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MATHEMATICAL, PHYSICAL & ENGINEERING SCIENCES

The domain of the scramjet

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Phil. Trans. R. Soc. Lond. A 1999 **357**, 2317-2334
doi: 10.1098/rsta.1999.0433

THE ROYAL SOCIETY

The domain of the scramjet

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As flight speed increases into the hypersonic regime, the stagnation pressure and temperature inside the engine become so great that, for practical structures of acceptable mass, the flow must pass through the engine at supersonic, rather than subsonic, speeds, hence the term 'scramjet' (supersonic combustion ramjet). The classic application for the scramjet is to the long-range airliner offering extended hypersonic flight. It is also widely accepted as a possible complement to the rockets conventionally used for space launchers. This paper explores the topic of how scramjets may best be used and is the first Frederick S. Billig Lecture in Hypersonics of the International Society for Air Breathing Engines, to be given at the 14th ISABE Symposium, 5–12 September 1999 in Florence, Italy.

Keywords: scramjet; scramjet second stages; small SSTO aerospaceplanes; external combustion; cooled compression process; kerosene fuel

1. Introduction

Recent publications (Murthy & Curran 1991, 1996; Heiser & Pratt 1994) address the propulsion of vehicles up to scramjet speeds, the design of oxidizer collection systems to enhance that propulsion, and the design of scramjets and aircraft themselves (Nonweiler, this issue; Pike, this issue; Curran & Murthy 1999). This paper considers the scramjet from several aspects. In the near term and in the future, what is the scramjet likely to propel? What fuel should it burn? And what does it offer in mission design?

The first question was only partly answered during the 1980s. The design of huge, single-stage-to-orbit (SSTO) aerospaceplanes burning hydrogen, taking off horizontally and offering massive payloads, yielded to technical and cost pressures, and has led to vehicles such as Hyper-X. Insistence on hydrogen-fuelled engines and hydrogen-cooled structures has moderated to acknowledgement that at least some hydrocarbons will offer smaller vehicles, logistic simplicity and scramjet propulsion to flight Mach numbers of 10 or so. This combination would suit a small military aerospaceplane and would allow access to orbit or hypersonic cruise on a vehicle that is not obliged to carry a large payload. Discussion of such a vehicle has been publicly acknowledged for many years and is associated with the establishment of 'global presence'. It is shown in this paper that a small-payload aerospaceplane can achieve orbit as an SSTO in three different forms: (1) as a vehicle that burns only hydrogen; (2) as a vehicle that burns hydrocarbons as an airbreather and hydrogen for the subsequent rocket-powered acceleration to orbit; and (3) as a vehicle that burns hydrocarbons from take-off until orbit. In no case does take-off mass exceed that of large airliners such as the Boeing 777, and all three vehicles are shorter than the SR71, but the largest and logistically least convenient burns hydrogen.

If future aerospaceplanes and/or commercial space launchers are to be based on airbreathing propulsion and are propelled, for example, by RBCC engines, there are two obvious questions: (1) given the RBCC, is the scramjet a requisite cycle? (2) given the scramjet, is there a more appropriate use than as part of the RBCC? The first question is addressed elsewhere (Czysz 1992), but the second invites the answer that, if the scramjet is to be given its maximum chance as a space launch acceleration engine, it would be on the second stage of a two-stages-to-orbit (TSTO) launcher in which it would accelerate a relatively small vehicle, uncomplicated by other engine cycles needed for self-acceleration from take-off to Mach number 5 or 6. It is shown in this paper that for a given 'commercial' payload (i.e. many thousands of kilograms) and by comparison with hydrogen-burning vehicles having the same second stage launch mass and payload, the scramjet second stage is much smaller if it uses hydrocarbons for the scramjet (while retaining hydrogen for the rocket), and it may be realistic to use hydrocarbons because with optimized trajectories, the scramjet should not be used beyond Mach number 10 or 11.

