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The SKYLON Spaceplane

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SKYLON is a single stage to orbit (SSTO) winged spaceplane designed to give routine low cost access to space. At a gross takeoff weight of 275 tonnes of which 220 tonnes is propellant the vehicle is capable of placing 12 tonnes into an equatorial low Earth orbit. The vehicle configuration consists of a slender fuselage containing the propellant tankage and payload bay with delta wings located midway along the fuselage carrying the SABRE engines in axisymmetric nacelles on the wingtips. The vehicle takes off and lands horizontally on it's own undercarriage. The fuselage is constructed as a multilayer structure consisting of aeroshell, insulation, structure and tankage. SKYLON employs extant or near term materials technology in order to minimise development cost and risk. The SABRE engines have a dual mode capability. In rocket mode the engine operates as a closed cycle liquid oxygen/liquid hydrogen high specific impulse rocket engine. In airbreathing mode (from takeoff to Mach 5) the liquid oxygen flow is replaced by atmospheric air, increasing the installed specific impulse 3-6 fold. The airflow is drawn into the engine via a 2 shock axisymmetric intake and cooled to cryogenic temperatures prior to compression. The hydrogen fuel flow acts as a heat sink for the closed cycle helium loop before entering the main combustion chamber.

Keywords: Spaceplane design, SSTO, SKYLON, SABRE, airbreathing propulsion.

1. Introduction

SKYLON is designed to provide cheap reliable transport into low Earth orbit. It is intended to replace the inadequate launcher systems used today with a practical transport system which will permit a rapid expansion in the volume of traffic into space. Current launchers (expendable multistage rockets) are expensive (approx $150M/flight), unreliable (approx 5% loss rate) and limited to a low launch rate.

To overcome these failings SKYLON has the following characteristics:

- Reusability: Reduced cost per flight by amortising vehicle production cost over several flights. Preliminary calculations indicate that 200 reuses is close to the economic optimum for a first generation spaceplane without incurring performance penalties to achieve a longer fatigue life.
- Single stage (SSTO): Multistage vehicles are more costly to develop and have greatly increased operational costs compared with single stage machines. This is due to the Quality Assurance operations necessary to safely mate and fuel two vehicles, particularly if one of the vehicles is piloted.
- Unpiloted: By employing onboard computers to handle flight control the need for a pilot is eliminated and the size of the mission control centre is reduced. Dispensing with the pilot reduces the vehicle development cost due to the relaxation of required safety standards during the development program and also increases the payload fraction due to saving the pilot cabin mass and life support equipment etc. Although SKYLON is unpiloted it would be capable of carrying passengers by fitting a dedicated module (containing seating and life support systems etc) into the payload bay.
- Abort capability: The vehicle is capable of flying to and landing safely at strategically placed abort sites with up to half its engines shutdown in a similar manner to aircraft. All flight critical systems would feature redundant channels to contain single point failures.
- User friendly operations: Simple ground handling (e.g. tractor towing the vehicle around the spaceport on its own undercarriage) and automated checkout procedures. Minimal maintenance in between flights achieved by robust thermal protection system (TPS) and reliable engines. Simple payload integration achieved through maximum use of containerised payloads dispensing with the requirement for "clean room" payload bay cleanliness and allowing simple interfacing with other forms of terrestrial transport.
- Reentry crossrange: The low reentry wing loading results in a hypersonic lift/drag (L/D) that is sufficient to pull substantial crossrange. This

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increases the number of opportunities that the vehicle can return to the launch site from a highly inclined orbit thus improving the operational flexibility of the system.

- Enviromental impact: SKYLON employs enviromentally friendly propellants (hydrogen/oxygen) and being a single stage vehicle will not add to the orbital debris problem.

2. Previous SSTO Vehicle/Engine Concepts

The operational attractions of single stage to orbit vehicles are widely recognised. However, up until now attempts to design such vehicles have been unsuccessful.

Ballistic SSTO rockets employing high pressure LOX/Hydrogen engines have marginal payload fractions and small reentry crossrange. Improvements in materials have made a substantial impact on the feasibility of these vehicles and McDonnell Douglas pursued this approach with their Delta Clipper project. However the proposed nose first reentry, whilst giving adequate lift/drag ratio, is an aerodynamically unstable attitude and generates a substantial trim imbalance. The addition of wings would solve the trim problem and give abort options but tends to cancel the payload.

To reduce the ΔV required during the rocket ascent the possibility of adding separate air breathing engines has been examined many times. However the low thrust/weight of air breathing engines tends to cancel the advantage of reduced fuel consumption. Therefore to reduce the mass penalty this route often evolves into a two stage to orbit (TSTO) vehicle in which the heavy air breathing engines are effectively disposed of at the end of the air breathing ascent, however in the process losing the operational attractiveness of the SSTO concept. Alternatively the air breathing engines are made to work harder by increasing the transition Mach number from 3-5 achievable with turbomachinery based cycles to 10-15 by the use of scramjets. However, scramjets are poorly suited to the launcher role due to their very low thrust/weight, inability to generate thrust at low Mach numbers and their exceptionally severe flight regime. A scramjet powered vehicle requires technology advances in all aspects of aerospace engineering and additionally cannot be ground tested.

A different approach to reducing the mass of the air breathing engines resulted in the LACE (Liquid Air Cycle) class of engines in which the rocket combustion chamber, pumps, preburner and nozzle are utilised in both modes. These engines employ the cryogenic hydrogen fuel to liquify the airflow prior to pumping. Unfortunately the thermodynamics of this type of cycle result in a very high fuel flow and a relatively massive precooler/condenser. By addressing the faults of the LACE cycle the RB545 engine (HOTOL powerplant) emerged which later evolved into the SABRE engine employed on SKYLON. These new engines cool the air down close to the vapour boundary (but avoid liquifaction) prior to compression in a relatively conventional turbocompressor. The resulting engines have approximately half the fuel flow of the LACE cycle and are capable of operation from zero forward speed up to Mach 5 with a thrust/weight ratio much higher than normal air breathing engines. At transition the air breathing machinery shuts down and the engine reverts to a high performance closed cycle rocket mode.

The British Aerospace HOTOL project represented the first attempt at designing a vehicle powered by such engines. However, this work ceased in 1989 due to lack of government support. The findings of the HOTOL project were analysed by Reaction Engines who then set about redesigning the engines and airframe in 1991, from which the SABRE/SKYLON project was born.

