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Preliminary Analysis of an Airbreathing and Rocket Combined-Cycle Engine

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In this paper the characteristics of the rocket engines are compared with those of the airbreathing combined-cycle engines. The types of airbreathing engines are classified based on their flight Mach numbers. The optimization of an airbreathing combined-cycle engine is presented. Based on the analyses of individual characteristics of hydrogen fuel, a rocket, and an airbreathing engine, a new combined engine cycle, known as the high-pressure hydrogen-expansion liquefied oxygen cycle engine (LOCE), is presented. It is one type of rocket-based airbreathing combined-cycle engines whose specific impulse can reach up to 3000 s, accounting for inlet air momentum loss, and 5300 s, without accounting for inlet air momentum loss. It successfully overcomes the inconsistency of the chamber pressure of a rocket engine and that of an airbreathing engine. Its liquefaction efficiency is 5–7 times higher than that of the typical liquefied air cycle engine cycle. It has a high specific impulse and a high thrust-to-weight ratio. The payload-to-takeoff weight ratio of a two stage to orbit reusable launch vehicle with LOCE is 3.13%, based on current analyses. This cycle, integrated with the advantages of rocket and airbreathing engines, is one of the most promising means of achieving high performance at the low-speed phase (Ma = 0–5) of flight.

Nomenclature

A	=	flow cross-sectional area
C_x	=	velocity loss coefficient from gravity and air dynamic drags (0.70)
c_p	=	specific heat
$\bar{c}_{p1air}$	=	mean specific heat of airflow
$\bar{c}_{p1n}$	=	mean specific heat of nitrogen in heat exchanger 1 (HX1)
M	=	mass (weight) of orbit vehicle, fuel, etc.
m_{air}	=	inlet air mass flux, 300 kg/s
$\dot{m}_{airin}v_{airin}$	=	inlet drag of engine, may be included in C_x
$\dot{m}_H$	=	hydrogen mass flow rate, 10.5 kg/s
$\dot{m}_n$	=	unliquefied airflow from the separator, about $0.8\dot{m}_{air}$, 240 kg/s
$\dot{m}v_e$	=	momentum of the engine
N_c	=	power of the air compressor, 31,260 kW
N_{tH}	=	hydrogen turbine output power, 31,273–31,281.6 kW
T_{Hin}	=	inlet temperature of hydrogen, 901–1030 K
T_{tair}	=	inlet temperature of the air turbine, 114 K
$T_{1airin}/T_{1airout}$	=	inlet/outlet temperature of airflow in HX1, 390.44/226.5
T_{1nin}/T_{1nout}	=	inlet/outlet temperature of nitrogen in HX1, 149.3/350
v_e, v	=	velocities of engine, velocity
γ	=	specific heat ratio
ΔP	=	pressure loss
ζ	=	coefficient of pressure loss
η	=	efficiency
η_c	=	efficiency of the air compressor, 0.83
η_{tair}	=	efficiency of the air turbine, 0.75
η_{tH}	=	efficiency of the hydrogen turbine, 0.87
λ	=	liquefaction ratio (liquefied oxygen/hydrogen needed)
π_c	=	pressure ratio of the air compressor, 12.3
π_{tair}	=	pressure ratio of the air turbine, 5.0
π_{tH}	=	pressure ratio of the hydrogen turbine, 2.5
ρ	=	density
τ	=	flight time

Subscripts

c	=	compressor
H	=	hydrogen
Hp	=	hydrogen pump
HX	=	heat exchangers
in	=	inlet parameters
n, N2	=	nitrogen
o	=	oxygen
op	=	oxygen pump
out	=	outlet parameters
s	=	condition at current flight time
$t$air	=	low temperature air turbine
tH	=	hydrogen turbine
0	=	condition at takeoff

Introduction

THE liquefied air cycle engine (LACE) concept is an attractive and challenging configuration for application to reusable space-transportation systems. Early in the 1950s, the U.S. (Marquardt) engaged in studies of various LACE concepts. Figure 1 shows the basic concept of LACE. The inlet air is cooled by liquid hydrogen (LH_2) and becomes liquefied air. The liquefied air, mixed with H_2, burns in the combustor, then provides propulsion power by flowing through the nozzle. It has a higher specific impulse than that of a pure liquid-propellant rocket engine.[1,2] However, it was impossible to achieve high-efficiency propulsion because of theoretical and

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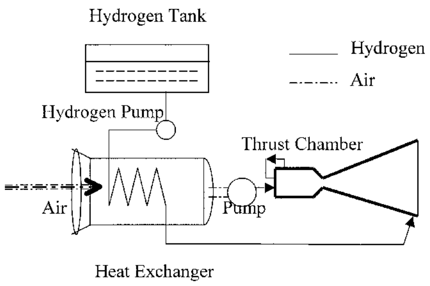

Fig. 1 Concept of an ideal LACE.