In application to the small military aerospaceplane, a similar trend is evident, but here the TSTO may offer one particular advantage: since the TSTO second stage will be far smaller than the first stage or the equivalent SSTO, it should present the smallest radar image of the three, especially if components and configuration are shaped with stealth in mind.

Finally, the paper turns to a form of vehicle that uses airbreathing combustion in a uniquely different way. External combustion is shown to offer significant drag reductions for hypersonic aerofoils, the possibility of lift enhancement, and the promise of high specific impulse if appropriately applied. It is also an appropriate subject for this paper because some of the earliest experiments in this field were performed in the 1950s by Fred Billig at Johns Hopkins University (Billig 1993).

2. SSTOs

A horizontal take-off SSTO (using scramjets to high Mach numbers and hydrogen as fuel) becomes huge for many commercially reasonable payloads, and although take-off mass is reduced by LOX collection (i.e. the in-flight filling of vehicle tankage with oxygen gleaned from the atmosphere), Balepin (1996) shows the vehicle gets far bigger. For a total mass of 350 t at take-off, LOX collection *increases* vehicle length from 85 to 96 m. Scramjets can be validly included in such vehicles, but the size of both vehicle and scramjet would impose phenomenal levels of cost and technical risk upon the builders, not least because of the difficulty of adequate ground testing. Thus any near-term scramjet should propel a smaller vehicle.

The purpose of in-flight collection is to eliminate the massive load of LOX, which is otherwise carried from take-off. However, if LOX collection reduces vehicle take-off mass at the cost of increasing vehicle size, then airframe dry mass rises and absorbs some of the gains. It is in fact equally logical to question the choice of LH_2 fuel, which contributes very high calorific value (heat content per unit mass) and very good cooling capacity, but also imposes pressurized tankage of enormous bulk because the heat content per unit volume of hydrogen (even of LH_2) is extremely poor. A denser fuel might allow a smaller and lighter airframe.

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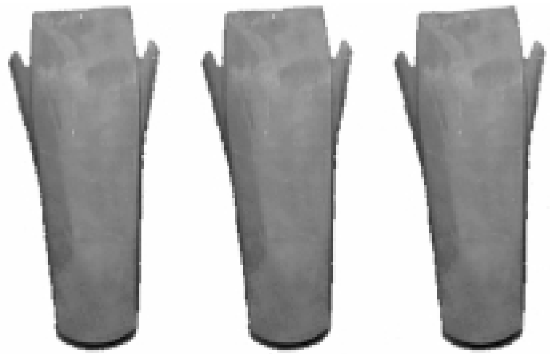

Figure 1. Echo 5.

Table 1. *SSTOs using kerosene or hydrogen airbreathers*

(Airbreathers are used up to $M_0 \approx 5.5$, LOX/LH_2 rockets are used to achieve orbit, and drag relief is provided by base burning at transonic speed.)

choice of fuel(s)	mass at take-off (Mg,t)	pull-up Mach number	mass to orbit (Mg,t)	payload mass (Mg,t)	vehicle length (m and (ft))
LH_2	275	5.5	63	*ca.* 12.5[a]	38.1 (125)
Ke + LH_2	275	5.5	50	*ca.* 12.5[a]	30.1 (98.7)

[a]Payload margin is *ca.* 1250 kg (1.25 Mg).

Many authors (e.g. Francis 1969; Martin 1987) have shown the value of using hydrocarbons for the initial stages of launcher acceleration and, if kerosene is chosen, the need for bulky tankage is much reduced. For example, consider a vehicle such as SKYLON, which takes off at 275 t and is 82 m long (and does not use an airbreather beyond $M_0 \approx 5.5$ or so). Vehicle length could be reduced (at some cost in complication but at the same take-off mass) by using an 'integrated' shape (and retaining LOX + LH_2), but an additional reduction in length might be achieved if kerosene replaces LH_2 for $M_0 \approx 0$–5.5, that is, if all airbreathing propulsion uses kerosene. This possibility has been studied as follows.

