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# A Comparison of Propulsion Concepts for SSTO Reusable Launchers

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This paper discusses the relevant selection criteria for a single stage to orbit (SSTO) propulsion system and then reviews the characteristics of the typical engine types proposed for this role against these criteria. The engine types considered include Hydrogen/Oxygen ($H_2/O_2$) rockets, Scramjets, Turbojets, Turborockets and Liquid Air Cycle Engines. In the authors opinion none of the above engines are able to meet all the necessary criteria for an SSTO propulsion system simultaneously. However by selecting appropriate features from each it is possible to synthesise a new class of engines which are specifically optimised for the SSTO role. The resulting engines employ precooling of the airstream and a high internal pressure ratio to enable a relatively conventional high pressure rocket combustion chamber to be utilised in both airbreathing and rocket modes. This results in a significant mass saving with installation advantages which by careful design of the cycle thermodynamics enables the full potential of airbreathing to be realised. The SABRE engine which powers the SKYLON launch vehicle is an example of one of these so called 'Precooled hybrid airbreathing rocket engines' and the conceptual reasoning which leads to its main design parameters are described in the paper.

**Keywords:** Reusable launchers, SABRE, SKYLON, SSTO

## 1. Introduction

Several organisations world-wide are studying the technical and commercial feasibility of reusable SSTO launchers. This new class of vehicles appear to offer the tantalising prospect of greatly reduced recurring costs and increased reliability compared to existing expendable vehicles. However achieving this breakthrough is a difficult task since the attainment of orbital velocity in a re-entry capable single stage demands extraordinary propulsive performance.

Most studies to date have focused on high pressure hydrogen/oxygen ($H_2/O_2$) rocket engines for the primary propulsion of such vehicles. However it is the authors opinion that despite recent advances in materials technology such an approach is not destined to succeed, due to the relatively low specific impulse of this type of propulsion. Airbreathing engines offer a possible route forward with their intrinsically higher specific impulse. However their low thrust/weight ratio, limited Mach number range and high dynamic pressure trajectory have in the past cancelled any theoretical advantage.

By design review of the relevant characteristics of both rockets and airbreathing engines this paper sets out the rationale for the selection of deeply precooled hybrid airbreathing rocket engines for the main propulsion system of SSTO launchers as exemplified by the SKYLON vehicle [1].

## 2. Propulsion Candidates

This paper will only consider those engine types which would result in politically and environmentally acceptable vehicles. Therefore engines employing nuclear reactions (eg: onboard fission reactors or external nuclear pulse) and chemical engines with toxic exhausts (eg: fluorine/oxygen) will be excluded.

The candidate engines can be split into two broad groups, namely pure rockets and engines with an airbreathing component. Since none of the airbreathers are capable of accelerating an SSTO vehicle all the way to orbital velocity, a practical vehicle will always have an onboard rocket engine to complete the ascent. Therefore the use of airbreathing has always been proposed within the context of improving the specific impulse of pure rocket propulsion during the initial lower Mach portion of the trajectory.

Airbreathing engines have a much lower thrust/weight ratio than rocket engines (≈10%) which tends to offset the advantage of reduced fuel consumption. Therefore vehicles with airbreathing engines invariably have wings and employ a lifting trajectory in order to reduce the installed thrust requirement and hence the airbreathing engine mass penalty. The combination of wings and airbreathing engines then demands a low flat trajectory (compared to a ballistic rocket trajectory) in order to maximise the installed performance (i.e. (thrust-drag)/fuel flow). This high dynamic pressure trajectory gives rise to one of the drawbacks of an airbreathing approach since the airframe heating and loading are increased during the ascent which ultimately reflects in increased structure mass. However the absolute level of mass growth depends on the relative severity of the ascent as compared with reentry which in turn is mostly dependant on the type of airbreathing engine selected. An additional drawback to the low trajectory is increased drag losses particularly since the vehicle loiters longer in the lower atmosphere due to the lower acceleration, offset to some extent by the much reduced gravity loss during the rocket powered ascent.

Importantly however, the addition of a set of wings brings more than just performance advantages to airbreathing vehicles. They also give considerably increased abort capability since a properly configured vehicle can remain in stable flight with up to half of its propulsion systems shutdown. Also during reentry the presence of wings reduces the ballistic coefficient thereby reducing the heating and hence thermal protection system mass, whilst simultaneously improving the vehicle lift/drag ratio permitting greater crossrange.

The suitability of the following engines to the SSTO launcher role will be discussed since these are representative of the main types presently under study within various organisations world-wide:

- Liquid Hydrogen/Oxygen rockets
- Ramjets and Scramjets
- Turbojets/Turborockets and variants
- Liquid Air Cycle Engines (LACE) and Air Collection Engines (ACE)
- Precooled hybrid airbreathing rocket engines (RB545/SABRE)

## 3. Selection Criteria

The selection of an ‘optimum’ propulsion system involves an assessment of a number of interdependant factors which are listed below. The relative importance of these factors depends on the severity of the mission and the vehicle characteristics.

1) Engine performance

   Useable Mach number and altitude range.

   Installed specific impulse.

   Installed thrust/weight.

   Performance sensitivity to component level efficiencies.