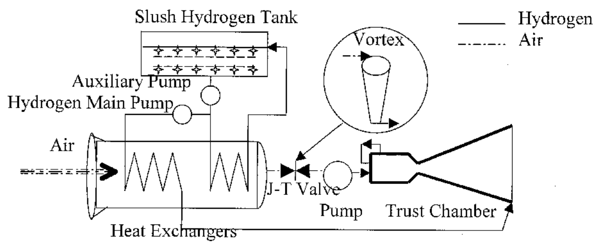

Fig. 2 LACE with feedback subsystem for slush hydrogen.

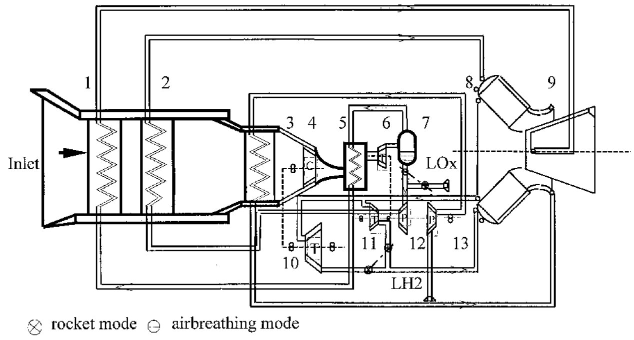

Fig. 3 High-pressure hydrogen expansion LOCE. 1, Nitrogen/air heat exchanger; 2, oxygen/air heat exchanger; 3, hydrogen/air heat exchanger; 4, centrifugal compressor; 5, low-temperature nitrogen/air heat exchanger; 6, expander; 7, gas liquid separator; 8, rocket thrust chamber; 9, aerospike nozzle[7]; 10, high-temperature hydrogen turbine; 11, hydrogen turbine for rocket mode; 12, liquid oxygen pump; 13, liquid hydrogen pump.

technical problems on portions of the cycle shown in Fig. 1. In the 1960s and in the last decade, many research groups have become interested in various kinds of LACE and its combination with other engine cycles such as scramjets, ramjets, and rockets[1–5,12,13] Many studies to improve the performance of LACE have been carried out.

Although the low-altitude flight regime is a rather small portion (20–30%) compared with the total flight mission of a spacecraft, it requires the expenditure of about 60% of the total propellant weight. If the oxidizer (85% of the propellant mass) can be oxygen from the air instead of being launched from the ground in tanks, this could reduce the takeoff gross weight dramatically, not only because of the oxygen saved but also because of the lighter tank and airframe. This also means that at the same takeoff gross weight the LACE can carry more payloads than that of a pure liquid rocket.

Using LH_2 to liquefy inlet air is a typical LACE technology (Fig. 1). Many investigators from the U.S. and Japan have discussed this technology.[1,2,4,15,16] The liquefaction ratio λ is 3.0–5.0, much lower than its stoichiometric ratio (about 34). The mixture ratio of oxygen/hydrogen in a LACE combustor is about 1:1, compared with 6:1 for current liquid-propellant rocket thrust chambers such as the Space Shuttle Main Engine (SSME). As a result, the specific impulse of such a LACE is not significantly higher than that of current LH_2/(liquid oxygen) LOX rocket engines. Furthermore, it has a low thrust-to-weight ratio.

One way to improve the performance of LACE (Fig. 2) is to use slush hydrogen (SH_2).[2] A two-phase pump is used to raise the pressure of SH_2 into a heat exchanger and (after the heat exchanger) feed part of the hydrogen back to the tank. Because of the extra cooling capacity of SH_2, it can increase the liquefaction ratio 12–15% and the specific impulse about 10%. The tank return feeding system is more complicated and more difficult to control compared with that of heat exchangers, and its feasibility and economy are reduced.