Since the commercially viable SSTO aerospaceplane is ambitious by any standards, it is realistic to say that successful design may depend on advanced materials and lightweight structures. In the study reported here, R. A. East (personal communication) gradually reduced the structure mass of an SSTO until a payload of *ca.* 12.5 t was obtained by an all-hydrogen-fuelled ECHO vehicle (see figure 1) weighing 275 t at take-off. For the structural technology level thereby indicated, East then assessed the ECHO vehicle which, for the same take-off mass and the same payload, used kerosene instead of hydrogen for the airbreather; this vehicle proved to be some 8 m shorter than the all-hydrogen original, a reduction of 21% on external dimensions (see table 1), except that wing area was increased in order to preserve performance at take-off.

For the commercial space launcher, the conclusions appear to be that the use of advanced structures may eventually offer a payload to take-off mass ratio of 4.5%, that the use of kerosene will still significantly reduce vehicle size, but that the rocket

Table 2. *Take-off mass and vehicle length for transport aircraft and SSTOs*

(Current Boeing plans include the Boeing 747-400X or 747-500. Its entry below would be 475 Mg, 85.4 (280), *ca.* 600 passengers (data based on Av. Wk and Sp. Tech. **150**(10), 8 March 1999).)

aircraft type	mass at take-off (Mg,t)	vehicle length (m and (ft))	comments on payloads
SSTO(KeAB + LH_2/LOX)	106.5	22 (72.2)	*ca.* 1 Mg (1000 kg)
SSTO(LH_2AB + LH_2/LOX)	119	29 (95.1)	*ca.* 1 Mg (1000 kg)
SSTO(KeAB + Ke/LOX)	294	26 (85.3)	*ca.* 1 Mg (1000 kg)
(APECS data due to R. A. East and J. Pike (personal communication).)			
Antonov 225	589	84 (275.6)	
Boeing 747-400	390	70.7 (231.8)	416–524 passengers
Lockheed C5B	373	75.5 (247.8)	
Airbus A340-600	359	75 (245.9)	380 passengers
Boeing 777-300	294	73.9 (242.3)	357–550 passengers
Concorde	182	62.1 (203.8)	100 passengers
Boeing 757-300	120	54.4 (178.6)	243 passengers
Boeing 727-200Adv.	86	46.7 (153.2)	145 passengers
(Data based on Av. Wk and Sp. Tech. **150**(2), 11 January 1999.)			

must still burn hydrogen. For this commercial vehicle, in fact, continued use of hydrogen fuel throughout the launch may be justified, but that in turn may depend on advanced technology having provided not only advanced structures but also oxidizer collection systems of greatly reduced volume.

For a large military aerospaceplane, the position is clearer: kerosene airbreathers hold out the prospect of vehicles with reduced radar signature, but logistically the large aerospaceplane would still be hampered by the need for LH_2 for the rocket. A small military SSTO, however, can usefully carry a light payload and this may enable kerosene alone to be used as fuel; this offers both the logistic appeal of avoiding LH_2 and the operational appeal of reduced radar signature.

(*a*) *The small-payload SSTO*

In late 1996, APECS conducted a study of how best to achieve orbit (with a payload of *ca.* 1 Mg) using an efficiently integrated aerodynamic shape, horizontal take-off, and airbreathing propulsion up to a Mach number to be determined, with rocket propulsion thereafter. The cases examined were (1) hydrogen fuel throughout the launch, (2) kerosene for the airbreather and hydrogen for the rocket, and (3) kerosene fuel throughout the launch (these are described in table 2 above as (1) LH_2AB + LH_2/LOX, (2) KeAB + LH_2/LOX and (3) KeAB + Ke/LOX).

Table 2 shows that the all-hydrogen SSTO is both longer and heavier than the vehicle that uses kerosene for the airbreather and LH_2/LOX for the rocket. The logistically convenient vehicle (defined as that which uses kerosene throughout) offers intermediate length but incurs a much higher take-off mass (294 Mg); however, its take-off mass is less than that of many current airliners. A summary of data is shown in figure 2.

