JOURNAL OF PROPULSION AND POWER
Vol. 16, No. 1, January–February 2000

Third Way of Development of Single-Stage-to-Orbit Propulsion

V. V. Balepin*
MSE Technology Applications, Inc., Butte, Montana 59702
M. Maita†
National Aerospace Laboratory, Tokyo 182, Japan
and
S. N. B. Murthy‡
Purdue University, West Lafayette, Indiana 47907-1003

Considering a number of options for a space-launch vehicle-propulsion system, between advanced rocket motors and airbreathers, in particular, thermally integrated rocket-based combined-cycles, a new cycle that has been given the name KLIN (meaning wedge in Russian), is proposed as the "third way." It consists of a combination of a liquid rocket engine and a deep-cooled turbojet (with oxygen addition to atmospheric air). Currently, the KLIN concept is considered for application with a vertical takeoff horizontal landing vehicle, with the turbojet and the rocket engine optimized individually, with a possibility for incorporating an aerospike-type nozzle. Retaining a rocket trajectory up to Mach 3, the turbojet and rocket engine are assumed to operate together from takeoff with a gradual reduction in the deep-cooled turbojet output, finally terminating the turbojet at Mach 6. It can be shown that the KLIN can be manufactured with available or foreseeable technology, and provides a combination of engine weight and specific impulse that yields twice the payload mass fraction, in addition to other advantages, compared with a rocket-engine-operated vehicle.

Nomenclature

C_O	= concentration of onboard oxygen, 100% by assumption
C_O^A	= concentration of oxygen in normal air, 23.15%
C_O^Σ	= concentration of oxygen in air/oxygen mixture, $C_O^A(1-\varphi)+\varphi C_O$
F	= local specific thrust
I	= local specific impulse
$\bar{I}_{ABE}$	= average specific impulse in the airbreathing phase
$\bar{I}_{LRE}$	= average specific impulse of the liquid rocket engine
K_A	= air-to-hydrogen ratio or cooling ratio
K_0	= oxygen–hydrogen stoichiometric ratio, 7.937
R_{TW}	= vehicle thrust-to-mass ratio
R_0	= initial thrust-to-mass ratio of vehicle
r	= initial thrust-to-mass ratio of propulsion system
V_f	= flight speed
W_g	= exhaust gas velocity
ΔV_{LRE}	= speed increment in rocket phase of flight
δ_{AF}	= airframe fraction in gross takeoff weight
δ_{EN}	= engine fraction in gross takeoff weight
δV_{ABE}	= drag and gravity losses in airbreathing phase
δV_{LRE}	= drag and gravity losses in liquid rocket phase
ε	= equivalence ratio of combined propulsion
φ	= mass fraction of onboard oxygen in the mixed oxidizer

Introduction

IT is universally agreed that the opportunities in space and space technologies may be realized in full only if the space-launch vehicle becomes recoverable at any stage of the mission, and with only a modest amount of refurbishing, is reusable over a sufficiently large number of mission flights. It is also agreed that propulsion is the major aspect of the vehicle that must be improved, whether the goals are long-range or near-term. For the long-term goals, there are sufficient arguments to show that the ultimate objective could be a horizontal takeoff single-stage-to-orbit (HTO SSTO) or TSTO system that is recoverable, except for the fuel consumed, and including the payload, unless it has been delivered; and that the largest part of the trajectory should be traversed using ambient atmospheric air, whereas rocket motors that are used should have the most effective specific thrust and specific impulse. For the near term, on the other hand, there is considerably greater debate in regard to propulsion and takeoff and landing. The parameters that can lead to success are, again, a major increase in specific impulse and thrust and a substantial decrease in the dry weight fraction of the vehicle. Two approaches have resulted from the near-term-use studies: 1) obtaining the best rocket motors, and 2) developing rocket-based combined-cycle engines.[1] Under approach 1, a variety of options have been considered, such as fuel improvement, dual fuel, dual mixture ratio, dual position nozzle, dual throat, etc. However, there is a concern, e.g., Refs. 2 and 3, that the mixed-mode rocket propulsion may only be marginally better than conventional, high-pressure hydrogen–oxygen rockets. However, more definitive positions may evolve from the X-33 project in the U.S. and somewhat similar studies elsewhere. Regarding approach 2, there is considerable scope for innovative improvements, as shown, e.g., in Refs. 4 and 5, although many of the required technologies must be proved in themselves and as part of an integrated propulsion system for a chosen mission objective.

With this background, one can ask if there is an option, a "third way" in the near term, between the airbreathing SSTO and the classic rocket: in particular, if there is a combined-cycle propulsion that can be realized in the near term without a need for large leaps in technology. The objective of this paper is to present such an option in the form of a combined cycle that is designated KLIN (meaning wedge in Russian). [KLIN is a trademark of V. Balepin and MSE Technology Applications, Inc., for a combined deep-cooled turbojet–liquid rocket engine (DCTJ–LRE) propulsion.]