To investigate the feasibility of LACE, Miki et al.[10] and Borisov et al.[1] are among the groups who have carried out some subsystem experiments on the LACE concept.[8] The specific impulse of LACE engine is from 600–1800 s. The key technology, which involves experiments on heat exchangers, has been studied in many ways. In the fall of 1996, Qi and Dai[11] successfully completed a hot fire test of a high-temperature heat exchanger that could be used to generate high-temperature hydrogen to drive the turbine, e.g., component 10 in Fig. 3, which is used to drive the compressor, e.g., component 4 in Fig. 3, for the inlet air to be liquefied. The maximum temperature is 1500 K and a very effective heat transfer coefficient has been obtained. Tanatsugu et al.[9] and Murray et al.[6] have also performed research on precoolers that could be used in LACE.

Another way to improve performance of LACE is to use the vortex-tube concept instead of a J–T valve[17] (Fig. 2), to increase the liquefaction ratio and reduce the engine thrust-to-weight ratio. It has been shown that vortex tubes are more efficient than J–T valves in processing the liquefied air.[17]

On the other hand, it has been shown that using low-temperature, high-density air without liquefaction as the oxidizer is less feasible to develop because of its need for large compressor work.[13] In this technology area, for example, Hotol was improved in the early 1990s to combine with the ramjet engine (Skylon).[18]

The main obstacle of all the preceding concepts is that with nitrogen as four-fifths of the air, oxygen as only one-fifth of the air takes part in the combustion process as the oxidizer. In this paper, we will present a solution for improving the performance of the liquefaction ratio from 1:1 to 5–7:1. Then the system would meet the final requirements of current rocket thrust chamber designs.

How to Improve the Performance of LACE

From the preceding discussion, it can be seen that the LACE concept is still underdeveloped and a way to greatly improve its performance represents a challenge. In this paper, various methods of improving LACE performance are proposed. First, we review some requirements of space transportation systems and their propulsion systems.

Requirements of Advanced Space Transportation Systems

Advanced Earth-to-orbit transportation systems may be divided into two categories: rocket-propulsion and airbreathing-propulsion systems. We can be sure that future advanced Earth-to-orbit transportation systems must incorporate both aeronautical and astronautical developments.

An ideal space transportation system requires 1) high thrust-to-weight ratio for all flight phases and 2) high specific impulse at high altitudes. Airbreathing and rocket systems have their own characteristics when considered as part of a space transportation system. Table 1 summarizes various single-stage-to-orbit (SSTO) technology programs. It indicates that if an LH_2/LOX rocket engine is used (column 3), the margin is only 1%, whereas a margin of 2.4% can be obtained if a tripropellant rocket engine is used (column 4).[3] On the other hand, if a rocket-based combined-cycle engine is adopted, the margin could be higher than 10%.[14]

Airbreathing technology may be divided into five categories according to the flight Mach number (Table 2). The fifth cat-

Table 1 SSTO technology development programs

	1960–1970	1970–1980	End of 1990s H/O	End of 1990s Tripropellants	2000
Requirement of propellant mass fraction	0.92	0.872	0.872	0.883	0.655
Feasible propellant mass fraction	0.838	0.848	0.882	0.907	0.76
Remarks	1) I_{sp} = 375 s 2) Aluminum structure	1) I_{sp} = 437 s 2) Aluminum and a few composite materials	1) I_{sp} = 437 s 2) Al/Li, C/C structure	1) I_{sp} = 437 s 2) Al/Li, C/C structure	1) I_{sp} > 1000 s 2) Combined engine (RBCC)

Table 2 Status and development of orbiting airbreathing engine technology

Mach number range	Technology level	Task requirement objectives or phase accomplishment	Engine technology
0–0.8/2.5	L-1011, feasible	——	Feasible
0–3.5/5	R&D	Prompted by hypersonic civil/military need	Turbofan variable cycle engine
0–6/8	R&D subsonic combustion and heat exchanger technology	——	LACE and LOCE
6/8–12/15∼25	Developing and exploring scramjet technology	Hypersonic cruise missile	Scramjet engine
>0–15/25 to orbit	Developing SSTO rocket (*X*-33/*X*-34)	——	Reusable

Table 3 Comparison of rocket and the airbreathing engine characteristics

Engine	Thrust to-weight ratio	Specific impulse	Pressure in thrust chamber	Takeoff weight	Oxidizer
Rocket	Highest	Low	High pressure	Relatively large	On-board
Air-breathing	Low	High	Difficult in high pressure	Relatively small	Using oxygen in the air
RBCC	High	High	Determined by specific cycles	Relatively small	Using oxygen in the air while carrying part oxygen by itself

egory is actually a rocket-powered single-stage-to-orbit system, e.g., *X*33/*X*34.