3. The SKYLON Airframe

The SKYLON vehicle consists of a slender fuselage containing the propellants and payload bay with a delta wing located roughly midway along the fuselage (Fig. 1). The SABRE engines are mounted in axisymmetric nacelles on the wingtips. Control authority whilst in the atmosphere is exerted by foreplanes in pitch, ailerons in roll and an aft mounted fin in yaw. During the ascent main engine gimballing takes over progressively as the dynamic pressure reduces until finally handing over to reaction control thrusters at main engine cutoff. The vehicle is capable of taking off and landing from conventional runways on its own undercarriage. The SKYLON configuration evolved from design review of the HOTOL airframe and represents an efficient resolution of the problems encountered by that project.

The HOTOL airframe was derived from conventional vertical takeoff rockets whereby the engines are mounted at the rear of a blunt based fuselage. Since the dry centre of gravity (CG) is dominated by the engine location, the wings and fuelled CG (i.e. lox tank) must also be at the rear. Consequently the payload bay and hydrogen tankage were fitted into a projecting forebody. This configuration suf-

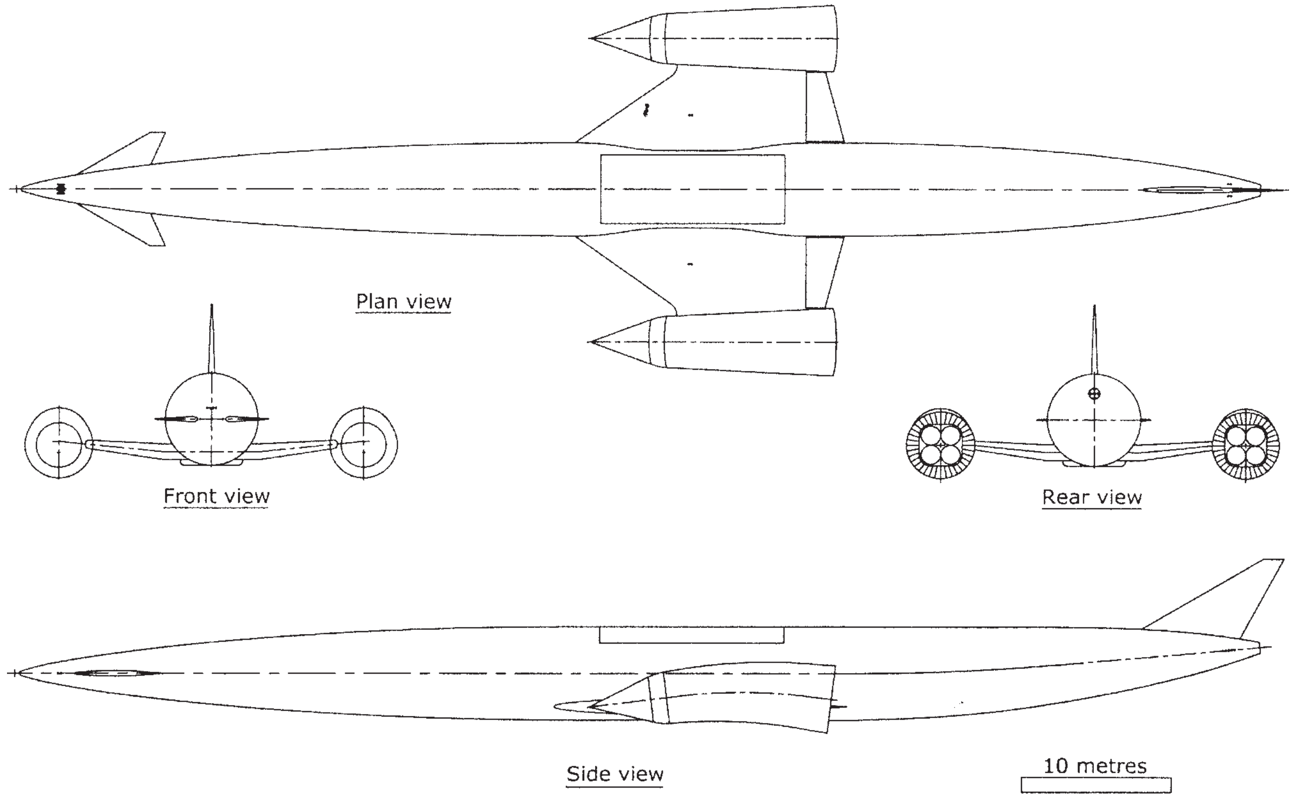

Fig. 1 SKYLON configuration C1.

fered from a severe CP/CG mismatch during the air breathing ascent. The centre of pressure (CP) shifted 10 m forward, partly due to the wide Mach number range and the large fuselage cross section to wing area ratio, but also due to the long overhang of the forward fuselage. To make matters worse the air breathing hydrogen fuel was drawn from the forward tankage, thereby bringing the CG rearwards. To trim the vehicle various alterations were made to the design all of which eroded the payload. At the outset of the project a conventional undercarriage was dispensed with and replaced by a specially designed takeoff trolley in order to improve the marginal payload fraction of the vehicle. Taken together the above problems resulted in a vehicle with serious operational disadvantages and a small payload, where recourse was often made to untried and speculative materials to counter the deficiencies of a poor design.

The SKYLON airframe is a new configuration that attempts to solve the trim and structural problems in a more efficient manner using broadly the same components. To solve the CP/CG problem greater independent control is required over the empty CG. This is achieved by installing the propulsion system in nacelles on the wing tips. By sliding the wing/engine assembly forward it is then possible to arrange the empty CG on the reentry hypersonic CP. To avoid disrupting the trim during reentry when varying payload masses might be carried, the payload bay must be coincident with the CG (ie:over the wing). By carefully arranging the CG slightly ahead of the CP it has been found that the active foreplanes can trim and control the vehicle throughout reentry even up to incidence angles of 70 degrees. Then to balance the fully fuelled vehicle during the ascent it is necessary to split the LOX tank into two, either side of the payload bay since the subsonic CP is close to the wing quarter chord. Finally the hydrogen tanks are also split into two and occupy the ends of the fuselage. By burning off the contents of the aft tank first the CG tracks forward during the ascent (until the aft tank empties), to approximately match the CP travel. Although the total CG movement at 2 m is rather less than the supersonic CP movement (4 m), the design is biased towards achieving a neutral trim at Mach 5.