Low-Speed Propulsion Tradeoffs

A brief discussion of propulsion tradeoffs is presented in the following text to explain the genesis of the third way.

Presented as Paper 96-4495 at the AIAA 7th International Space Planes and Hypersonic Systems and Technologies Conference, Norfolk, VA, 18–22 November 1996; received 6 February 1998; revision received 30 November 1998; accepted for publication 5 December 1998.

*Staff Scientist. Senior Member AIAA.
†Head of Spaceplane System Studies. Senior Member AIAA.
‡Senior Reseacher. Associate Fellow AIAA.

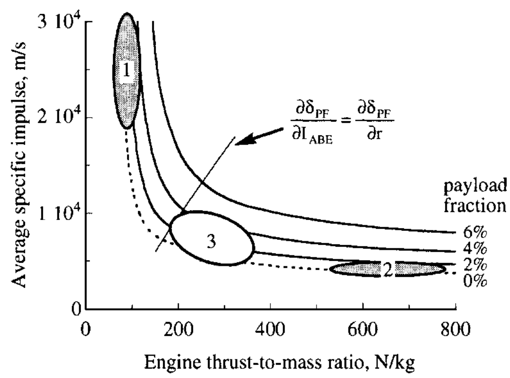


Fig. 1 Variation of payload fraction as a function of thrust-to-mass ratio and average specific impulse of engine: 1, airbreathing propulsion; 2, rocket propulsion; and 3, combined propulsion.

The performance of a launch vehicle may be examined in a diagram representing the payload mass fraction δ_{PF} as a function of the specific impulse and initial thrust-to-mass ratio of the propulsion system. The payload mass fraction is given by the following equation based on the Tsiolkovsky analysis:

$$\delta_{PF} = \frac{\exp[-(V_{ABE} + \delta V_{ABE})/I_{ABE}]}{\exp[(\Delta V_{LRE} + \delta V_{LRE})/I_{LRE}]} - \delta_{\mathrm{AF}} - \delta_{\mathrm{EN}} \tag{1}$$

Here, V_{ABE} corresponds to the vehicle speed at the termination of the airbreathing phase, e.g., at Mach 5–6.

Referring to Fig. 1, which illustrates Eq. (1) in the $I_{ABE} - r$ space, it can be observed that I_{ABE} and r have completely different influences on the payload fraction, except along the line on which the following relation is satisfied:

$$\frac{\partial \delta_{PF}}{\partial I_{ABE}} I_{ABE} = \frac{\partial \delta_{PF}}{\partial r} r$$

and that is described by the relation:

$$r = I_{ABE} \bar{R}_0 \exp\left(\frac{V_{ABE} + \delta V_{ABE}}{I_{ABE}}\right)\left[\exp\left(\frac{V_{ABE} + \delta V_{ABE}}{I_{ABE}}\right) - 1\right] \Big/ V_{ABE} + \delta V_{ABE} \tag{2}$$

Values of average specific impulse and engine thrust-to-mass ratio typical for airbreathing propulsion correspond to region 1 in Fig. 1. It is seen that an increase of thrust-to-mass ratio is the proper way of increasing the payload fraction. An increase in specific impulse has only a modest influence on the payload fraction. In contrast, for pure rocket propulsion, indicated by region 2 in the same figure, an increase in specific impulse becomes the preferable way to a higher payload fraction, and an increase of engine thrust-to-mass ratio beyond the modern level of $r = 600$–800 N/kg has very little effect on efficiency.

Such direct ways of improving vehicle performance are extremely difficult to realize. In the case of airbreathing propulsion, one needs to reduce the turbomachinery mass in an engine that has to operate over a wide range of air inlet temperatures and intake mass flows. Similarly, in the case of rocket propulsion, one needs to increase the specific impulse that is currently close to the theoretical limit for a hydrogen–oxygen cycle. Some gain, of course, can be made by incorporating an aerospike nozzle that improves the specific impulse at low altitudes compared with other nozzles. However, neither of the engines can be improved with respect to the desired parameters to the extent that the vehicle efficiency can increase substantially.

Now, turning attention to region 3 of Fig. 1, this corresponds to combined-cycle engines with moderate specific impulse and a relatively lightweight structure for the engine. That region is in the vicinity of the line represented by Eq. (2), and thus provides considerable opportunity for vehicle perfomance improvement. If a combined cycle is considered that includes a turbojet, one can introduce deep cooling of incoming air and realize both a high-pressure ratio and a reduction in the structural weight of the turbomachinery, which can be manufactured with light alloys or even composites, and thus obtain a high specific impulse. One can also visualize an improvement in air-processing efficiency with a gain in specific thrust, and a reduction in intake and turbomachinery size. Thus, the combined-cycle engines of region 3 may be considered as the broad third way between the pure airbreathing engines (considered in the NASP, the HOTOL, and the TU-2000 vehicle plans) and the pure rocket with no or minimal aerodynamic lift force (Delta Clipper and Venture Star).