2) Engine/Airframe integration

   Effect on airframe layout (Cg/Cp pitch trim & structural efficiency).

   Effect of required engine trajectory (Q and heating) on airframe technology/materials.

3) Technology level

   Materials/structures/aerothermodynamic and manufacturing technology.

4) Development cost

   Engine scale and technology level.

   Complexity and power demand of ground test facilities.

   Necessity of an X plane research project to precede the main development program.

## 4. Hydrogen/Oxygen Rocket Engines

Hydrogen/oxygen rocket engines achieve a very high thrust/weight ratio (60-80) but relatively low specific impulse (450-475 secs in vacuum) compared with conventional airbreathing engines. Due to the relatively large $\Delta V$ needed to reach low earth orbit (approx 9 km/s including gravity and drag losses) in relation to the engine exhaust velocity, SSTO rocket vehicles are characterised by very high mass ratios and low payload fractions.

The $H_2/O_2$ propellant combination is invariably chosen for SSTO rockets due to its higher performance than other alternatives despite the structural penalties of employing a very low density cryogenic fuel. In order to maximise the specific impulse, high area ratio nozzles are required which inevitably leads to a high chamber pressure cycle in order to give a compact installation and reduce back pressure losses at low altitude. The need to minimise back pressure losses normally results in the selection of some form of altitude compensating nozzle since conventional bell nozzles have high divergence and overexpansion losses when running in a separated condition.

The high thrust/weight and low specific impulse of $H_2/O_2$ rocket engines favours vertical takeoff wingless vehicles since the wing mass and drag penalty of a lifting trajectory results in a smaller payload than a steep ballistic climb out of the atmosphere. The ascent trajectory is therefore extremely benign (in terms of dynamic pressure and heating) with vehicle material selection determined by re-entry. Relative to airbreathing vehicles a pure rocket vehicle has a higher density (gross take off weight/volume) due to the reduced hydrogen consumption which has a favourable effect on the tankage and thermal protection system mass.

In their favour rocket engines represent broadly known (current) technology, are ground testable in simple facilities, functional throughout the whole Mach number range and physically very compact resulting in good engine/airframe integration. Abort capability for an SSTO rocket vehicle would be achieved by arranging a high takeoff thrust/weight ratio (eg: 1.5) and a large number of engines (eg: 10) to permit shutdown of at least two whilst retaining overall vehicle control. From an operational standpoint SSTO rockets will be relatively noisy since the high takeoff mass and thrust/weight ratio results in an installed thrust level up to 10 times higher than a well designed airbreather.

Reentry should be relatively straightforward providing the vehicle reenters base first with active cooling of the engine nozzles and the vehicle base. However the maximum lift/drag ratio in this attitude is relatively low (approx 0.25) limiting the maximum achievable crossrange to around 250 km. Having reached a low altitude some of the main engines would be restarted to control the subsonic descent before finally effecting a tailfirst landing on legs. Low crossrange is not a particular problem providing the vehicle operator has adequate time to wait for the orbital plane to cross the landing site. However in the case of a military or commercial operator this could pose a serious operational restriction and is consequently considered to be an undesirable characteristic for a new launch vehicle.

In an attempt to increase the crossrange capability some designs attempt nosefirst re-entry of a blunt cone shaped vehicle or alternatively a blended wing/body configuration. This approach potentially increases the lift/drag ratio by reducing the fuselage wave drag and/or increasing the aerodynamic lift generation. However the drawback to this approach is that the nosefirst attitude is aerodynamically unstable since the aft mounted engine package pulls the empty center of gravity a considerable distance behind the hypersonic center of pressure. The resulting pitching moment is difficult to trim without adding nose ballast or large control surfaces projecting from the vehicle base. It is expected that the additional mass of these components is likely to erode the small payload capability of this engine/vehicle combination to the point where it is no longer feasible.

Recent advances in materials technology (eg: fibre reinforced plastics and ceramics) have made a big impact on the feasibility of these vehicles. However the payload fraction is still very small at around 1-2% for an Equatorial low Earth orbit falling to as low as 0.25% for a Polar orbit. The low payload fraction is generally perceived to be the main disadvantage of this engine/vehicle combination and has historically prevented the development of such vehicles, since it is felt that a small degree of optimism in the preliminary mass estimates may be concealing the fact that the ‘real’ payload fraction is negative.

One possible route forward to increasing the average specific impulse of rocket vehicles is to employ the atmosphere for both oxidiser and reaction mass for part of the ascent. This is an old idea dating back to the 1950’s and revitalised by the emergence of the BAe/Rolls Royce ‘HOTOL’ project in the 1980’s [2]. The following sections will review the main airbreathing engine candidates and trace the design background of precooled hybrid airbreathing rockets.