Since it weighs less at take-off than recent versions of the Boeing 777, the Boeing 747, the Airbus A340 and the Lockheed C5, the small SSTO should be compatible

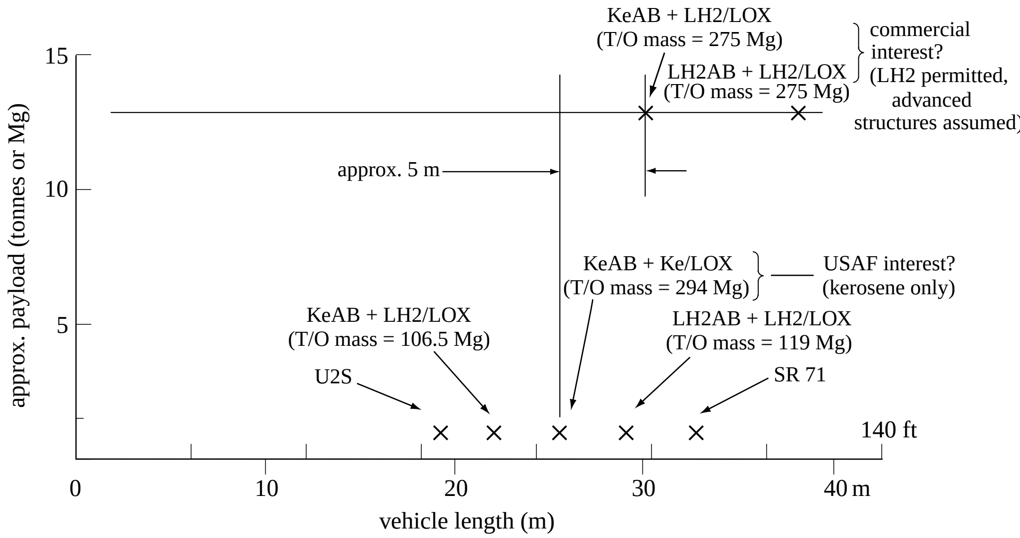

Figure 2. SSTOs large and small (data due to East and Pike).

with many existing runways (even if it uses kerosene throughout the mission and at take-off may require a trolley to reduce runway loading and the mass of undercarriage carried on the vehicle itself).

In all the SSTO calculations for tables 1 and 2, the value of pull-up Mach number was taken as 5.5, and a scramjet was not incorporated. Thus the small military SSTO might then be unable to cruise at Mach numbers exceeding 5.5, which is too low a value to outperform SR71 derivatives. To fulfil the cruise/reconnaissance mission that may be required on some flights, the airbreather must be able to accelerate to maybe twice the above flight Mach number and in consequence, the airbreather must be a scramjet, or an RBCC designed to operate at Mach numbers of up to 10 or more. It is therefore important that the scramjet, if incorporated to provide efficient cruise, should at least 'pay for itself' on space access missions, and this will probably raise the pull-up Mach number on space access missions towards 11 ± 1. Alternatively, reconnaissance might be curtailed if an RBCC is providing cruise propulsion (Czysz, this issue).

3. TSTOs

By the end of the 1980s, it had become clear that, for commercial payloads, the SSTO using scramjets for airbreathing acceleration (up to Mach number 15, say) was too risky an undertaking and offered payloads that could easily become zero due to mass growth in the airframe and elsewhere. If the scramjet was to have a chance to contribute to the orbiting of large payloads, it seemed more likely to succeed on the second stage of a TSTO, because

(i) this would allow the first stage to achieve a maximum Mach number of only 7 (or less);

(ii) the scramjet would be accommodated on a smaller and simpler vehicle than an SSTO, with only a rocket as additional propulsion; and

Table 3. *TSTO second stages*

(Stages use kerosene-burning ramjet/scramjets from launch to pull-up, and scaled RD 120 kerosene/LOX rocket(s) from pull-up to orbit (or kerosene-burning scramjets for cruise). Second stage launch occurs at Mach number 3, and pull-up Mach number implies the ability to cruise at Mach 10+.)