A number of combined-cycle engines have been considered in the literature. However, a proper choice depends on the vehicle mission. In fact, very few airbreathers are suitable for combining with vertical takeoff (VTO) rockets.[4–6] Vehicles with no or minimal aeroassist are very sensitive to propulsion system mass. Also, wingless rockets have a rather high trajectory that is not optimal for most of the airbreathers. With such constraints, three choices of airbreathers for combining with VTO rockets are discussed in Refs. 5–7; they are based on the optimal use of hydrogen properties: 1) a liquid air cycle engine with mixed (air + oxygen) oxidizer (LACE–M), 2) LRE combined with an air-separation system, and 3) a deep-cooled turbojet thermally integrated with LRE. The hydrogen usage in each case brings about thermal integration of the LRE and the airbreather. These engines have relatively high thrust-to-mass ratios at moderate specific impulses, and belong to the class of engines in area no. 3 of Fig. 1.

They are truly rocket-based combined cycles with internal functional integration between the rocket and the airbreathing units, both of which are to be operated with hydrogen fuel. The integration is not only on a thermal basis, but can also be achieved in hardware, as in the case of LACE–M. The vehicle body can be chosen as a lifting body, providing a limited lifting force. They do not require added thermal protection for atmospheric ascent. They have the added advantage of requiring minimum onboard fuel for powered descent.

Thermally Integrated Rocket-Based Combined-Cycle Engines

The three combined-cycle engines chosen are shown in Figs. 2–4. They may be represented schematically by a single, generalized scheme, as in Fig. 5, in which the class of thermally integrated engines is shown as consisting of an air-processor system, a propellant-feed system that includes storage of the collected oxygen, a combustor that could be common to the rocket and the airbreather, and a thrustor nozzle.

The contribution of the (liquid) rocket engine to the performance of the combined-cycle engine can be defined indirectly through its relation to the oxygen concentration in the air–oxygen (the oxygen from the onboard supply) mixture oxidizer. To be able to compare such diverse engines on a common basis, we proceed as follows.

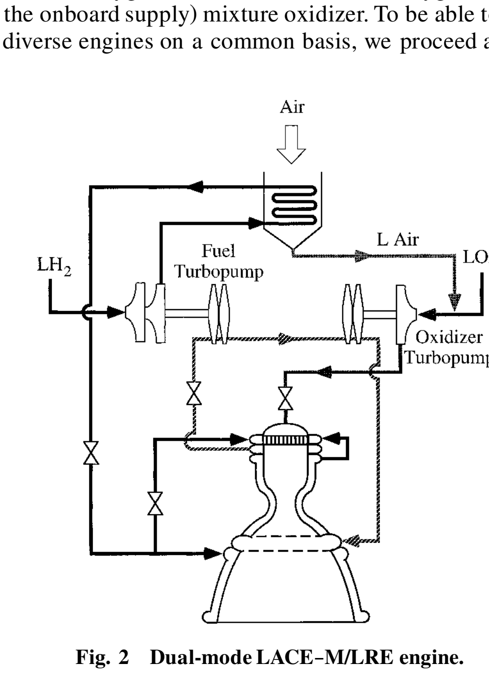


Fig. 2 Dual-mode LACE–M/LRE engine.

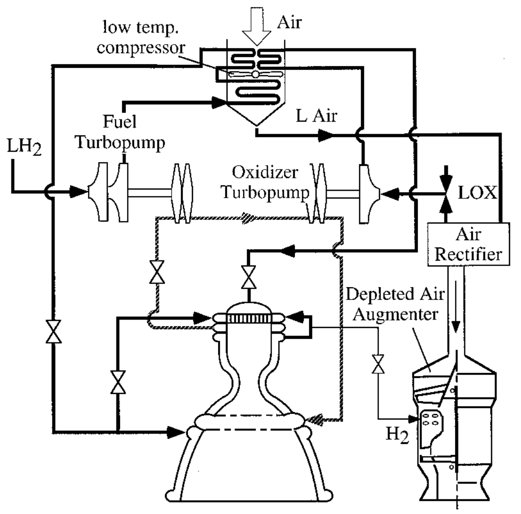


Fig. 3 LRE thermally integrated with air-separation system.

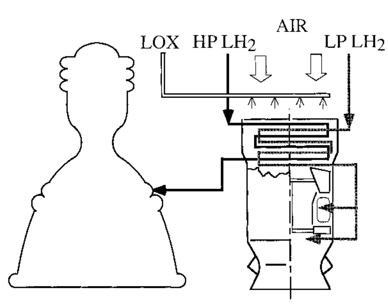


Fig. 4 LRE thermally integrated with DCTJ with oxygen addition, the KLIN cycle.

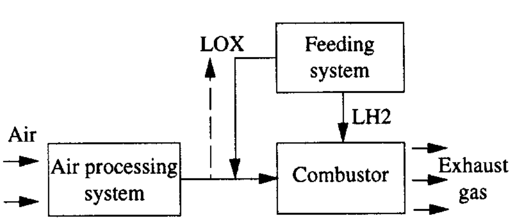


Fig. 5 Schematic representation of thermally integrated rocket–airbreather combination.