## 5. Ramjet and Scramjet Engines

A ramjet engine is from a thermodynamic viewpoint a very simple device consisting of an intake, combustion and nozzle system in which the cycle pressure rise is achieved purely by ram compression. Consequently a separate propulsion system is needed to accelerate the vehicle to speeds at which the ramjet can takeover (Mach 1-2). A conventional hydrogen fuelled ramjet with a subsonic combustor is capable of operating up to around Mach 5-6 at which point the limiting effects of dissociation reduce the effective heat addition to the airflow resulting in a rapid loss in nett thrust. The idea behind the scramjet engine is to avoid the dissociation limit by only partially slowing the airstream through the intake system (thereby reducing the static temperature rise) and hence permitting greater useful heat addition in the now supersonic combustor. By this means scramjet engines offer the tantalising prospect of achieving a high specific impulse up to very high Mach numbers. The consequent decrease in the rocket powered $\Delta V$ would translate into a large saving in the mass of liquid oxygen required and hence possibly a reduction in launch mass.

Although the scramjet is theoretically capable of generating positive nett thrust to a significant fraction of orbital velocity it is unworkable at low supersonic speeds. Therefore it is generally proposed that the internal geometry be reconfigured to function as a conventional ramjet to Mach 5 followed by transition to scramjet mode. A further reduction of the useful speed range of the scramjet results from consideration of the nett vehicle specific impulse ((thrust-drag)/fuel flow) in scramjet mode as compared with rocket mode. This tradeoff shows that it is more effective to shut the scramjet down at Mach 12-15 and continue the remainder of the ascent on pure rocket power. Therefore a scramjet powered launcher would have four main propulsion modes: a low speed accelerator mode to ramjet followed by scramjet and finally rocket mode. The proposed low speed propulsor is often a ducted ejector rocket system employing the scramjet injector struts as both ejector nozzles to entrain air at low speeds and later as the rocket combustion chambers for the final ascent.

Whilst the scramjet engine is thermodynamically simple in conception, in engineering practice it is the most complex and technically demanding of all the engine concepts discussed in this paper. To make matters worse many studies including the recent ESA 'Winged Launcher Concept' study have failed to show a positive payload for a scramjet powered SSTO since the fundamental propulsive characteristics of scramjets are poorly suited to the launcher role. The low specific thrust and high specific impulse of scramjets tends to favour a cruise vehicle application flying at fixed Mach number over long distances, especially since this would enable the elimination of most of the variable geometry.

Scramjet engines have a relatively low specific thrust (nett thrust/airflow) due to the moderate combustor temperature rise and pressure ratio, and therefore a very large air mass flow is required to give adequate vehicle thrust/weight ratio. However at constant freestream dynamic head the captured air mass flow reduces for a given intake area as speed rises above Mach 1. Consequently the entire vehicle frontal area is needed to serve as an intake at scramjet speeds and similarly the exhaust flow has to be re-expanded back into the original streamtube in order to achieve a reasonable exhaust velocity. However employing the vehicle forebody and aftbody as part of the propulsion system has many disadvantages:

- The forebody boundary layer (up to 40% of the intake flow) must be carried through the entire shock system with consequent likelihood of upsetting the intake flow stability. The conventional solution of bleeding the boundary layer off would be unacceptable due to the prohibitive momentum drag penalty.
- The vehicle undersurface must be flat in order to provide a reasonably uniform flowfield for the engine installation. The flattened vehicle cross section is poorly suited to pressurised tankage and has a higher surface area/volume than a circular cross section with knock-on penalties in aeroshell, insulation and structure mass.
- Since the engine and airframe are physically inseparable little freedom is available to the designer to control the vehicle pitch balance. The single sided intake and nozzle systems positioned underneath the vehicle generate both lift and pitching moments. Since it is necessary to optimise the intake and nozzle system geometry to maximise the engine performance it is extremely unlikely that the vehicle will be pitch balanced over the entire Mach number range. Further it is not clear whether adequate CG movement to trim the vehicle could be achieved by active propellant transfer.
- Clustering the engines into a compact package underneath the vehicle results in a highly interdependant flowfield. An unexpected failure in one engine with a consequent loss of internal flow is likely to unstart the entire engine installation precipitating a violent change in vehicle pitching moment.

In order to focus the intake shock system and generate the correct duct flow areas over the whole Mach range, variable geometry intake/combustor and nozzle surfaces are required. The large variation in flow passage shape forces the adoption of a rectangular engine cross section with flat moving ramps thereby incurring a severe penalty in the pressure vessel mass. Also to maximise the installed engine performance requires a high dynamic pressure trajectory which in combination with the high Mach number imposes severe heating rates on the airframe. Active cooling of significant portions of the airframe will be necessary with further penalties in mass and complexity.

Further drawbacks to the scramjet concept are evident in many areas. The nett thrust of a scramjet engine is very sensitive to the intake, combustion and nozzle efficiencies due to the exceptionally poor work ratio of the cycle. Since the exhaust velocity is only slightly greater than the incoming freestream velocity a small reduction in pressure recovery or combustion efficiency is likely to convert a small nett thrust into a small nett drag. This situation might be tolerable if the theoretical methods (CFD codes) and engineering knowledge were on a very solid footing with ample correlation of theory with experiment. However the reality is that the component efficiencies are dependant on the detailed physics of poorly understood areas like flow turbulence, shock wave/boundary layer interactions and boundary layer transition. To exacerbate this deficiency in the underlying physics existing ground test facilities are unable to replicate the flowfield at physically representative sizes, forcing the adoption of expensive flight research vehicles to acquire the necessary data.