This bias assists the foreplanes whose control effectiveness is reduced at high Mach number compared with the high lift curve slope and dynamic pressure available during transonic flight. The foreplanes have complete pitch authority during the airbreathing ascent (fixed engines). Since the foreplanes have a large moment arm about the CG, the foreplane trim drags are correspondingly low (i.e. generally less than 1% of total vehicle drag). The engines are actively gimballed during the rocket ascent to minimise the pitching moment and dynamically control the vehicle in pitch, roll and yaw. Once the engines are capable of taking over complete pitch control, the foreplanes are feathered. The engine mean thrust vector points through the empty CG, to give a neutral vehicle during the final stages of the rocket ascent. The mean thrust vector angle (7 degrees) is close to the payload optimum, whilst the requirement to point the thrust vector through the dry CG determines the wing tip location relative to the fuselage centreline. The rocket ascent modelling indicates that +/-3degrees gimbal angle is sufficient to track the vertical movements in CG whilst leaving ample margin for dynamic control. Figure 2 indicates the residual pitching moment about the vehicle CG during the powered ascent.

Additional advantages of the SKYLON configuration are briefly described below:

- The wing area can be optimised for maximum ascent performance (since the trim problem is solved) which combined with the elimination of the flaperons results in a lighter wing structure.
- The hydrogen tank aerodynamic and inertial bending moments are reduced to approximately 25% of their previous values due to the reduced overhang and mass.
- Due to design breakthroughs the takeoff trolley has been eliminated and replaced by an undercarriage capable of a rolling takeoff.
- The base area of the fuselage is eliminated. By increasing the fuselage fineness ratio the fuselage drag is reduced, improving vehicle performance particularly during transonic flight.
- The integrated intake/engine nacelles result in minimum combined base area and reduce the length and mass of inlet ducting. The axisymmetric intake is a simple 2 shock design which, due to the efficient structure and modest recovered pressures (approximately 1.3 bar), is extremely light. The intake operates in a relatively clean flowfield ahead of the wing and therefore does not have to deal with wing or fuselage boundary layers. The increased frontal area of the intake due to the absence of precompression results in only a small mass penalty due to the gains from an efficient structure.

Extant or near term materials have been adopted for the SABRE/SKYLON design in order to minimise development risk, placing emphasis on advanced manufacturing techniques and novel structural concepts to achieve lightweight designs. To illustrate the design philosophy of the vehicle the following sections describe in some detail the fuselage construction and the undercarriage design. The fuselage is built up as a multilayer structure consisting of aeroshell, insulation, structure and tankage (Fig. 3).

3.1 Aeroshell

The aeroshell forms the outer surface of the aeroplane and therefore must withstand the local aerodynamic pressure loads and kinetic heating. The aeroshell is passively radiation cooled and during the ascent rises to a maximum of 855 K at the bottom of the dive. During reentry the temperature is kept down to 1100 K by dynamically controlling the trajectory via active feedback of measured skin temperatures. This is possible by virtue of the low ballistic coefficient and controllability of a lifting vehicle

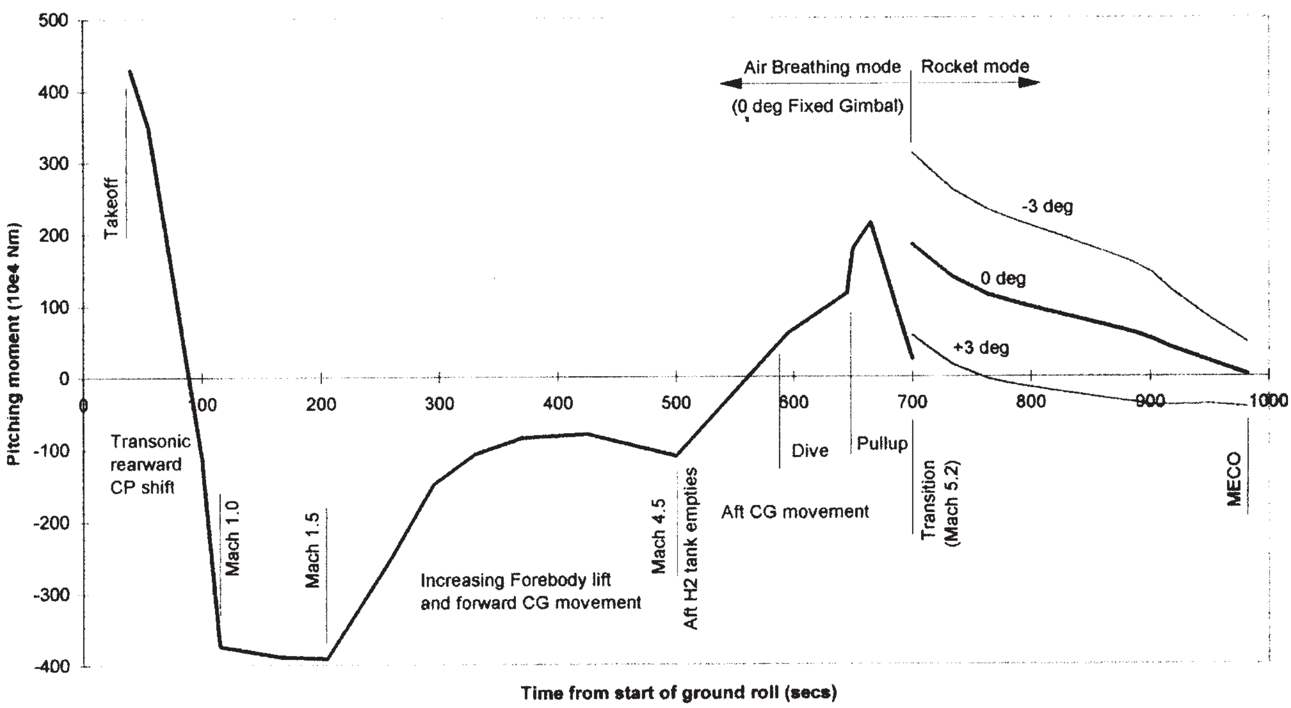

Fig. 2 Configuration C1 - untrimmed ascent pitching moment.