The overall performance of the combined cycle can be described by the local values of I and F generated, which can be related to φ and ε, as follows:

$$I = \frac{\left\{1 + \left[C_O^A(1-\varphi) - \varphi C_O \,/\, K_0/\varepsilon\right]\right\} W_g - (1-\varphi)V_f}{\chi\varphi + \left(C_O^A(1-\varphi) - \varphi C_O \,/\, K_0/\varepsilon\right)} \tag{3}$$

$$F = \left\{\frac{1}{1-\varphi} + \frac{C_O^A + [\varphi/(1-\varphi)]C_O}{K_0/\varepsilon}\right\} W_g - V_f \tag{4}$$

It may be pointed out that the foregoing relations apply to combined-cycle engines using an air–oxygen mixture as the oxidizer as well as to those with oxygen extraction and collection and afterburning of the resulting oxygen-depleted air. In the latter case, φ is less than zero or negative because oxygen is extracted and not added to air. The factor χ is equal to zero for oxygen-collection systems, and for onboard oxygen-consuming systems, χ is equal to unity.

As an example, assuming a nozzle expansion ratio π_n equal to 15, the calculated values of relative specific impulse and relative specific thrust at sea-level conditions, normalized with respect to the value based on $C_O^\Sigma = 23.15\%$, are shown in Fig. 6 as a function of oxygen concentration in the oxidizer for different values of equivalence ratio.

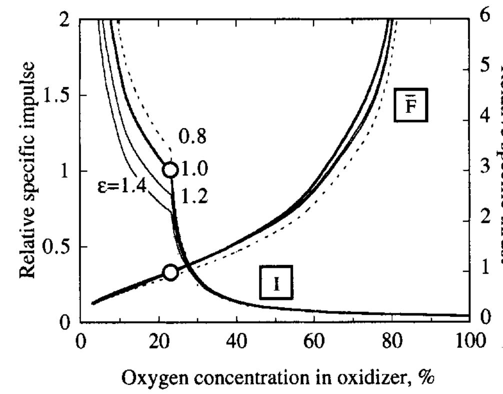


Fig. 6 Variation of relative specific impulse and relative thrust as a function of oxygen concentration in oxidizer with equivalence ratio as parameter.

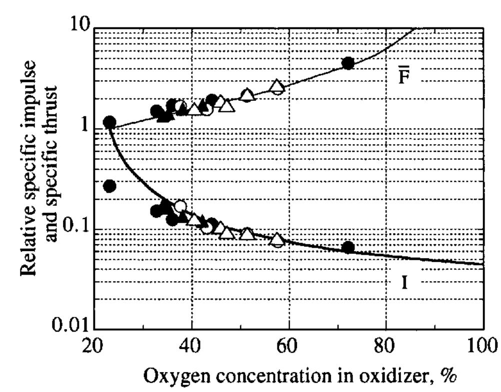


Fig. 7 Performance of different thermally integrated systems at equivalence ratio of unity. •, LACE–M, Ref. 5; ○, LRE + FLOX, Ref. 5; ▲, LRE + DCTJ, Ref. 6; and △ LRE + DCJ_{ox}, Ref. 7.

When onboard oxygen is utilized in the oxidizer, $C_O^\Sigma > 23.15\%$, specific impulse undergoes a dramatic reduction, e.g., by 45% when C_O^Σ is increased to 25%. As shown in Fig. 6, the specific impulse continues to decrease steeply until C_O^Σ is nearly 40%, and then, by a gradual reduction, reaches the value for a rocket for which C_O^Σ is 100% by definition. On the other hand, specific thrust, as in Fig. 6, increases gradually at C_O^Σ around the value of 23.15%, and beginning with $C_O^\Sigma = 60\%$ it increases rapidly until it becomes infinite, for a rocket with a C_O^Σ equal to 100%.

In the special case of oxygen collection, if the depleted air–hydrogen mixture provides all of the thrust required for the collection mode, the oxygen concentration in the air utilized for propulsion C_O^A is obviously below the standard atmosphere value. Such a special case yields an increase in specific impulse. For example, assuming a recovery factor of 80% during air separation, oxygen concentration in air becomes $C_O^\Sigma = 5\%$, and the stoichiometric ratio for depleted air–hydrogen mixture becomes very high ($L_0 = 160$), and if that mixture is utilized for propulsion, one can observe the substantial increase in specific impulse obtained in Fig. 6. However, the special case is not included for consideration in this paper.