The technology difficulties in the middle three categories shown in Table 2 increase with the increasing flight Mach number. The second category may be realized in the next few years, owing to the needs of civil aviation in the world. If airbreathing engines become available, then its incorporation into a two-stage-to-orbit space transportation system, with the separation Mach number at 3–5, is the most possible solution. The third and fourth categories are still in research and development stages because of the technology difficulties involved.

Recently, the rocket-based combined-cycles (RBCC) engines have been discussed more frequently because of their attractive performance characteristics, such as increased payload-to-take-off-weight ratio and higher specific impulse. RBCC include a wide variety of engines that use the rocket thrust chamber (combustion chamber and nozzle), such as a scramjet combined with rocket, LACE, ATR, ATREX, and rocket-eject ramjet.[4,11,12,15] Currently the scramjet combined with rockets as ejectors becomes a major type of RBCC as a SSTO transportation system, in which a way to use scramjet is found. However, its low-altitude performance is not as favorable when compared with other airbreathing engine cycles.

Characteristics of Hydrogen

Hydrogen (in liquid form) is a powerful fuel that has the following advantages, which must be utilized:

1) High combustion heating value and high specific impulse.

2) Its power potential for driving the turbine is 3.5 times compared with that of hydrogen/air combusting gas, and 2.5 times compared to the hydrogen/oxygen combusting gas because of its low molecular weight. As a result it is an excellent power resource for turbine.

3) The cooling potential is high, so it is one of the best choices as a coolant for regenerative cooling and film cooling.

4) No toxicity and no pollution.

These four characteristics make liquid hydrogen one of the best working fluid choices in the astronautical field, and they also make special thermodynamics cycles based on these characteristics more attractive. The hydrogen expansion cycle[12] not only has high efficiency but also makes full use of the four characteristics just mentioned. It demonstrates excellent integration of the working fluid with a cycle and the best path of integrating rocket and airbreathing engines.

Characteristics of Engines

Engines must use available energy more effectively. The most effective way is to increase the pressure of the combustion process, which can improve the engine efficiency and the thrust-to-weight ratio.

The most important parameters of engine performance are the thrust-to-weight ratio and the specific impulse. It can be seen from Table 3 that these two parameters for rocket and airbreathing engines have opposite tendencies. The proper development direction is to use their synergetic characteristics. The combined-cycle engine, which combines the advantages of the rocket and the airbreather, is a more important direction for future development. However, the following difficulties should not be ignored.

1) Rockets have a high thrust-to-weight ratio but a relatively low specific impulse because they do not utilize oxygen in the atmosphere. Airbreathing engines have a high specific impulse; however, the thrust-to-weight-ratio is low because of their high-power air-compression equipment and their combustion at low pressure.

Table 4 Characteristics of four heat exchangers

	HX1 (Air/N_2)		HX2 (Air/O_2)		HX3 (Air/H_2)		HX4 (Air/N_2)	
	$M = 2$	$M = 5$	$M = 2$	$M = 5$	$M = 2$	$M = 5$	$M = 2$	$M = 5$
Air side								
Inlet temperature, K	390.4	1039.2	226.5	520.18	130	360	160	160
Outlet temperature, K	226.5	520.18	130	360	84	84	114	114
Inlet pressure, MPa	0.152	——	0.135	——	0.12	——	0.3	——
Outlet pressure, MPa	0.135	——	0.12	——	0.1	——	0.28	——
Mass flow, rate kg/s	300	——	300	——	300	——	300	——
Exchanging heat flux (10^6W)	49.385	170.34	28.51	49.322	13.552	83.362	13.59	13.59
Coolant side								
Inlet temperature, K	149.3	155	120	120	40	40	94	94
Outlet temperature, K	350	999.2	191.6	496.9	122	138.3	149.3	149.3
Inlet pressure, MPa	0.25	——	10.2	——	26.5	——	0.28	——
Outlet pressure, MPa	0.23	——	10.05	——	26.4	——	0.28	——
Mass flow rate, kg/s	240	——	60	——	12.6	——	240	——

2) One of the important criteria for assessing a combined cycle is how to deal with the nitrogen that accounts for 80% of the air.