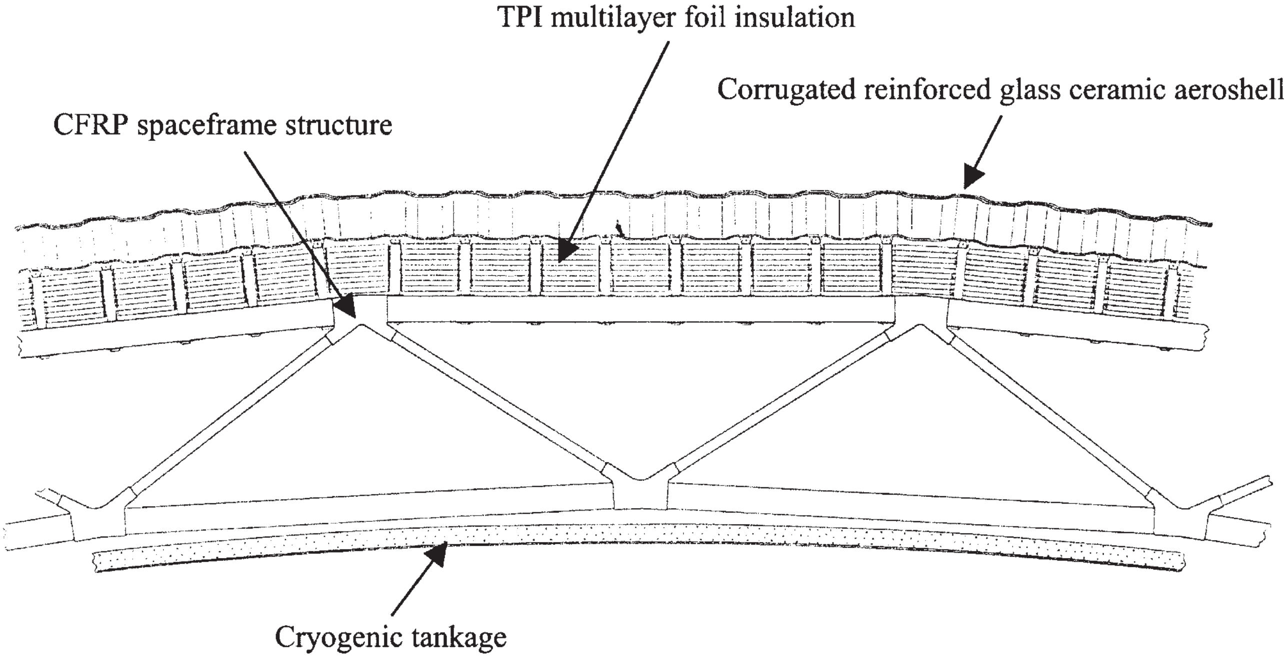

Fig. 3 Fuselage transverse section.

with active foreplanes. Finally the aeroshell must be airtight to prevent reentry gases leaking into and destroying the vehicle interior.

Typically, aeroshells proposed for spaceplane applications consist of two minimum gauge skins manufactured in reinforced ceramic separated by high temperature honeycomb to form a sandwich construction. However by referring to the SR-71 corrugated wing skins, an alternative structural concept emerged which is considerably lighter than the sandwich panel option. Since the pressure loading is relatively low (maximum of 0.1 bar), it is possible to eliminate one of the sandwich skins and the core material and arrange adequate structural stiffness by simply corrugating the remaining skin (retaining the minimum skin thickness of 0.5 mm). The corrugations give the aeroshell longitudinal bending stiffness and strength whilst allowing relative circumferential thermal expansion with respect to the ambient fuselage structure. The penalty for halving the aeroshell panel mass in this way is increased fuselage surface area which reflects in the skin friction drag. However this sacrifice is well worthwhile since the payload sensitivity greatly favours saving structural mass at the expense of a small increase in drag.

During reentry the aeroshell is approximately 800 ^{o}C hotter than the CFRP substructure from which it is mounted and consequently attempts to expand by 0.25%. However the corrugated sheet has a low spring stiffness in the circumferential direction since under a circumferential load the vertical (2 mm deep) webs twist, allowing the flats to bow. This property permits the aeroshell to be rigidly mounted off the substructure without the requirement for sliding seals. The aeroshell is riveted to circumferential hairpins every 0.3 m which are supported by radial posts attached to the fuselage ring frames (Fig. 3). The support pitch determines the fuselage frame spacing and hence represents a compromise with the corrugation depth. The hairpins (25 mm deep) are also corrugated in the circumferential direction (for thermal compliance) and accommodate the axial thermal expansion mismatch between the aeroshell and the substructure.

The aeroshell was initially designed in C/SiC. This material has the advantage of low density (2100 kg/m^3) and an upper temperature limit of 1800 K. However it is formed via a time consuming and energy intensive vapour deposition route. Following dialogue with the Atomic Energy Authority (Harwell) the baseline material has now been changed to their proprietary System2 SiC fibre reinforced glass ceramic. This material has the advantage of an oxidation resistant fibre and matrix (with matched thermal expansion characteristics α=3.10^{-6}) which is formed in a simple hot pressing operation. The reduction in process time and improvement in quality (much lower voidage and better process control) is expected to result in a hundredfold reduction in cost. The maximum use temperature of this material (1470 K) is somewhat less than C/SiC although still adequate for the aeroshell (maximum 1100 K). The disadvantage of employing System 2 is an increase in density (2620 kg/m^3) which, due to the minimum gauge restriction, reflects in a 25% increase in skin mass. However this penalty is considered acceptable given the advantages.

3.2 Fuselage Insulation System

The fuselage insulation system is layered consisting of a multifoil blanket (thermal protection insulation), an air gap and finally the tank foam insulation (Fig. 3). Each layer has a unique function. However their design has to be considered simultaneously in order to arrive at the correct system characteristics.

Reentry occurs at a relatively high altitude on SKYLON (typically 10 km higher than the Space Shuttle) due to the lower ballistic coefficient (mass per unit plan area). The air pressure inside the vehicle for the whole of this period is very low (i.e. less than 2 Pa), resulting in a relatively long molecular mean free path (greater than 20 mm) and consequently an effective air thermal conductance close to vacuum insulation. Therefore the most efficient TPI system for SKYLON is a reflective multifoil blanket (similar to spacecraft practice) with closely spaced foils (less than the mean free path) to prevent air conduction "shorting out" the blankets high resistance to radiation (Fig. 3). Following detailed analysis of the TPI, a design with 10 foils spaced 3mm apart was selected. The foils (10 μm thick with low emmisivity coatings) are manufactured in titanium since they are relatively hot, ranging from a maximum temperature of 640 K on the inner foil to 1070 K on the outer foil.