Returning to the thermally integrated combined-cycle systems, Fig. 7 provides the specific impulse and specific thrust for the generalized case of Fig. 5, with ε equal to unity. The performance values for the four different engines are also shown in Fig. 7. It is clear that the performance values generally coincide with the general curves of Fig. 7, a remarkable fact considering the very different nature of the four engines considered. The only noticeable disagreement arises in the case of the LACE cycle (black points) at a lower oxygen concentration, particularly without onboard oxygen usage, i.e., $C_O^\Sigma = 23.15\%$. It should be noted, however, that in this case, ε is about 3.0–4.5, whereas the base predictions of Fig. 7 are for ε equal to unity. Also, the far left and right (black point) predicted values correspond to a very small payload fraction, a regime of little practical interest. Most of the other cases shown in Fig. 7 correspond to a payload fraction increase of 2.0–2.5% and lie in the range of C_O^{Σ} equal to 35–50%, or φ of 15–35%.

It is also of interest to note that, for the LRE–DCTJ and the LRE–deep-cooled turbojet with oxygen addition to air for ice prevention ($DCTJ_{ox}$) engines, considering the reasonable ranges of air-cooling ratios, the air-to-hydrogen ratio becomes about 6–16, which corresponds to 50–70% contribution of thrust by the DCTJ engine to the total sea-level thrust.

In conclusion, the LRE–DCTJ incorporating existing or feasible technology appears to be the best of the three thermally integrated systems considered and a significant option for near-term SSTO application. This is the basis for the proposed KLIN cycle.

KLIN Cycle Configuration and Operation

Main Features

The main features of the KLIN cycle are as follows:

1) The LRE–$DCTJ_{ox}$ combined-cycle engine incorporates several rocket and DCTJ units. The rockets operate from takeoff. The LRE units may be throttled or even turned off for a part of the trajectory once the vehicle has passed through the transonic regime, returning to full usage when the DCTJ are turned off. The DCTJ units are intended for operation over the initial ascent from takeoff with a gradual reduction in thrust output until they are finally turned off at Mach 6. They could be detachable from the vehicle, recoverable, and reusable. The mission abort capability is also built into the system.

2) The vehicle is currently intended for VTO operation. Several options can be considered for landing, including horizontal landing; however, landing is not discussed in this paper.

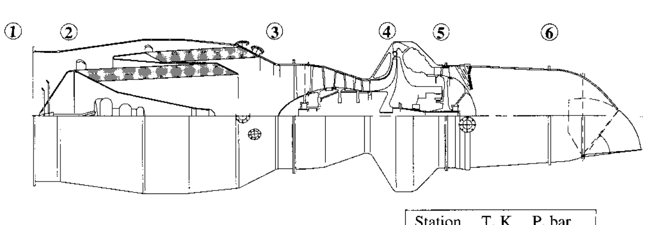
The mission is assumed to be SSTO. However, because the DCTJ could be detachable and the fuel tank can be integrated in several ways, a TSTO mission vehicle is also feasible.

3) The total hydrogen fuel required for the rocket and the DCTJ units is utilized for deep cooling of air to about 110 K SLS design conditions and to 200–250 K at Mach 6.

The DCTJ (Fig. 8) incorporates LOX addition to incoming air upstream of the precooler, in the order of 4–6%, to prevent any possibility of icing of the precooler from sea level to different altitudes in different seasons.

4) The LRE and DCTJ could be operated under near-stoichiometric conditions for best volumetric specific impulse. However, this is not found to be optimal, and the DCTJ engines need to be overfueled from the start, with an increase in overfueling as the flight speed increases. The two component engine areas were assumed to have separate combustors, unlike some other cases, e.g., RB545, ATRDC, and SABRE, with a single combustor.

5) Because the entry temperature of air into turbomachinery is quite low, the turbomachinery Mach number increases. However, the DCTJ can incorporate light, supersonic through-flow compressor because the maximum air temperature behind it is not expected to exceed 400–450 K, even at Mach 6. DCTJ units are cut off one-by-one during acceleration, and so the corrected airflow through the compressor can be kept nearly constant, a significant advantage.



Station	T, K	P, bar
1	288	0.96
2	240	0.96
3	110	0.82
4	330	24.6
5	1700	23.9
6	2300	13.0

Fig. 8 Schematic of cryojet engine.

6) During ascent, the optimal vehicle trajectory depends on the use of the component engines, e.g., the vehicle trajectory may be the rocket trajectory up to Mach 3.0.

KLIN Cycle Operation

The LRE units must provide a thrust-to-weight ratio for the vehicle R_{TW} larger than unity at transition from the combined LRE–DCTJ mode to one of LRE operations to accelerate the vehicle further. This requires that during the combined LRE–DCTJ mode of operation, either all of the available LRE units are utilized and the DCTJ contribution is reduced, or only a limited number of available rocket motors is utilized along with the DCTJ units, the remaining rocket motors being utilized in the posttransition regime. (In the latter case, it is of interest to note that the LRE thrust-to-mass ratio may not be much different from that for deeply throttled LRE; i.e., there may not be much difference between the cases in which a few of the total number of rocket motors are turned off or all of them are deeply throttled.) However, in both types of LRE–DCTJ systems, there is a loss of efficiency; in the first case, due to the reduction in specific impulse, and in the second case, due to the increased mass of the overall propulsion system.

