluding Remarks

A feasibility study of using ALES propulsion for Earth-to-orbit systems was completed. The study results indicate that high overall fuel specific impulse and engine thrust are possible using the air liquefaction and enrichment system. Since the GTOW of vehicles are greatly reduced by collecting oxidizer in-flight, HTOL fully reusable launch systems are feasible. Horizontal takeoff and landing capability has many advantages, including launch flexibility, simplified ground operations, and lower life cycle costs.

The ALES TSTO system requires only moderate technology development with current materials and design techniques. The TSTO system GTOW is relatively insensitive to the carrier weight and performance, but is extremely sensitive to the orbiter weight. Since technically immature components (air separator, heat exchanger, air turborocket) are located in carrier, technical advancement criticality is lessened.

An ALES SSTO is considerably more challenging. Since the payload fraction (payload weight divided by GTOW) of a typical SSTO is small, the inevitable vehicle weight growth during development and testing phases may wipe out the entire payload. The approach discussed in this article reduces the risks by showing several paths to success, rather than being dependent on achieving technical goals in all areas. As shown above, the ALES SSTO is feasible by either improving hypersonic aerodynamic performance using advanced design concepts, or using advanced materials (probably derived from the NASP), or using an innovative structure concept to reduce the dry weight. In addition to the promise of attractive payload capability for a horizontal-takeoff single-stage-to-orbit vehicle, the following favorable characteristics are attributes of a vehicle with this propulsion system: 1) high overall specific impulse and thrust can deliver over 45% of takeoff gross weight into orbit; 2) quick turnaround because horizontal takeoff and landing does not require either erection or a launch tower; 3) experiences a comparatively modest flight thermal environment by using rockets and by climbing out of the atmosphere above Mach 5 or 6; 4) development ground facilities only need to provide airbreathing propulsion simulation to Mach 5 or 6; 5) avoids high-speed airbreathing thrust decay and the related large inlet size requirement by using rockets and avoids significant aerodynamic drag above Mach 5 or 6; 6) uses a turbofan to pressurize the LACE heat exchanger and condenser subsystem, facilitating antifouling of the heat exchanger, higher condenser pressure, and higher air-to-fuel ratio; and 7) in the collection mode, the fan increases the collection air pressure more at the lower collection speeds then at high collection speeds. This maintains a more constant pressure into the separator, (which is not tolerant of large changes in inlet pressure).

Given that it is imperative to significantly reduce Earth-to-orbit transportation cost in order for future space operations to flourish, this article attempts to rationally consider the alternative technologies available to accomplish this. The conclusion is that the ALES propulsion system, coupled with materials, structures, manufacturing technology, and aerodynamics reasonably expected to be available for a 2005–2010 IOC, is a very promising approach. Further continuous study, test and evaluation of alternatives, and appropriately paced development of the technologies just mentioned should result in excellent prospects for a practical single-stage-to-orbit vehicle and a robust low-cost-to-orbit transportation system.

Acknowledgments

This study was partially sponsored by NASA Lewis Research Center and Air Force Aero Propulsion and Power Directorate under NAS 3-25559. The authors acknowledge the helpful comments made by C. L. Gould and the reviewers.

References

[1]Yi, A. C., and Hill, C. S., "Single-Stage-to-Orbit Vehicle Conceptual Design Using an Air Liquefaction and Enrichment Propulsion System," NASA CR 187157, June 1991.

[2]Yi, A. C., "Reusable SSTO/TSTO Launch Systems Using Air Liquefaction and Enrichment System (ALES) Propulsion," NASA CR 191156, June 1993.

[3]Bond, W. H., and Yi, A. C., "Prospects of Utilization of Air Liquefaction and Enrichment System (ALES) Propulsion in Fully Reusable Launch Vehicles," AIAA Paper 93-2025, June 1993.

[4]Leingang, J. L., Maurice, L. Q., and Carreiro, L. R., "Space Launch Systems Using Oxidizer Collection and Storage," 43rd Congress of the International Astronautical Federation, IAF 92-0664, Aug.–Sept. 1992.

[5]Kelly, H. N., Rummler, D. R., and Jackson, L. R., "Research in Structure and Material for Future Space Transportation Systems—An Overview," *Journal of Spacecraft and Rockets*, Vol. 20, No. 1, 1983, pp. 89–96.