Following payload integration the vehicle is towed to the fuelling apron where automated fuelling services plug into the propellant system via the undercarriage wells. The propellants are loaded subcooled (H_2 at 16 K, O_2 at 80 K) since this enables the tanks to be sealed, thereby eliminating the safety hazard associated with venting propellant boiloff. By design, the fuselage insulation system is capable of storing the propellants for 2 hours before they have to be recycled back into the ground facility for reprocessing. This permits the vehicle to wait on the runway if necessary for an extra orbital pass of the target thereby improving the operational flexibility of the system. To meet the 2 hour ground hold requirement the maximum allowable heat leak is 100 W/m^2, representing a rise in the liquid hydrogen temperature from 16 K to 18 K (limited by boost pump NPSH limits).

Prior to propellant loading the internal spaces of the vehicle are purged and then filled with dry nitrogen at ambient pressure to prevent any hydrogen leakage creating an inflammable mixture. Due to the low temperatures and the relatively conductive gas between the foils, the radiation resistance of the TPI blanket is "shorted out" and it consequently behaves as a stagnant air gap. However, due to the chilling effect of the hydrogen tanks, significant natural convection currents are set up in the large air gap with convection rather than conduction as the dominant mode of heat transfer in this region. Therefore the ground hold insulation is effectively provided by the combination of the tank foam insulation and the air gaps in series. Apart from ensuring that the insulation system has the correct overall conductance, an additional requirement is to prevent nitrogen condensation within the airgap since this would represent a possible safety hazard and an increased heat transfer rate. Consequently the conductance of the foam and the airgap have to be correctly proportioned to ensure that the outer surface temperature of the foam is above the nitrogen dewpoint (approx 77.35 K). The final design consists of a 130 mm airgap which houses the fuselage structure with 10 mm of 60 kg/m^3 PVC foam glued to the tank wall.

3.3 Fuselage Structure

The fuselage can be regarded as a beam subject to aerodynamic and inertia loads along it's length. The highest loads are in the vertical pitch plane (notably during the 2g pullup manoever). However significant lateral loads exist due to the aft fin and during yawed flight, whilst torsional moments are generated by the fin and the nosewheel. In addition the aerodynamic crossflow component due to the vehicle incidence attempts to distort the fuselage cross section out of round. An unusual design difficulty, unique to spaceplanes, is that during reentry a temperature difference of over 1000 degrees exists over the radial gap between the tankage and the outer fuselage surface.

Three main structural possibilities exist, each of which has been proposed for various spaceplanes:

1) **Structural aeroshell (Hot structure)**

   Due to the relatively recent development of reinforced ceramics, the aeroshell could be transformed into the main fuselage structure by stiffening the skin with frames and stringers in a similar manner to conventional aluminium airliner construction. Apart from the practical difficulties of mounting the internal equipment and tankage off such a hot structure the main drawback of this approach is: a) stiffened skins are relatively inefficient for lightly loaded structures and b) the relatively low stiffness/density and strength/density of reinforced ceramics.

2) **Structural tankage (Cold structure)**

   The main propellant tankage could be pressurised to the point that it can handle the maximum fuselage bending moment without buckling (e.g. similar to

Atlas or Blue Streak). However since the bending moments on a horizontally launched flying vehicle are greater than a vertically launched ballistic rocket, the pressure required to stabilise the tankage (approx 2.5 bar) is much higher than the pressure required to satisfy pump NPSH limits (approximately 1 bar). Unfortunately the extra material needed to support the increased tank pressure is incorrectly disposed for an efficient beam and employed at a low stress level when resisting the applied bending moment. Also since the tankage does not extend the entire length of the fuselage, additional structure of a different type is required in the payload bay, nose and tailcone regions which is an extra complication. Finally the tankage must remain pressurised at all times otherwise the fuselage will collapse, characterising a rather delicate vehicle that is not easily handled on the ground.

3) **Separate ambient structure**

The remaining structural option is to introduce a separate tailored structure running down the length of the fuselage from which the tankage and the aeroshell are suspended. In reviewing the C1 structure it was realised that the novel fuselage insulation system had created an ambient radial airgap outside the tankage into which a cylindrical fuselage girder could be inserted. This enables the tank pressure to be reduced to the minimum governed by boost pump NPSH limits, and the aeroshell to be designed to carry the local pressure loading only. By suitable design this option is by far the best choice for SKYLON.

This type of structure must be capable of handling compressive loads over considerable distances without buckling. Since the vehicle has a relatively low area loading (due to the hydrogen fuel and 2g stressing limit) and the fuselage resembles a relatively deep beam, a very small amount of material is sufficient to support the loading. Therefore to satisfy elastic stability criteria whilst retaining high stress levels, the cylindrical girder must be locally condensed into small stable sections, frequently supported to satisfy global stability (i.e. a spaceframe). The spaceframe is composed of ring frames (which support the aeroshell on radial posts) spaced at 0.3 m intervals by shear diagonals. Four main longerons (roughly at the 45 degree points to maximise the section depth) carry the overall fuselage bending moment whilst avoiding cutouts and sudden changes in slope. The upper longerons form the payload bay door sill whilst the lower longerons crest the wing carry through in order to avoid a cutout for main undercarriage stowage.

The design of the structure is dominated by stiffness rather than strength considerations at both a global and a local scale. For example the forebody bending deflections are limited by the tank axial wall stress whilst the frame deflections are limited by the acceptable distortions of the fuselage cross section. Meanwhile at a local level the design of the individual spaceframe members is controlled by Euler instability and local wall buckling. Therefore to maximise the stress levels and hence minimise the structure mass, a material with the highest possible E/ρ is required which indicates CFRP as the correct choice since the structure is close to ambient. A spaceframe has the additional advantage that unidirectional CFRP can be specified for the members with further improvements in specific properties (due to the removal of the crossply strain limitation). Consequently the SKYLON structure has been designed in a high modulus CFRP with the following properties: E = 290 Gpa, ρ = 1600 kg/m^3, σ_{ult} = +/-1500 Mpa.