The estimates have been carried out assuming that all of the rocket motors are in operation all of the time ($\psi = 1$) and setting R_0 equal to 1.3, this is sometimes stated as the optimum for an SSTO rocket vehicle.[2]

The results of the calculations are presented in Fig. 9 as a function of air-to-hydrogen ratio, or equivalently K_A at different values of φ, ranging from 0 to 20%. Overall efficiency is characterized by the difference between the payload fraction of the KLIN-powered vehicle and the basic pure rocket SSTO shown in Fig. 9a. Several observations from Fig. 9 are as follows:

1) The maximum payload fraction corresponds to lower values of $K_A = 10$–12, depending on oxygen addition (Fig. 9a).

2) The lower optimal range of K_A corresponds to high specific impulse (Fig. 9b), intermediate (or average) value of specific thrust (Fig. 9c), intermediate and beginning of a sharp increase in value of the engine mass (Fig. 9d) and of tanks and engine mass (Fig. 9e), and an intermediate (or average) value for oxygen and hydrogen mass fractions (Fig. 9f).

3) The influence of additional φ on a payload fraction is not very significant (Fig. 9a). However, this parameter provides some flexibility in the selection of other parameters, e.g., a higher φ value reduces the engine mass fraction (Fig. 9d).

It should be noted that a design point with lower air-cooling ratios may be preferable. Although this leads to a reduction in the payload fraction value, there is an important advantage of reduced risk in the development of such components as the inlet and the cooler, that become smaller and lighter with the reduced cooling ratio.

The feasibility and reliability of the KLIN cycle can be verified in most respects through on-ground simulation of various components of the engine. A demonstrator of the DCTJ of the KLIN cycle can be developed based on an existing small, low-cost turbojet.

Mass Breakdown

A preliminary estimation of mass breakdown for the KLIN cycle vehicle, assuming the LRE–DCTJ reference system of Ref. 7 and an equatorial mission and 200-km orbit, is presented in Table 1.

Table 1 Mass breakdown for SSTO rockets[a]

Component mass	SSTO rocket[8]	Basic SSTO with pure rocket propulsion[7]	SSTO with KLIN propulsion ($DCTJ_{ox}$)[7]
LOX	75.71	75.93	68.63
LH_2	12.62	12.67	14.80
LOX tank	——	1.14	1.03
LH_2 tank	——	2.28	2.66
Propulsion system	1.51	2.28	4.71
Orbital mass	11.67	11.42	16.57
Payload fraction	2.39	Basic level	Basic level + 2.49

[a]% of GTOW.