Scramjet development could only proceed after a lengthy technology program and even then would probably be a risky and expensive project. In 1993 Reaction Engines estimated that a 130 tonne scramjet vehicle development program would cost $25B (at fixed prices) assuming that the program proceeded according to plan. This program would have included two X planes, one devoted to the subsonic handling and low supersonic regime and the other an air dropped scramjet research vehicle to explore the Mach 5-15 regime.

## 6. Turbojets, Turborockets and Variants

In this section are grouped those engines that employ turbocompressors to compress the airflow but without the aid of precoolers. The advantage of cycles that employ onboard work transfer to the airflow is that they are capable of operation from sea level static conditions. This has important performance advantages over engines employing solely ram compression and additionally enables a cheaper development program since the mechanical reliability can be acquired in relatively inexpensive open air ground test facilities.

### 6.1 Turbojets

Turbojets (Fig. 1) exhibit a very rapid thrust decay above about Mach 3 due to the effects of the rising compressor inlet temperature forcing a reduction in both flow and pressure ratio. Compressors must be operated within a stable part of their characteristic bounded by the surge and choke limits. In addition structural considerations impose an upper outlet temperature and spool speed limit. As inlet temperature rises (whilst operating at constant $W\sqrt{T}/P$ and $N/\sqrt{T}$) the spool speed and/or outlet temperature limit is rapidly approached. Either way it is necessary to throttle the engine by moving down the running line, in the process reducing both flow and pressure ratio. The consequent reduction in nozzle pressure ratio and mass flow results in a rapid loss in nett thrust.

However at Mach 3 the vehicle has received an insufficient boost to make up for the mass penalty of the airbreathing engine. Therefore all these cycles tend to be proposed in conjunction with a subsonic combustion ramjet mode to higher Mach numbers. The turbojet would be isolated from the hot airflow in ramjet mode by blocker doors which allow the airstream to flow around the core engine with small pressure loss. The ramjet mode provides reasonable specific thrust to around Mach 6-7 at which point transition to rocket propulsion is effected.

Despite the ramjet extension to the Mach number range the performance of these systems is poor due mainly to their low thrust/weight ratio. An uninstalled turbojet has a thrust/weight ratio of around 10. However this falls to 5 or less when the intake and nozzle systems are added which compares badly with a $H_2/O_2$ rocket of 60+.

### 6.2 Turborocket

The turborocket (Fig. 2) cycles represent an attempt to improve on the low thrust/weight of the turbojet and to increase the useful Mach number range. The pure turborocket consists of a low pressure ratio fan driven by an entirely separate turbine employing $H_2/O_2$ combustion products. Due to the separate turbine working fluid the matching problems of the turbojet are eased since the compressor can in principle be operated anywhere on its characteristic. By manufacturing the compressor components in a suitable high temperature material (such as reinforced ceramic) it is possible to eliminate the ramjet bypass duct and operate the engine to Mach 5-6 whilst staying within outlet temperature and spool speed limits. In practice this involves operating at reduced nondimensional speed $N/\sqrt{T}$ and hence pressure ratio. Consequently to avoid choking the compressor outlet guide vanes a low pressure ratio compressor is selected (often only 2 stages) which permits operation over a wider flow range. The turborocket is considerably lighter than a turbojet. However the low cycle pressure ratio reduces the specific thrust at low Mach numbers and in conjunction with the preburner liquid oxygen flow results in a poor specific impulse compared to the turbojet.

### 6.3 Expander Cycle Turborocket

This cycle is a variant of the turborocket whereby the turbine working fluid is replaced by high pressure regeneratively heated hydrogen warmed in a heat exchanger located in the exhaust duct (Fig. 3). Due to heat exchanger metal temperature limitations the combustion process is normally split into two stages (upstream and downstream of the ma-