3.4 Tankage

Since SKYLON has an efficient fuselage structure there is no requirement for high tank pressures. Therefore the design tradeoff strongly favours low tank pressures plus tank mounted boost pumps. The low tank pressure (1 bar gauge for both oxygen and hydrogen tankage) permits ordinary welded aluminium tank technology without incurring an unacceptable mass penalty. Consequently all the tankage is designed in 2014-T6 material with typical wall sections varying from 0.3 mm to 0.6 mm. The tank foam insulation is stuck on the outside of the tank wall in order to capitalise on the improvement in the specific properties of aluminium at cryogenic temperatures. In the longer term stronger tank materials will give scope for mass reductions. For example titanium or aluminium/lithium are possibilities, although achieving a leak before burst condition in these materials may erode any theoretical advantage. Alternatively carbon fibre reinforced PEEK is a possibility. However the increased manufacture cost is expected to outweigh the practical mass savings achieved once the effects of minimum gauge and finite ply thickness have been taken into account on a graded thin wall structure. The tanks are axially located at their bases and allowed to expand or contract axially relative to the fuselage structure. Tangential ties perpendicular to the vehicle axis (picking up on the 4 longerons) react the tank wall shear load and thereby carry the tank weight whilst allowing the tank to expand and contract radially. The ties consist of twisted Kevlar 49 fibre wires due to the low conductivity, expansion and high strength of Kevlar at cryogenic temperatures.

3.5 Undercarriage Design

Conventional undercarriages for large jets weigh 3-4% of the takeoff mass, which would be unacceptable for an SSTO (9.6 tonnes at 275 tonne GTOW). For this reason HOTOL was designed to takeoff from a specially designed launch trolley in order to lighten the undercarriage for landings only. Early SKYLON designs retained the landing undercarriage but rejected the trolley in favour of a "Hotlift" (rocket boost) takeoff system. However review of the Hotlift system led to the conclusion that this method was operationally undesirable, expensive and represented 7 seconds of the ascent in which there were no abort options. In view of the indisputable operational simplicity of a rolling takeoff it was decided to review the engineering of undercarriage components in detail, particularly bearing in mind the unusual operational features of spaceplanes. Following this analysis it was found that a takeoff/landing capable undercarriage for SKYLON could be engineered for a mass of 1.5% of GTOW providing attention was paid to the tyres/wheels, oleos and the brakes.

The mass of wheels and tyres can be reduced on the condition that spaceports are built with stronger pavements than usual, thereby overriding current runway loading limitations. Consequently a highly loaded tightly spaced wheel assembly can be specified, consisting of 2 oleos with 4 wheels per oleo. The tyre width and diameter are determined by stowage requirements necessitating a tyre pressure of 26.5 bar (385 psi) which is similar to carrier based fighters. The high takeoff speed (155 m/s) is not considered unduly severe since unmodified Concorde tyres have been satisfactorily tested up to these speeds on a rolling road facility.

The oleo and bogie masses are determined by the landing impact/skidding case since the takeoff bending moments are small. Therefore providing the vehicle always lands empty of fuel a considerable mass saving results due to the 5:1 ratio between the takeoff/landing mass (airliners are normally designed to land at 80-90% of GTOW). In the event of an aborted mission, the vehicle will perform a slow powered circuit of the airfield whilst dumping fuel before joining the correct glideslope and shutting the engines down as normal.

An engine failure during the takeoff ground roll just before rotation represents an exceptionally severe braking problem since the vehicle has a kinetic energy of 3.24 x 10^9 Joules. Conventional carbon/carbon brakes would weigh over 4 tonnes based on allowable temperature rise. Having examined a number of options including deceleration rockets, trip wires and parachutes etc. it was concluded that by far the best solution was a water cooled braking system. The specific energy required to boil water off as saturated steam at 30 bar is 2.7 MJ/kg which compares favourably with carbon/carbon at approx 1 MJ/kg at ΔT = 1000 ºC. Therefore the brakes are sized with adequate thermal mass to cope with the landing condition (only 3.5% of the takeoff kinetic energy) which reduces the mass of the entire braking assembly to an estimated 415 kg. Meanwhile 1200 kg of cooling water is carried to handle a rejected takeoff, by blowdown through the brake via a one shot pyrotechnic pressurisation device. Assuming a successful takeoff the water is simply dumped overboard. Therefore the effective mass of the cooling system carried into orbit is approximately 100kg representing the tanks, plumbing, valves and pyrotechnic etc.

4. The SABRE Engine

The SABRE engine is designed to deliver a high airbreathing thrust/weight ratio with moderate specific impulse, whilst reverting to a high specific impulse rocket engine at transition. Since the airbreathing mode operates on a turbomachinery based cycle the engine is capable of generating static thrust (unlike ramjet cycles) and engine development can therefore take place on open testbed facilities. Optimum transition from airbreathing to rocket mode with this type of powerplant occurs at around Mach 5 and 26 km, after which the vehicle climbs steeply out of the atmosphere to minimise drag losses. The resulting ascent trajectory is relatively benign to both engine and airframe, leaving a reasonable choice of airframe materials capable of withstanding the ascent and reentry temperatures without active cooling. The SABRE engine is designed with state of the art technology for turbomachinery, pumps and combustion chambers etc. Current materials are specified for the engine machinery whilst the nacelle shell is manufactured in SiC reinforced glass and the bypass system in C/SiC.

By employing the rocket combustion chamber, nozzle and pumps in both modes the mass penalty of adding a separate airbreathing engine is reduced, whilst also eliminating the base drag penalty of the ‘dead’ rocket engine during airbreathing ascent. To generate reasonable thrust levels during airbreathing whilst preserving high nozzle area ratio the airflow must be pumped up to typical rocket chamber pressures (i.e. approximately 100-200 bar). The aim of the thermodynamic cycle therefore, is to provide this high pressure airstream with minimum fuel flow, assisted by the remarkable properties of liquid hydrogen (temperature and specific heat). By treating the engine as a ‘black box’ it is possible to show that the minimum fuel/air ratio to achieve a 150 bar air delivery is 0.0433, at which the combined entropy rise of the two streams is zero. However, practical cycles can only achieve a fuel/air ratio approximately twice this value partly due to their thermodynamics and partly due to component inefficiencies and finite temperature differences. In order to reduce the cycle power requirement and to achieve reasonable air compressor outlet temperatures it is necessary to cool the incoming airflow, particularly at high Mach numbers. However the airflow can also be viewed as a source of heat to drive a thermodynamic cycle operating between the high inlet air temperature and the low hydrogen stream 'sink' temperature. Then to allow the engine to operate over a range of speeds, the reduced incoming air enthalpy at low Mach number requires 'topping up' by combustion heat release in a preburner to ensure constant turbine inlet temperature. The resulting engines run at a nearly constant turbomachinery operating point and fuel flow over the whole airbreathing Mach number range.