One way to resolve these problems is to raise the combustor pressure of the airbreathing engine and to change from compressing air to pumping liquid. A liquefying cycle is thus desirable. The second difficulty makes most of the LACE impractical or less available for implementation. To solve the inadequacy of cooling, combustion, and turbine driving, a new RBCC is presented here.

Innovative Airbreathing Rocket Combined-Cycle Engine

Based on the preceding discussions and analyses of hydrogen characteristics, an innovative idea is presented to improve the performances of the LACE concept. Applying the heat-transfer technology (Rankine cycle), the high-pressure hydrogen-expansion liquefying oxygen cycle airbreathing/rocket combine-cycle engine (called LOCE), is proposed and schematically shown in Fig. 3. This engine is a dual-cycle-mode engine that can also function in the rocket mode. Here only the airbreathing mode is briefly described.

After passing through three heat exchangers, HX1, HX2, and HX3 (components 1, 2, and 3 in Fig. 3), the air temperature is cooled to 84 K. Compressed by a centrifugal compressor (component 4) to a pressure of about 1.5 MPa, the air passes through the fourth heat exchanger (HX4, component 5) to withdraw the compression heat. Passing through an expander (component 6), most of the oxygen is liquefied with a small fraction of nitrogen liquefied. After a separator (component 7), most nitrogen is reheated through HX4, and then through HX1 to a temperature lower than 40 K of the inlet air when the nitrogen leaves the first heat exchanger (see Table 4), then goes to cool the aerospike nozzle (component 9). In the end, nitrogen is ejected at 600 K and 0.15 MPa, which can offset most of the loss of the inlet momentum. If it is discharged at 800 K or at 0.75 MPa, it can offset all of the inlet momentum's loss.

With a pump (component 12), the pressure of the liquefied oxygen (the concentration is higher than 92%) is increased to 11.0 MPa and heated to about 200 K by the second heat exchanger, then goes to the rocket chamber. The pressure of the liquefied hydrogen is increased to 23 MPa by the pump (component 13), then it exchanges heat by flowing through the third heat exchanger, with its temperature raised to 125 K or more, then goes to cool the body of the rocket chamber and other cycle components. The heated hydrogen drives the turbine (component 10), which drives the compressor to compress low temperature air to 1.5 MPa. In the end, the hydrogen flows to the thrust chamber to burn with oxygen. The thrust chamber pressure is about 10 MPa. If the proper nozzle expansion ratio is selected, the engine specific impulse may reach up to 5300 s.

In the following, a primary analysis of LOCE is presented. In addition, an analysis of a two-stage-to-orbit (TSTO) vehicle powered by LOCE as the first stage is also carried out.

Analysis of LOCE

In the first heat transfer exchanger (HX1, Fig. 3), the energy equation (assuming no heat loss in HX1)

$$\dot{m}_{\mathrm{air}}\bar{c}_{p1\mathrm{air}}(T_{1\mathrm{airin}} - T_{1\mathrm{airout}}) = \dot{m}_{\mathrm{n}}\bar{c}_{p1\mathrm{n}}(T_{1\mathrm{nout}} - T_{1\mathrm{nin}})$$

The pressure loss across the heat exchanger is

$$\Delta P_{1\mathrm{air}} = \zeta_{1\mathrm{air}}(\dot{m}_{\mathrm{air}}/A_{1\mathrm{air}})^2/\rho_{\mathrm{air}}$$

$$\Delta P_{1\mathrm{n}} = \zeta_{1\mathrm{n}}(\dot{m}_{\mathrm{n}}/A_{1\mathrm{n}})^2/\rho_{\mathrm{n}}$$