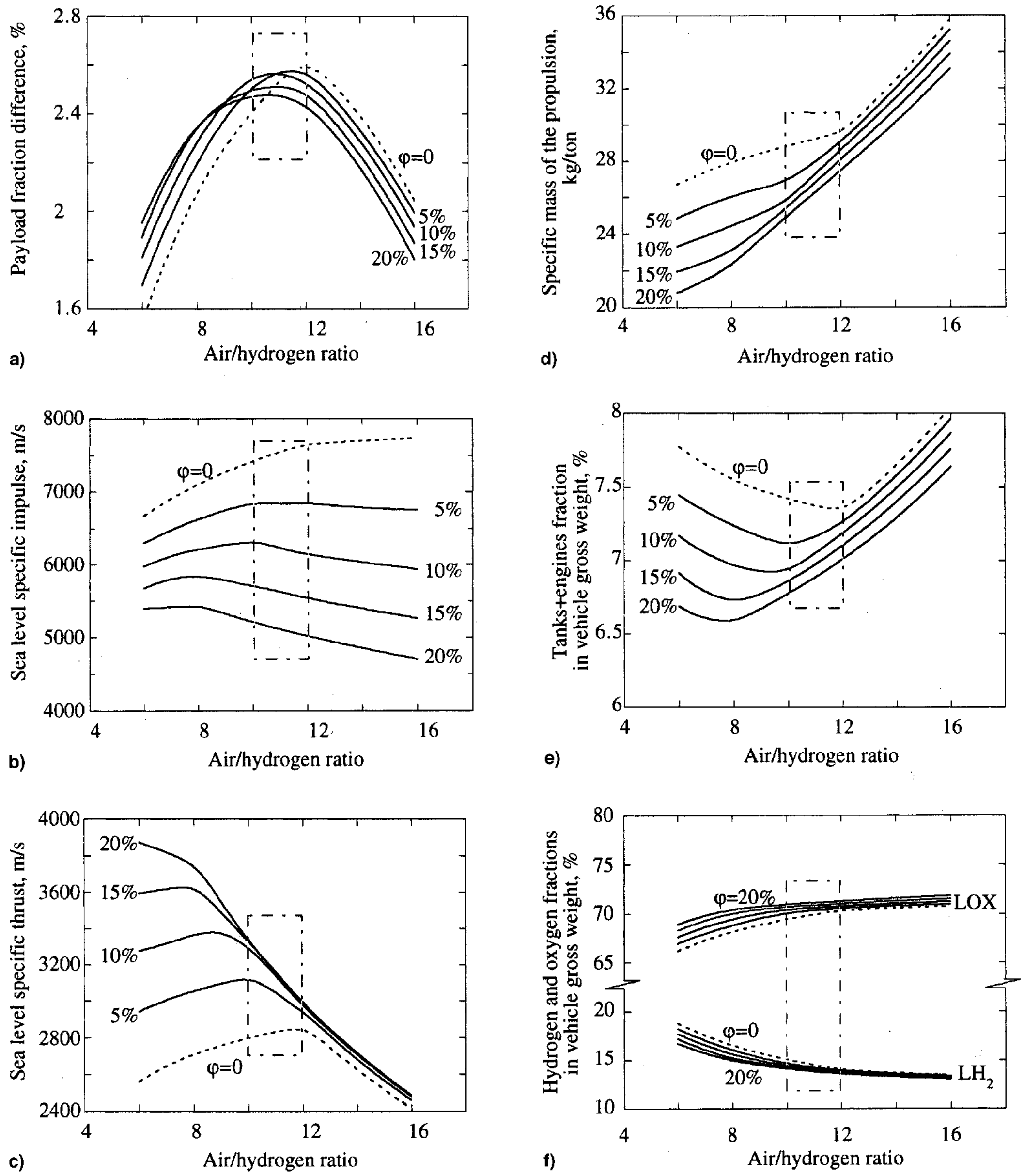


Fig. 9 Comparison of mass and efficiency parameters for LRE–DCTJ combined propulsion system with oxygen addition as a parameter.

The table includes a comparison with an SSTO rocket, and also, an SSTO rocket with a lifting body.[8] The KLIN cycle vehicle provides an additional payload mass fraction of 2.49% relative to the SSTO with pure rocket propulsion. The benefit arises from substantial savings in oxygen with the KLIN cycle propulsion. The net payload fraction gain is illustrated in Fig. 10.

The savings in oxygen mass [about 7.3% of gross takeoff weight (GTOW) for the case in Fig. 10] not only compensates for the increase in engine mass (2.4% of GTOW), tankage (0.27%), and the increased hydrogen consumption (2.1% of GTOW) but, in fact, permits an increase in payload mass by the previously mentioned 2.49% of GTOW.

Flight Scenario for Lifting-Body Launcher with KLIN Cycle Propulsion

The KLIN cycle will reveal its full advantage when incorporated in a lifting-body-shape vehicle. In this case, optimal propulsion control provides higher oxygen savings. One can visualize a flight scenario consisting of four rather different operational modes.[9] A profile of the vehicle thrust-to-weight ratio in the four modes is given in Fig. 11.

Mode 1 (from takeoff until Mach ≈ 0.8) corresponds to the simultaneous operation of all DCTJ units with oxygen augmentation and all or part of LRE units (LREs could also be throttled). Maximum absolute thrusts necessary for vertical takeoff and moderate specific impulse are produced. Oxygen injection in front of the precooler protects the device from icing and provides about 20% of DCTJ thrust increase with nearly the same engine mass.

Mode 2 (in the range of Mach ≈ 0.8–1.2) begins after oxygen injection cutoff. Initially, thrust decreases along with a specific impulse increase. Later, a gradual recovery of thrust follows.

Mode 3 (in the range of Mach ≈ 1.2–6.0) begins with the LREs cutoff. Only DCTJ units operate in this mode. Thrust decreases dramatically with specific impulse reaching its highest value in this mode. During acceleration, thrust continues to decrease, and aeroassist is required for ascent. By the end of this mode, the thrust-to-weight ratio of the vehicle is significantly below unity.

Mode 4 (from Mach = 6.0 to orbital speed), a pure rocket mode begins after DCTJ cutoff and LRE switch-on. The use of lifting force is no longer necessary. The condition of vehicle thrust-to-weight ratio being equal to or greater than unity should be assumed for the beginning of the rocket mode.

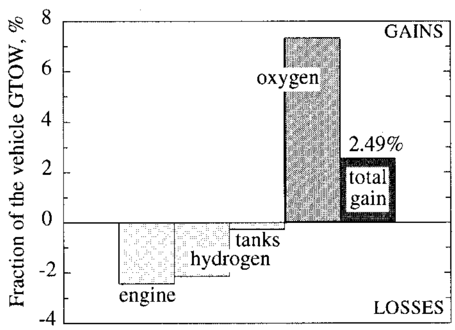


Fig. 10 Breakdown of payload fraction gain in KLIN cycle engine compared with a LRE vehicle.