The first attempt at designing an engine embodying the principles outlined above was the RB545 (HOTOL) engine. In this cycle the high pressure hydrogen flow was used to cool the airstream directly, following which the hydrogen stream split. Approximately one-third passed to the combustion chamber via the preburner whilst the remainder was expanded through the turbocompressor turbine prior to exhaust. This cycle variant has a higher fuel flow than the SABRE engine particularly at high Mach numbers due to precooler metal temperature limitations caused by hydrogen embrittlement. In addition the precooler frost control system was relatively crude, resulting in significant payload penalty.

4.1 Cycle Description

The SABRE engine has traded simplicity for a lower fuel flow by interposing a Brayton cycle power loop in between the 'hot' airstream and the 'cold' hydrogen stream (Fig. 4). The closed power loop requires a working fluid that remains gaseous at temperatures as low as 30K, restricting the choice to hydrogen or helium. Helium was chosen since:

- Helium's higher γ reduces the loop pressure ratio
- Helium's higher molecular weight results in lighter turbines with fewer stages
- Helium's inert nature permits a wider choice of precooler alloys.

Referring to fig. 4, the air enters the cycle via the air intake where it is cooled to low temperature prior to entry into the air compressor. Following compression the airflow is divided, part flowing to the main combustion chamber and the remainder passing to the fuel rich preburner. The hot gas from the preburner passes through heat exchanger HX3 to raise the helium outlet temperature from the precooler to a constant turbine entry temperature. The preburner gas then flows to the main combustion chamber to complete its combustion with the remaining air prior to expansion through the exhaust nozzle. From HX3 the helium is expanded through the main drive turbines to drive the air compressor, and

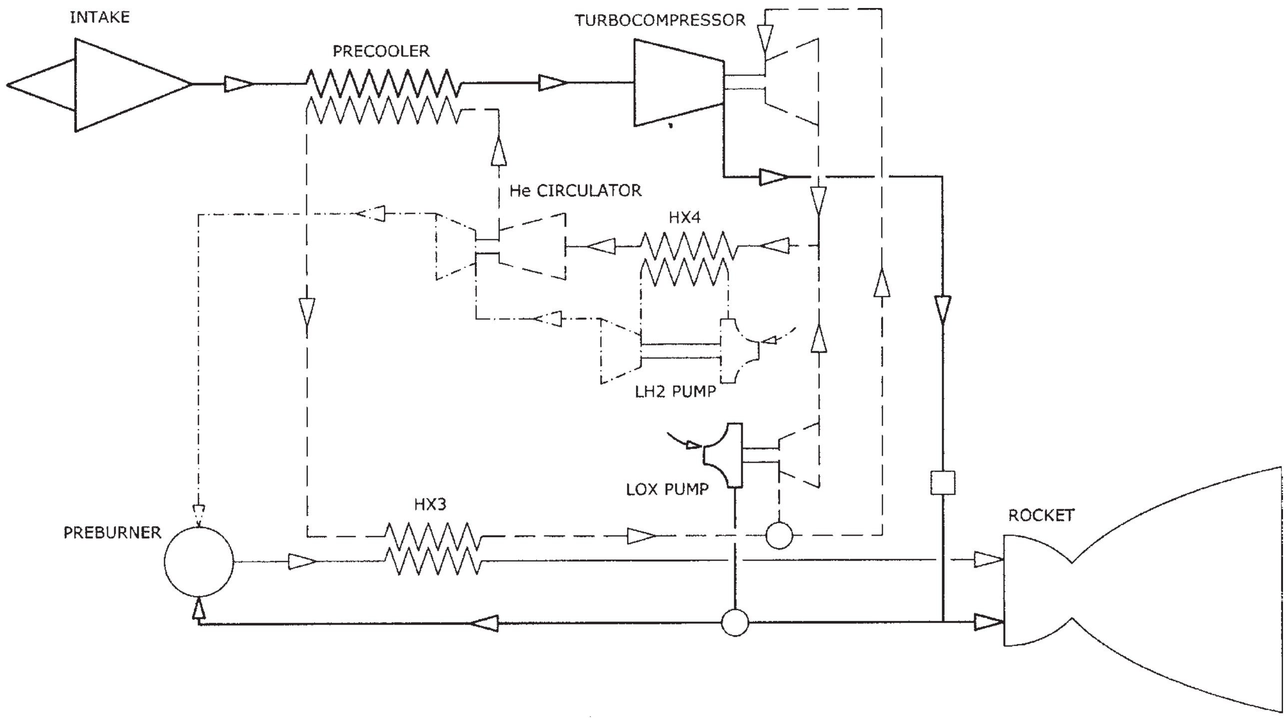

Fig. 4 Simplified SABRE cycle

then to HX4 where it is cooled by hydrogen delivered from the hydrogen pump. The cold helium then flows to the circulator and thence to the precooler to pass around the cycle once more. The warm hydrogen emerging from HX4 drives the hydrogen turbopump and then the helium circulator before passing into the preburner.

At transition the air inlet closes and the turbocompressor is rundown whilst simultaneously the liquid oxygen turbopump is runup. The preburner temperature is reduced in rocket mode reflecting the reduced power demand of the LOX turbopump. The high pressure liquid oxygen is evaporated in the main chamber cooling jacket in order to allow the same oxidiser injectors to be used in both modes.

4.2 SABRE Component Design

The majority of the SABRE engine components (e.g. combustion chamber, nozzle, pumps, turbocompressor) are relatively conventional. Lightweight high power heat exchangers however are a new feature peculiar to this type of engine and pose a challenging manufacturing problem.