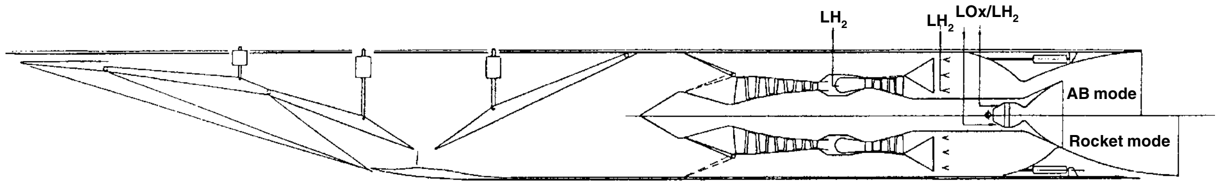


**Fig. 1 Turbo-ramjet Engine (with integrated rocket engine).**

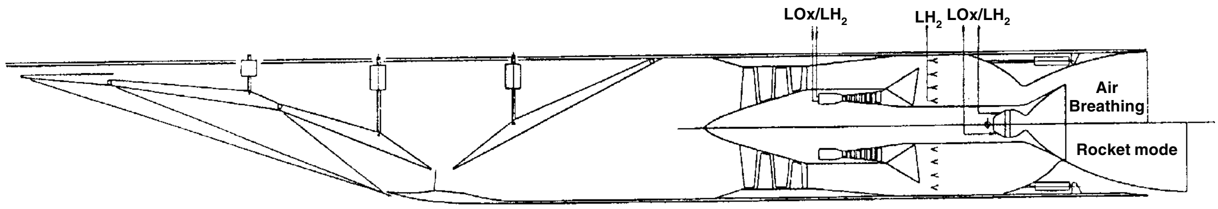


**Fig. 2 Turborocket.**

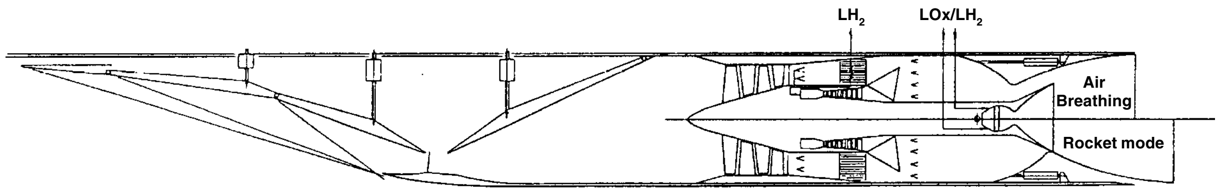


**Fig. 3 Turbo-expander engine.**

trix) and the turbine entry temperature is quite low at around 950K. This variant exhibits a moderate improvement in specific impulse compared with the pure turborocket due to the elimination of the liquid oxygen flow. However this is achieved at the expense of additional pressure loss in the air ducting and the mass penalty of the heat exchanger.

Unfortunately none of the above engines exhibit any performance improvement over a pure rocket approach to the SSTO launcher problem, despite the wide variations in core engine cycle and machinery. This is for the simple reason that the core engine masses are swamped by the much larger masses of the intake and nozzle systems which tend to outweigh the advantage of increased specific impulse.

Due to the relatively low pressure ratio ramjet modes of these engines, it is essential to provide an efficient high pressure recovery variable geometry intake and a variable geometry exhaust nozzle. The need for high pressure recovery forces the adoption of 2 dimensional geometry for the intake system due to the requirement to focus multiple oblique shockwaves over a wide mach number range. This results in a very serious mass penalty due to the inefficient pressure vessel cross section and the physically large and complicated moving ramp assembly with its high actuation loads. Similarly the exhaust nozzle geometry must be capable of a wide area ratio variation in order to cope with the widely differing flow conditions ($W\sqrt{T}/P$ and pressure ratio) between transonic and high Mach number flight. A further complication emerges due to the requirement to integrate the rocket engine needed for the later ascent into the airbreathing engine nozzle. This avoids the prohibitive base drag penalty that would result from a separate 'dead' nozzle system as the vehicle attempted to accelerate through transonic.

## 7. Liquid Air Cycle Engines (LACE) and Air Collection Engines (ACE)

Liquid Air Cycle Engines were first proposed by Marquardt in the early 1960's. The simple LACE engine exploits the low temperature and high specific heat of liquid hydrogen in order to liquify the captured airstream in a specially designed condenser (Fig. 4). Following liquifaction the air is relatively easily pumped up to such high pressures that it can be fed into a conventional rocket combustion chamber. The main advantage of this approach is that the airbreathing and rocket propulsion systems can be combined with only a single nozzle required for both modes. This results in a mass saving and a compact installation with efficient base area utilisation. Also the engine is in principle capable of operation from sea level static conditions up to perhaps Mach 6-7.

The main disadvantage of the LACE engine however is that the fuel consumption is very high (compared to other airbreathing engines) with a specific impulse of only about 800 secs. Condensing the airflow necessitates the removal of the latent heat of vaporisation under isothermal conditions. However the hydrogen coolant is in a supercritical state

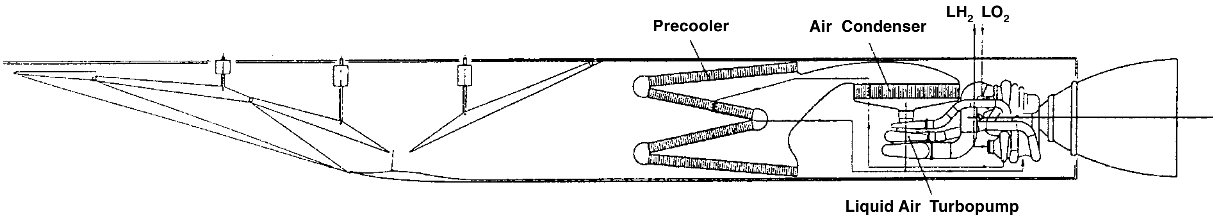


**Fig. 4 Liquid Air Cycle Engine (LACE).**

following compression in the turbopump and absorbs the heat load with an accompanying increase in temperature. Consequently a temperature 'pinch point' occurs within the condenser at around 80K and can only be controlled by increasing the hydrogen flow to several times stoichiometric. The air pressure within the condenser affects the latent heat of vaporisation and the liquifaction temperature and consequently has a strong effect on the fuel/air ratio. However at sea level static conditions of around 1 bar the minimum fuel/air ratio required is about 0.35 (ie: 12 times greater than the stoichiometric ratio of 0.029) assuming that the hydrogen had been compressed to 200 bar. Increasing the air pressure or reducing the hydrogen pump delivery pressure (and temperature) could reduce the fuel/air ratio to perhaps 0.2 but nevertheless the fuel flow remains very high. At high Mach numbers the fuel flow may need to be increased further, due to heat exchanger metal temperature limitations (exacerbated by hydrogen embrittlement limiting the choice of tube materials). To reduce the fuel flow it is sometimes proposed to employ slush hydrogen and recirculate a portion of the coolant flow back into the tankage. However the handling of slush hydrogen poses difficult technical and operational problems.