Formulas in the second to fourth heat exchanger are similar. Table 4 is a summary of the four heat exchanger analyses for two typical Mach numbers of 2 and 5. The hydrogen mass flow is only 3–4% of the total air mass flow, and the outlet temperature of hydrogen in HX3 is about 122–138 K. For the low-temperature air compressor, the power can be expressed as

$$N_c = \dot{m}_{\mathrm{air}}c_pT[\pi_c^{(\gamma-1)/\gamma} - 1]/\eta_c$$

The turbine power and the low-temperature air turbine output power are

$$N_{t\mathrm{H}} = \dot{m}_H[\gamma/(\gamma - 1)]RT_{\mathrm{Hin}}(1 - 1/\pi_{t\mathrm{H}}^{\gamma-1/\gamma})\eta_{t\mathrm{H}}$$

$$N_{t\mathrm{air}} = 9500 \text{ kW}$$

The required hydrogen and oxygen pumps power are

$$N_{\mathrm{op}} = \dot{m}_{\mathrm{o}}\Delta P_{\mathrm{o}}/(\gamma_{\mathrm{o}}\cdot\eta_{\mathrm{op}}) = 716.4 \text{ kW}$$

$$N_{\mathrm{H}p} = \dot{m}_{\mathrm{H}}\Delta P_{\mathrm{H}}/(\gamma_{\mathrm{H}}\cdot\eta_{\mathrm{H}p}) = 5511.2\text{–}6298.4 \text{ kW}$$

From this we have $N_{t\mathrm{air}} > N_{\mathrm{H}p} + N_{\mathrm{op}}$.

The thrust of the engine is the total of the rocket aerospike nozzle thrust, F_{rocket}, and the nitrogen-ejecting thrust, F_{N2}, minus the inlet air momentum:

$$F_{\mathrm{rocket}} = I_{\mathrm{sp}}\cdot\dot{m}_{\mathrm{thrst}}, \qquad F_{\mathrm{N2}} = \dot{m}_{\mathrm{N2}}\cdot v_{\mathrm{exitN2}}, \qquad \dot{m}_{\mathrm{N2}} = 0.8\dot{m}_{\mathrm{air}}$$

$$v_{\mathrm{exit}} = \left\{\frac{2\gamma}{\gamma-1}RT\left[1 - \left(\frac{P_e}{P_c}\right)^{(\gamma-1)/\gamma}\right]\right\}^{1/2}$$

$$F_{\mathrm{airin}} = \dot{m}_{\mathrm{air}}\cdot v_{\mathrm{airin}}$$

Table 5 Thrust and specific impulse of LOCE at two flight conditions

Flight condition	F_{rocket}, kN	F_{N2}, kN	F_{airin}, kN	F_{total}, kN	F'_{total}, kN	I_{sp}, s	I'_{sp}, s
$Ma = 2.0$ $H = 13$ km	270.97	38.67	−176.76	308.37	485.13	2996.8	4711.4
$Ma = 5.0$ $H = 24.5$ km	287.25	265.2	−446.98	105.47	552.45	1024.98	5365.2

Table 6 Mass of heat exchangers and the engine

HX1, kg	HX2, kg	HX3, kg	HX4, kg	HX_{total}, kg	Engine, kg	Remarks
1520	2480	1116	1570	6686	7841	$a_{HX} = 3000$, $\rho = 7.79$
1048	1710	770	1082	4610	5765	$a_{HX} = 5000$, $\rho = 7.79$
605.4	987.8	444.8	625	2663	3818	$a_{HX} = 5000$, $\rho = 4.5$

The specific impulse of the engine is

$$I_{sp} = (F_{rocket} + F_{N2} - F_{airin})/\dot{m}_H$$

Here, P_e is the exist pressure and P_c is the pressure of the combustor cooling channel.

In the preceding equation, it is assumed that all inlet air is brought to stagnation; therefore having maximum momentum loss. However, not all of the momentum of inlet air is lost, so that the specific impulse is underestimated. If the loss in the air inlet momentum is reduced, the specific impulse may reach up to 4700–5300 s. In this case, the formula for specific impulse is

$$I'_{sp} = (F_{rocket} + F_{N2})/\dot{m}_H$$

From this analysis the mixture ratio in the combustion chamber is $r = 5$ and the chamber pressure is 10 MPa. It is assumed that the capture area of the inlet can be changed to fit the air mass flow. The results are summarized in Table 5.