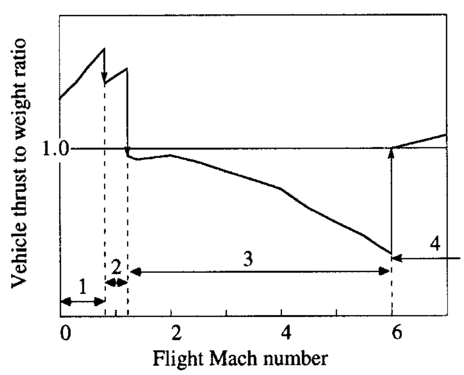


Fig. 11 Profile of the thrust-to-weight ratio for the lifting-body vehicle powered by KLIN cycle.

Cycle fuel efficiency is as always in conflict with engine mass. It should be noted that the main task of low-speed modes 1 and 2 is to provide the lowest engine weight-to-thrust ratio, even at the cost of the cycle efficiency. The main task during a mode with more extensive use of the lifting force (mode 3) is to provide high specific impulse and significant onboard oxygen savings. According to Ref. 9, oxygen savings of up to 30–35% of the vehicle GTOW compared with the pure rocket system could be provided by a KLIN cycle with temporary LRE cutoff in the Mach range of 1.2–6.0.

Conclusions

Rocket-based combined cycles involving air precooling, liquefaction, and separation may be considered as the third way between the pure airbreathing engines (considered in the NASP, the HOTOL, and the TU-2000 vehicle plans) and the pure rocket with none or minimal aerodynamic lift force.

The KLIN cycle based on the thermally integrated LRE–DCTJ system is a truly combined cycle with all of its benefits, and provides a significant option, with a payload mass fraction of nearly 4%, for near-term space launch vehicles with fully abortable, recoverable, and reusable capability. Although the inclusion of turbomachinery increases the mass of the propulsion system, the introduction of deep precooling provides a savings in oxygen mass and a major reduction in vehicle mass.

Considering the projected and demonstrated developments under the U.S. IHPTET and other turbomachinery-centered developments, and the practically demonstrated, nonicing, precooler technology with an airturboramjet (ATREX) engine in Japan,[10] the KLIN cycle engine is fully within a foreseeable near-term industrial capability.

The KLIN cycle includes several margins in the cooling ratio, the onboard oxygen usage, the fuel equivalence ratio, etc., so that the full potential of the system can be developed on a systematic basis with entirely on-ground testing.

The current study deals with a conical vehicle for what could be considered as either an SSTO vehicle with part of the propulsion system recoverable or a TSTO vehicle with other advantages of staging. The KLIN geometrical configuration can be integrated with aeroassist lift combined with the use of features such as an aerospike nozzle, yielding both major aerothermodynamic and structure advantages, with an appropriate trajectory optimization.

Acknowledgment

This paper was submitted through the Western Environmental Technology Office under DOE Contract DE-AC22-96EW96405.

References

[1]Escher, W. J. D., "A Winning Combination for Tomorrow's Spaceliners," *Aerospace America*, Feb. 1996, pp. 38–43.

[2]Mansci, D., and Martin, J. A., "Evaluation of Innovative Rocket Engines for Single-Stage Earth-to-Orbit Vehicles," *Journal of Propulsion and Power*, Vol. 7, No. 6, 1991, pp. 929–937.

[3]Mansci, D., and Fina, A., "Advanced Rocket Propulsion Systems for Reusable Ballistic Single-Stage-to-Orbit Vehicles BETA and Delta-Clipper," AIAA Paper 94-3316, June 1994.

[4]Czysz, P. A., "Rocket Based Combined Cycle (RBCC) Propulsion Systems Offer Additional Options," *11th International Symposium on Air Breathing Engines (ISABE)*, Tokyo, Sept. 1993.

[5]Balepin, V. V., Yoshida, M., and Kamijo, K.,"Rocket Based Combined Cycles for Single Stage Rocket," *SAE Aerospace Atlantic Conference*, Dayton, OH, Society of Automotive Engineers, Paper 94-1166, April 1994.

[6]Balepin, V. V., Folomeev, E. A., Galkin, S. M., and Tjurikov, E. V., "Rocket Based Combined Cycles for Vertical Take-Off Space Vehicles," AIAA Paper 95-6078, April 1995.

[7]Balepin, V. V., Maita, M., Tanatsugu, N., and Murthy, S. N. B., "Deep Cooled Turbojet Augmented with Oxygen (Cryojet) for a SSTO Launch Vehicle," AIAA Paper 96-3036, July 1996.

[8]Baumgarther, R. I., and Elvin, J. D., "Lifting Body—An Innovative RLV Concept," AIAA Paper 95-3531, July 1995.

[9]Balepin, V. V., and Maita, M., "KLIN Cycle: Combined Propulsion for Vertical Take Off Launcher," AIAA Paper 97-2854, July 1997.

[10]Tanatsugu, N., Balepin, V., Sato, T., Mizutani, T., Hamabe, K., and Tomike, J., "ATREX Engine Development: First Practical Experience of Precooled Turbomachinery," *Proceedings of the 5th International Symposium Propulsion in Space Transportation*, Association Aeronautique et Astronautique de France, France, 1996, pp. 19.25–19.29.