Mach number at launch	mass at launch (Mg,t)	pull-up Mach number	mass to orbit (Mg,t)	payload mass (Mg,t)	vehicle length (m and ft)
3	80	10	12.5	0.03[a]	18.3 (60)
3	90	10	13.8	0.80[a]	19.1 (62.6)
3	95	10	14.4	1.20[a]	19.4 (63.6)

[a]Payload margin taken to be zero.

(iii) the use of hydrocarbons for the scramjet might make the vehicle smaller still.

The scramjet second stage was considered by APECS and discussed with Billig in the late 1980s. A formal analysis was included in Heiser & Pratt (1994) and specific studies of scramjet second stages were published by Hardy *et al*. (1993) of Boeing and Koelle (1993) of DASA, but both Hardy and Koelle assumed hydrogen to be the fuel. Drawing on the work of Jamison, Austin, Hawkins and Lane (e.g. Jamison (1966) and Hawkins (1966), who were working in the late 1960s on behalf of Bristol-Siddeley Engines), APECS performed in 1994–95 a study of scramjet second stages burning a much denser fuel than hydrogen so as to secure a smaller volume of tankage, and a smaller, lighter vehicle.

For 'commercial' payloads and for a given second stage launch mass, comparison with scramjet second stages using hydrogen (or hydrogen plus neon as studied by Rudakov & Krjtchenko (1990)) confirmed that, at the same launch mass and launch Mach number, the kerosene-burning scramjet second stage offered much smaller vehicles. Unlike the hydrogen-burning scramjet second stage, the kerosene-burning vehicle grew smaller as pull-up Mach number was increased, and the usual problems of rapidly rising vehicle bulk were thereby avoided.

Nonetheless, even the kerosene-burning scramjet did not easily provide a payload significantly different to that of a Sanger-launched rocket-powered HORUS (Weingartner 1993). This was partly because the fuel tanks were now so much smaller that the vehicle needed to be repackaged further, so that additional reductions in size could be achieved thereby. On the other hand, these data suggested that the scramjet second stage could well offer both space access and hypersonic cruise on an even smaller aerospaceplane than the SSTO of § 2 *a* (table 2).

(*a*) *The small-payload second stage scramjet orbiter*

Results for three possible vehicles are given in table 3, for which it is assumed that packaging efficiency is good (i.e. the ratio of void to available volume is *ca.* 10%), and that both systems and structure demand only moderate technology levels. The vehicles thus depend on good engineering of available materials and structural techniques rather than on the application of potential properties as yet unachieved.

Clearly, the particular results of table 3 depend on using a scramjet which can operate (as a ramjet or as a scramjet) from flight Mach numbers of 3. If the transition

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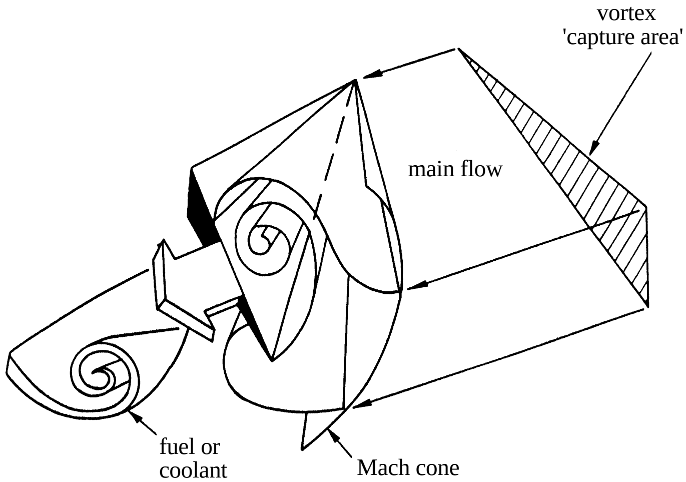