Preliminary Analysis of a TSTO Reusable Launch Vehicle with LOCE

The TSTO is composed of a first-stage carrier airplane and an orbital vehicle. For the LOCE in the first stage, the weight of the engine and the ratio of payload-to-takeoff weight are estimated as

$$M_{LOCE} = \sum M_{HXs} + M_{rocket} + M_{aircompr} + M_{turbines}$$

$$M_{aircompr} = 350 \text{ kg}, \qquad M_{turbines} = 400 \text{ kg}$$

$$M_{rocket} = \frac{\bar{I}_{spRocket} \cdot \dot{m}_{rocket}}{(T/W)_{rocket}} = 405 \text{ kg}$$

The thrust-to-weight ratio $(T/W)_{rocket}$ of the rocket is 70. The weights of heat exchangers are calculated by

$$M_{HX} = \rho \cdot V, \qquad V = \frac{A}{a_{HX}}, \qquad \Delta T_{HX} = \frac{\Delta T_{max} - \Delta T_{min}}{\ell n(\Delta T_{max}/\Delta T_{min})}$$

$$\Delta T_{max} = \max[\text{abs}(T_{airin} - T_{N2out}), \text{abs}(T_{airout} - T_{N2in})]$$

$$\Delta T_{min} = \min[\text{abs}(T_{airin} - T_{N2out}), \text{abs}(T_{airout} - T_{N2in})]$$

$$Q_{HX} = \dot{m}_{air} \int_{T_2}^{T_1} c_p \, dT$$

The compact efficiency (a_{HX}, heat transfer area per volume) of the heat exchangers is assumed to be between 4000–5000 m^2/m^3 and the material density is 7.79. If new materials can be used, the density would drop to 4.5. The weight of the four heat exchangers and the engine are listed in Table 6.

Next we consider a simple TSTO vehicle model. The analysis focuses only on its feasibility. It is assumed that the primary (empty) weight of the first-stage carrier airplane M_{empty} in terms of engine weight is

$$M_{empty} \cong 3M_{LOCE}$$

From Newton's law

$$m_{carrier} \frac{dv}{d\tau} = F = C_x \dot{m} v_e$$

and momentum of the engine

$$\dot{m} v_e = 6\dot{m}_H v_{rocket} + \dot{m}_{N2} v_{N2} - \dot{m}_{airin} v_{airin}$$

thus[11]

$$\frac{dv}{d\tau} = \frac{C_x}{m_{carrier}} (6v_{rocket} + 22.86 v_{N2} - 28.57 v_{airin}) \frac{dm}{d\tau}$$

$$\int_0^{Ma=5} dv = a \int_{M_0}^{M_s} d\, \ell n\, m$$

Here, $a = C_x(6v_{rocket} + 22.86v_{N2} - 28.57v_{airin})$ or $a = C_x(6v_{rocket} + 22.86v_{N2})$, for the case without accounting for air momentum loss:

$$\Delta v = a\, \ell n(M_0/M_s), \qquad \mu = M_0/M_s$$

It is assumed that the orbiting vehicle weight is 112 tons, which fixes the rocket engines as a pair of 50-ton LH_2/LO_X thrust engines. The takeoff weight of total TSTO is

$$M_{takeoff} = M_{empty} + M_{fuel} + M_{orbit}$$

$$M_{fuel} = M_0 - M_s = M_0[1 - (1/\mu)]$$

For a 0.8 thrust-to-weight ratio of the first stage carrier airplane at takeoff

$$NF_0/0.8 = \mu N M_{empty} + 112{,}000$$

From this equation, the number of LOCE engines, N, required can be determined.

The orbit engines are two LH_2/LO_X expansion-cycle rockets with an aerospike nozzle. Its specific impulse is 455 s. The velocity increment to orbit is[3]

$$\Delta v_{orbit} = v_{final} - v_{carrier} + \Delta v_{loss} = 6910 \text{ m/s}$$

$$\mu_{orbit} = M_{0orbit}/M_{sorbit} = \exp[\Delta v_{orbit}/(I_{sp} \cdot g)] = 4.71$$

$$M_{propellant} = M_{orbit}[1 - (1/\mu_{orbit})]$$

$$M_{orbit} = M_{struct} + M_{propellant} + M_{payload}$$

$$M_{struct} = 3.5M_{engine} + (13{,}000 \sim 15{,}000)$$

Table 7 Main characteristics of TSTO

Technology level	Number of engines	Takeoff weight, ton	Empty weight of carrier, ton	Fuel of carrier, ton	Orbit total weight, ton	Propellants of orbiter, ton	Orbiter empty weight, ton	Payload, ton	Payload-to-takeoff weight ratio, %
~2000	11	432.66	258.75	56.43	117.48	92.54	18–20	4.94–6.94	1.14–1.6
~2005–2010	7	275.33	121.07	35.91	118.35	93.22	18–20	5.13–7.13	1.86–2.59
~2010–2020	5	196.67	57.27	25.65	113.75	89.6	18–20	4.15–6.15	2.11–3.13