The precooler is an efficient counterflow design, consisting of many thousand small bore thin wall tubes. The cold helium flows inside the tubes whilst the hot air is arranged in external crossflow. Due to the high pressure difference between the two fluid streams plain tubes give the lightest matrix design. External fins and turbulence stimulators do not prove mass effective despite the low airside heat transfer coefficient. The matrix is divided into two parts, the first high temperature matrix is manufactured in nickel alloy whilst the cooler second matrix is aluminium. The matrix dimensions result from an optimisation exercise whereby effectiveness and airside pressure drop are traded off against matrix mass. The resulting matrix has a large airside frontal area but is relatively shallow. Consequently the matrix is wrapped into a tapered cylinder in order to minimise the nacelle cross section.

4.3 SABRE Installation (Fig. 5)

To maximise the core engine thrust in airbreathing mode the engine is operated at constant chamber pressure, which results in nearly constant core airflow. The intake capture area is determined by the requirement to supply this airflow at the maximum airbreathing Mach number, notionally M5 at 26 km. The compressor flow capacity is determined by specifying that the engine should achieve full chamber pressure at Mach 0.6 sea level. This specification yields the required intake recovered pressure to achieve full chamber pressure along the trajectory (nearly constant at 1.3 bar). The intake pressure recovery schedule (PRF) must be carefully chosen in order to match the airframe drag characteristic to the engine demand. A relatively low PRF intake (1 oblique and 1 normal shock) is capable of supplying air at the correct pressure whilst following a trajectory that is close to the airframe minimum drag schedule. The simple shock structure enables a lightweight axisymmetric intake to be designed which results in efficient intake/engine integration. The intake face is positioned ahead of the wing in close to freestream conditions, thereby preventing wing or fuselage boundary layer ingestion and removing the possibility of wing shock interaction. The intake axis is drooped 7 degrees nosedown relative to the fuselage centreline to minimise the effects of the vehicle incidence. At transition the intake centrebody is translated forward whilst picking up 3 conical frustrums, thereby sealing the nacelle for the rocket ascent and reentry.

At low Mach numbers a fixed capture intake ingests more air than is required to feed the core engine. Spilling the excess air around the outside of the nacelle results in excessive drag and therefore the SABRE nacelle contains an internal bypass system. Above Mach

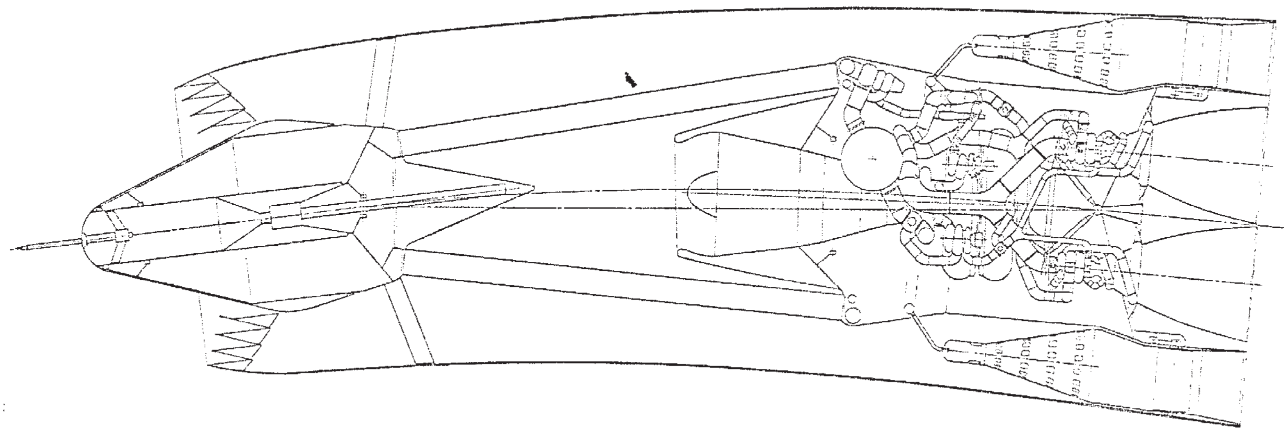

Fig. 5 SABRE nacelle vertical cross section.

TABLE 1: *Mass Estimates for SKYLON Configuration C1.*

Item			Mass (kg)
Main Engines		10870	
Nacelle, Inlet, Bypass		3922	
Wing		5115	
Fuselage: aeroshell, insulation, structure, payload bay		8130	
Main tankage, cryogenic insulation		2816	
Undercarriage		4170	
Aerodynamic control, hydraulics		2660	
Auxiliary systems, pressurants, coolants etc		5016	
	Basic Mass		42699
OMS/RCS propellant		2357	
Ascent Fuel		66807	
Ascent Oxidiser		150235	
Propellant margins and residuals		1282	
	Total fluids		220681
Mass margin			0
Payload			11620
	Gross takeoff mass		275000

2 the intake centrebody position is adjusted to focus the oblique shock on the intake lip thereby ensuring maximum capture with zero forespill. The captured airflow divides inside the nacelle with part flowing to the core engine whilst the remainder is diverted down an annular bypass duct. The bypass flow (and intake recovered pressure) are controlled by the variable throat area bypass nozzle. In order to minimise the momentum drag penalty for ingesting the bypass flow, it is necessary to maximise the exhaust velocity of the bypass system. Expanding the bypass flow at the intake recovered temperature would result in nett bypass drag due to the intake shock and internal flow losses. However since the core engine is overfuelled it is possible to warm the bypass stream by diverting hydrogen from the core engine to a bypass burner system without increasing the overall fuel flow. The bypass combustion temperature (and hence thrust) are maximised along the trajectory subject to nozzle choking and structural limits (1800 K). The performance of the bypass system can be further enhanced by operating the intake supercritically (swallowing the normal shock into the expanding subsonic diffuser) giving freedom to select a trajectory that maximises the overall nacelle nett thrust whilst leaving the core engine unaffected. This new trajectory dips below the original between Mach 2 and 5 resulting from a tradeoff between bypass thrust and intake cowl drag.

5. Conclusion

The SABRE/SKYLON combination is capable of placing a 12 tonne nominal payload into an equatorial low Earth orbit for a takeoff mass of only 275 tonnes. Mass estimates for SKYLON are given in Table 1. Structural mass estimates incorporate a contingency generally 15% which equates to approximately 5 tonnes of the total vehicle. This impressive performance has been achieved despite employing broadly today's technology and materials due to innovative design of the airframe and engine. It is anticipated that the user-friendly operational characteristics of the SKYLON vehicle and its low recurring costs will trigger a substantial expansion in the volume of traffic into space.

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