From a technology standpoint the main challenges of the simple LACE engine are the need to prevent clogging of the condenser by frozen carbon dioxide, argon and water vapour. Also the ability of the condenser to cope with a changing 'g' vector and of designing a scavenge pump to operate with a very low NPSH inlet. Nevertheless performance studies of SSTO's equipped with LACE engines have shown no performance gains due to the inadequate specific impulse in airbreathing mode despite the reasonable thrust/weight ratio and Mach number capability.

The Air Collection Engine (ACE) is a more complex variant of the LACE engine in which a liquid oxygen separator is incorporated after the air liquifier. The intention is to takeoff with the main liquid oxygen tanks empty and fill them during the airbreathing ascent thereby possibly reducing the undercarriage mass and installed thrust level. The ACE principal is often proposed for parallel operation with a ramjet main propulsion system. In this variant the hydrogen fuel flow would condense a quantity of air from which the oxygen would be separated before entering the ramjet combustion chamber at a near stoichiometric mixture ratio. The liquid nitrogen from the separator could perform various cooling duties before being fed back into the ramjet airflow to recover the momentum drag.

The oxygen separator would be a complex and heavy item since the physical properties of liquid oxygen and nitrogen are very similar. However setting aside the engineering details, the basic thermodynamics of the ACE principal are wholly unsuited to an SSTO launcher. Since a fuel/air mixture ratio of approximately 0.2 is needed to liquify the air and since oxygen is 23.1% of the airflow it is apparent that a roughly equal mass of hydrogen is required to liquify a given mass of oxygen. Therefore there is no saving in the takeoff propellant loading and in reality a severe structure mass penalty due to the increased fuselage volume needed to contain the low density liquid hydrogen.

## 8. Precooled Hybrid Airbreathing Rocket Engines

This last class of engines is specifically formulated for the SSTO propulsion role and combines some of the best features of the previous types whilst simultaneously overcoming their faults. The first engine of this type was the RB545 powerplant designed for the HOTOL spaceplane and originally devised by Alan Bond in 1982. Building on the experience gained in the course of the HOTOL project the thermodynamics of this first engine were refined during 1989-90 resulting in the SABRE powerplant designed for SKYLON.

The global specification for this type of engine results from an assessment of the necessary propulsion characteristics for a successful SSTO spaceplane:

- At the termination of the airbreathing ascent the engine must revert to a high specific impulse closed cycle rocket mode.

- The airbreathing mode must be capable of open test bed operation in order to minimise development costs. This implies some form of turbomachinery to compress the airflow and has the added advantage that the engine is capable of accelerating the vehicle from rest.
- The airbreathing trajectory must be relatively benign to both the engine and the airframe implying a maximum airbreathing/rocket transition of around Mach 6-7. Equally the airbreathing mode must be capable of propelling the vehicle up to at least Mach 5 in order to achieve a worthwhile reduction in the rocket powered $\Delta V$.
- The installed thrust/weight in airbreathing mode must be higher than conventional ramjets and turborockets whilst the specific impulse must be a considerable improvement on the LACE engine.
- The mass penalty of the airbreathing machinery must be as small as possible whilst the propulsion system must occupy a minimum base area. Employing a common nozzle system for both modes results in a mass saving and eliminates the base drag penalty of a 'dead' nozzle. A further mass saving can be achieved if the same pumps and preburner assemblies can be employed in both modes.

From the above list it is clear that the LACE engine is close to meeting the requirements except for its high fuel consumption. Therefore the RB545 engine resulted as an evolution from the LACE cycle in order to improve its specific impulse.

The excessive fuel flow of the LACE engine is entirely due to the quantity of coolant required to effect the condensation process. Yet the work capacity of the hot high pressure gaseous hydrogen stream that emerges from the precooler/condenser remains largely unexploited since the power demands of the liquid air turbopump are minimal. The RB545 cycle was born out of recognition that a more efficient split between the cooling and work demands of the cycle could be achieved by avoiding the air liquifaction process altogether. By terminating the cooling process close to but not below the vapour boundary ($\approx$80K) the 'pinch point' was avoided with a very large saving in the required coolant flow, whilst still leaving sufficient hydrogen to drive a high pressure ratio turbocompressor sufficiently powerful to compress the airflow up to typical rocket combustion chamber pressures. With this cycle the optimum compressor inlet temperature that minimises the total hydrogen flow is actually on the vapour boundary since the compressor work demands are greater than the cooling requirements. Nevertheless by deep precooling of the incoming airstream the compressor work demand is greatly reduced and excessive compressor outlet temperatures are avoided particularly at high Mach numbers. Also unlike a simple turbojet the engine does not suffer from a reduction in gross thrust with increasing Mach number since the precooler 'irons out' the intake recovery air temperature variation allowing the compressor to operate with a nearly constant inlet temperature.