Table 7 lists the TSTO characteristics at different technology levels. Here the calculation of performance characteristics of the orbiter is based on 1990's technology. The payload-to-takeoff ratio is from 1.14 to 3.13%. This feature of TSTO is reasonable and feasible. The key technology of LOCE then is the development of compact heat exchangers.

Characteristics of LOCE and Key Technologies

The greatest difference between LOCE and a LACE is how to deal with the nitrogen that constitutes four-fifths of the air. The efficiency at which LOCE liquefies oxygen is about 5–7 times greater than that of LACE. Thus, the mixture ratio of H_2/O_2 in a LOCE's thrust chamber can reach the same level as that of current rocket engines, and many proven rocket engine technologies can be applied to LOCE. The cycle arrangement is similar to that of LACE, but it makes full use of the four advantages of the hydrogen. Because its chamber pressure is much higher than that of LACE and other airbreathing-combined engines, its volume and weight will be substantially reduced. It is predicted that its thrust-to-weight ratio may reach 30–40. Because of its compatibility with rocket engines, LOCE not only can use many proven rocket technologies, but also can be shifted to the rocket mode quite simply and conveniently. Its flight regime is enlarged and the engine adaptation is considerably enhanced.

LOCE has the following characteristics that are different from those of the LACE.

1) LOCE's thrust chamber is totally compatible with that of the rocket engine, its propellant mixture ratio is 6, and the combustion pressure can reach up to 10 MPa. Transition from the airbreathing mode to the rocket mode is simple and convenient. Compared with LOCE, the mass flow in the LACE combustor is 3–4 times more than that of LOCE because of the great amount of inert nitrogen. LACE cannot reach such high-combustion pressures; therefore, the LACE combustor is much larger.

2) In the LOCE cycle, separated nitrogen is reheated and used to cool the nozzle expansion part, then it ejects at moderate temperature and pressure to offset the additional drag. In the LACE, nitrogen goes directly into the combustion chamber and incurs large drag.

3) The turbine expander is used in LOCE to liquefy air as well as to drive the hydrogen and oxygen pumps. LOCE works along an isentropic cycle whose efficiency is much higher, whereas in LACE, the J–T valve performs at constant enthalpy.

4) High-temperature hydrogen drives the turbine to compress the low-temperature air, thus effective liquefaction is realized.

5) Nitrogen, which accounts for four-fifths of the air, is compressed to low pressure at its highest density, thus the turbine power is reduced greatly.

6) The gas-liquid separator can be used as a liquid-oxygen container to increase engine thrust in the takeoff phase.

7) Specific impulse may reach 5300 s, and is five times compared with the LACE under the same conditions.

8) The thrust chamber size is reduced greatly, so that the thrust-to-weight ratio is expected to reach 30–40. At the same conditions, the ratio for LACE is expected to reach only about 10–20.

Conclusions

1) It is possible to combine rocket technology with airbreathing technology. This is the technology coordination needed for future aeronautical and astronautical fields. The most important issue involved in this technology is to resolve the mismatch of the high chamber pressures of rocket engines and the low chamber pressures of airbreathing engines.

2) An innovative airbreathing/rocket combined-cycle engine is presented in this paper. LOCE has the advantages of both rocket and airbreathing engines. It makes full use of the four characteristics of liquid hydrogen and its technology challenges are moderate. Its liquefaction efficiency is 5–7 times higher than that of the conventional LACE cycle. It has high specific impulse and high thrust-to-weight ratio. It is a real combination of rocket technology and airbreathing technology and one of the most promising propulsion systems at the low-speed phase (Ma = 0–5) of flight.

3) The TSTO system based on LOCE has a payload-to-takeoff weight ratio from 1.14 to 3.13. The takeoff gross weight is from 196.67 to 432.66 tons according to the extent of adoption of advanced technologies.

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