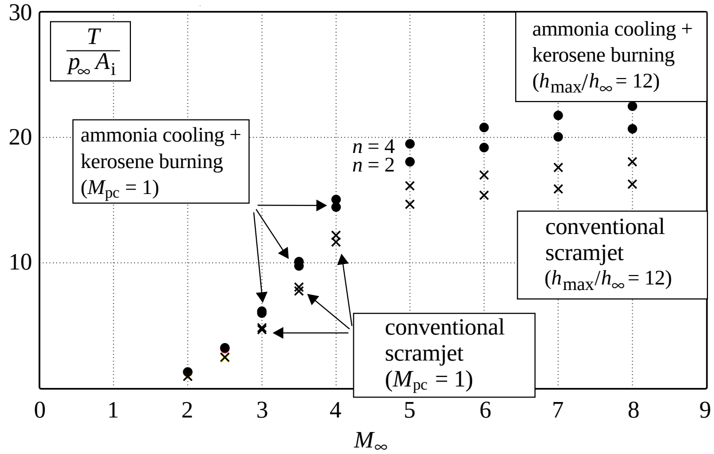


Figure 3. Scramjet specific thrust at design conditions (see Townend (1966)).

from subsonic to supersonic combustion is to be avoided, the use of an injected precoolant to chill the intake flow allows a scramjet to retain supersonic combustion at flight Mach numbers as low as 3 or 4, and at specific thrust levels which may still allow acceleration on scramjet power. These data are due to Nonweiler (his third paper, this issue), in whose work the precoolant was taken to be ammonia (see figure 3), but as Nonweiler pointed out, water injection could also be effective.

The mixing of ammonia into a supersonic airstream could possibly be achieved by the use of liquid jets and streamwise vorticity as was proposed (Townend 1966) for supersonic combustors in 1966 and subsequently developed by NASA.

The possibility of the ammonia coolant burning in the combustion chamber may offer some temporary economies in kerosene consumption, but its use as an airflow coolant may also ease the partial achievement of stealth (see § 3 *b*).

If neither the transition from subsonic to supersonic combustion nor airflow chilling is acceptable, the first stage launcher would need to achieve a higher Mach number, such as 6 or 7. At the cost of this more advanced first stage (Weingartner 1993), the scramjet would be simpler and the second stage smaller still.

(*b*) *Hypersonic stealth*

Hypersonic cruise vehicles are liable to surface temperatures that render the airframe easily detectable. Whereas, for an aircraft at Mach number 2, even the stagnation temperature will be below 500 K, a large vehicle cruising at $M_0 \approx 10$ will incur (with turbulent flow) temperatures of 900 K or more at the underside trailing edge, and the remaining undersurface will be hotter. If $M_0 = 10$, if the flow is turbulent and if radiation is taken as proportional to T^4, then at positions along a chordlength of 100 ft:

(x/c)	1	0.5	0.25	0.10	0.05	0
T (K)	1090	1110	1140	1180	1220	T_{LE}
T^4	14×10^{11}	15×10^{11}	17×10^{11}	19×10^{11}	22×10^{11}	T^4_{LE}.

All of these values of T^4 are much greater than the worst possible value for $M_0 = 2$ (i.e. $T^4 = 500^4 \approx 0.6 \times 10^{11}$), but the worst case for $M_0 = 10$ will depend on the leading-edge temperature T_{LE}, and that will depend on edge radius, cooling and sweep. For leading edges swept at 75° and having a nose radius of 2 mm (millimetres), Capey (see Townend 1978) shows that conductive cooling offers T_{LE} values between 1314 K and 1527 K, that is T^4_{LE} values of 30×10^{11} and 54×10^{11}.

Uncooled leading edges invariably provide local hotspots on hypersonic vehicles, especially if they are unswept. Either their temperature (and T^4) must be very high, or bluntness must be increased; in either case, the total radiation increases and the vehicle becomes more vulnerable to early detection and attack. There is therefore a case to examine hypersonic intakes that combine leading-edge sweep with leading-edge sharpness, and thus to study means by which to select the intake shape, to retain the requisite compression flow, and to cool the sharpened edge.