In practice the RB545 employed the high pressure hydrogen delivery from the hydrogen turbopump to cool the airstream directly, following which the hydrogen stream split. Approximately one-third passed to the main combustion chamber via the preburner whilst the remaining two-thirds was expanded through the turbocompressor turbine prior to exhaust. This cycle reduced the fuel/air ratio to approximately 0.1. However this was eroded at high Mach numbers due to precooler metal temperature limitations caused by hydrogen embrittlement. Apart from the improved specific impulse of this cycle most of the technology problems of the LACE engine are avoided (eg: two phase heat exchangers and liquid air handling). The design of the turbomachinery and heat exchanger surfaces are relatively conventional although there remains the problem of preventing atmospheric moisture clogging the precooler with frost. Somewhat surprisingly the total engine mass is no greater than the LACE engine since the addition of the turbocompressor is roughly balanced by the elimination of the air condenser and turbopump. The turbocompressor is much lighter than an equivalent compressor drawing ambient air since the low air inlet temperature reduces the physical size of the unit due to the higher air density, and also reduces the rotational speeds (and hence inertial loading) due to the lower speed of sound. Also the low air delivery temperature permits light alloy or composite construction for most of the compressor which combined with the previous factors reduces the mass to approximately one-quarter that of an ambient machine.

The SABRE engine (Figs. 5 and 6) is a more complex variant of the original cycle in which a lower fuel flow has been achieved at the expense of a small mass penalty. In this engine a Brayton cycle helium loop has been interposed between the 'hot' airstream and the 'cold' hydrogen stream. The work output of the helium loop provides the power to drive the air compressor. Employing helium as the working fluid permits superior heat resisting alloys for the precooler matrix and also results in more optimally matched turbine stages. The improved thermodynamics of the SABRE engine result in an air/fuel ratio of about 0.08 which produces a useful

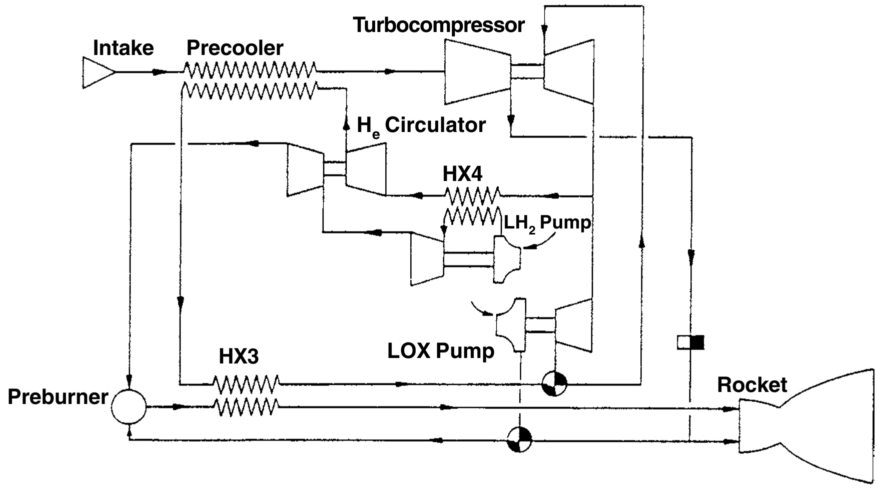


Fig. 5 SABRE engine thermodynamic cycle.

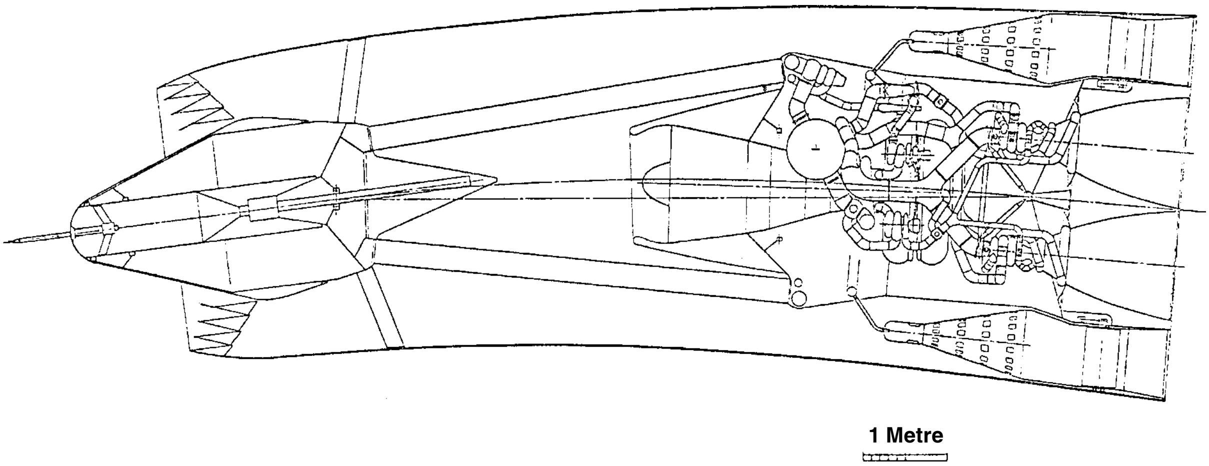


Fig. 6 SABRE vertical cross section.

saving in hydrogen consumption during the airbreathing ascent. Frost control is effected in a more efficient manner on SABRE which also has a favourable impact on the vehicles' payload capability.

A very important factor in the success of these engines is that due to the high internal pressure ratio of the core engine it is not necessary to equip the vehicle with a highly efficient intake system. The SABRE engine operates over the whole trajectory with an inlet pressure of only around 1.3 bar, which enables maximum chamber pressure to be achieved with minimal variations in the turbomachinery operating point. Consequently a 2 shock intake (one oblique and one normal shock) is able to meet the engine demands, enabling the complicated and heavy 2-dimensional intakes typical of low pressure ratio engines to be dispensed with in favour of a simple axisymmetric inlet with a translating centrebody. The simplification of the intake mechanism and the gains from an axisymmetric structure result in an intake mass saving of approximately 80% compared with a high pressure recovery 2D intake. Also, since forebody precompression is unnecessary, the intake assembly can be removed from underneath the vehicle dramatically improving the design freedom to solve trim and airframe layout problems.

At rocket transition the air inlet is closed and the turbocompressor is run-down whilst simultaneously the liquid oxygen turbopump is run-up. The preburner temperature is reduced in rocket mode reflecting the reduced power demand of the liquid oxygen turbopump. The engine features a liquid oxygen cooled main combustion chamber since this permits the same oxidiser injectors to be used in both modes.

## 9. Comparison and Conclusions

The installed specific impulse and thrust/weight ratio of the SABRE engine are shown in Figs. 7 and 8 with the other engine candidates shown for comparison. It is important to note that all the candidates have been assessed using broadly extant materials and aerothermodynamic technology. These figures show that the SABRE engine achieves a specific impulse comparable with turborockets whilst simultaneously attaining installed thrust/weight ratios similar to LACE engines. It is this com-

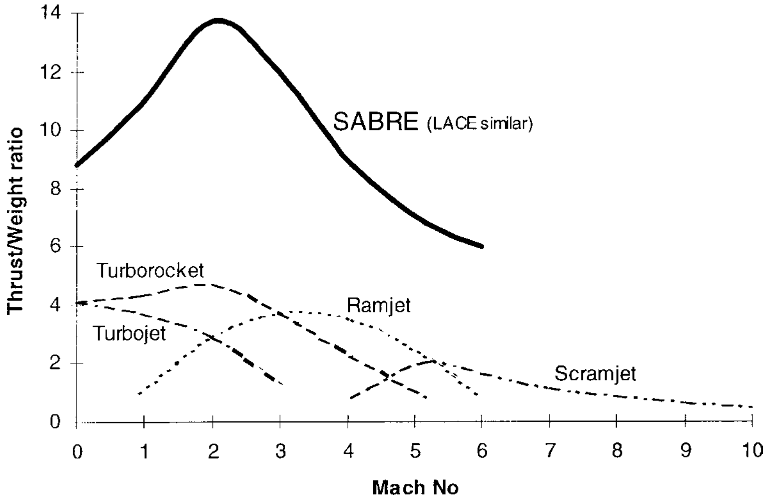


**Fig. 7 Installed thrust/weight ratio of the combined SSTO propulsion installation in airbreathing mode.**

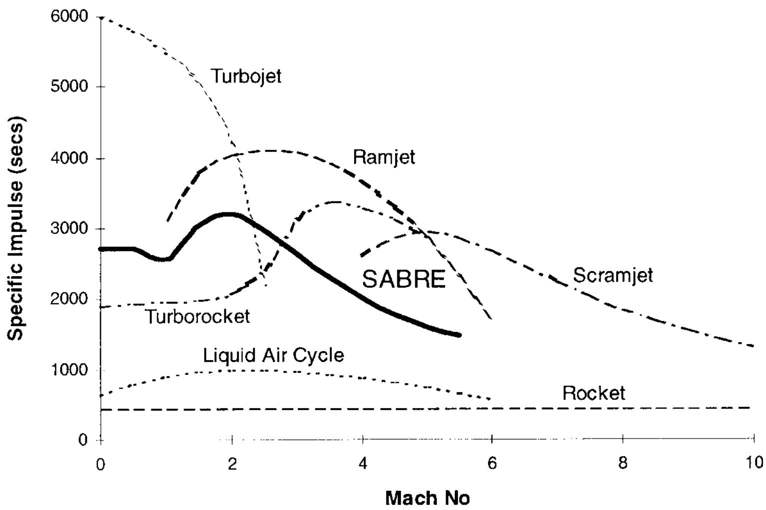


**Fig. 8 Installed specific impulse of SSTO propulsion systems**

bination of moderate specific impulse with low installed weight that makes precooled hybrid engines uniquely suitable for SSTO launch vehicles.

The application of the SABRE engine is described in more detail in [1] which also covers the design of a suitable airframe (SKYLON) which properly harnesses the full potential and unique characteristics of this engine type. The final SABRE/SKYLON combination is capable of placing a 12 tonne payload into an equatorial low Earth orbit at a gross takeoff mass of 275 tonnes (payload fraction 4.36%).

## References

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