As far as basic geometry is concerned, all-swept intakes were designed in the 1960s (for example, by Mölder & Romeskie (1968) and Townend (1966)), and the application of conductive cooling was examined by Nonweiler (this issue) and Capey (see Townend 1978) giving leading-edge temperatures of *ca.* 750 K at $M_0 = 5$ (for 2 mm leading-edge radius and 75° of sweep) and 1300–1400 K at $M_0 = 10$ (again for 2 mm leading-edge radius and 75° of sweep). Typical intakes are as shown in figure 4, and the swept edges should not only run cooler but will reduce the radar signature offered to frontal attack. Variants of such intakes can be installed on modifications of the ECHO shape or on configurations of stealthier design.

For a given release of heat, intake size will vary as between kerosene and hydrogen. From data given by Küchemann & Weber (1968), observe that 1 kg of kerosene will release 10 200 kcal and 1 kg of H_2 will release 27 600 kcal, so to release 27 600 kcal will need (1) 1 kg of H_2 + at least 34.2 kg of air, or (2) 27 600/10 200 kg of kerosene + at least 14.8(27 600/10 200) kg of air, that is, the air intake must capture at least 34.2 kg of air for H_2, or 40 kg of air for kerosene. Thus the use of kerosene must reduce vehicle areas to, at most, 34/40 of those for the hydrogen-burning vehicle and vehicle length must fall to $\sqrt{(34/40)}$, that is, to 92% or lower. If it does not, the kerosene scramjet may be more easily detectable in terms of intake radiation and the flame itself will be more exposed in terms at least of flame area discernible from the front.

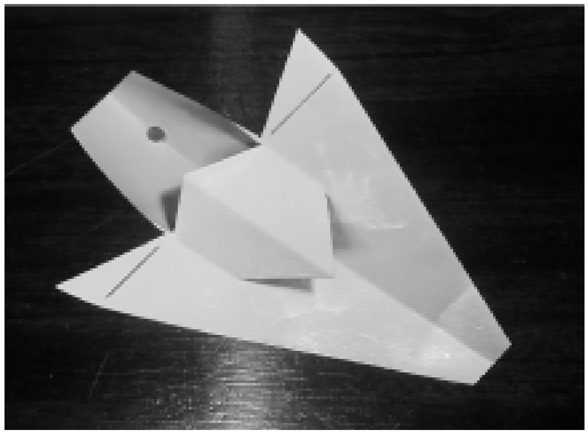


Figure 4. All-swept intakes (1966).

Luckily, the use of kerosene will frequently reduce the length of the vehicle by considerably more than 8% (see table 2 in which SSTO length is reduced from 29 m to 26 or 22 m, respective reductions of 10 and 24%). Also, where coolant injection is used to chill the intake flow at 'low' flight Mach numbers, its intermittent use in cruise could impair the chance of interception by heat-seeking missiles, since it would mask the forward radiation from the scramjet flame.

As far as detection of the jet is concerned and again, for a given release of heat, the kerosene flame temperature will be *ca.* 85% of that for hydrogen and its radiation some 0.85^4, about half of that for hydrogen. Cycle design for stealth may reintroduce the topic of propellant selection.

4. Flameriders

There are at least three well-established situations in which a hypersonic aircraft may suffer from a so-called thrust-minus-drag 'pinch', that is, not enough thrust or too much drag. The first occurs transonically where base drag on the nozzles (which are sized for high speeds) becomes a significant proportion of the total. The second occurs hypersonically where the thrust of a scramjet is falling with flight Mach number, and its capacity to accelerate a launch vehicle eventually falls short of requirements. The third occurs in lifting re-entry where a gliding vehicle may need more crossrange (and hence a higher lift-to-drag ratio) than can be provided at a vehicle attitude which permits a high enough lift coefficient (and thus sufficient altitude to avoid excessive heating of the structure). In all three cases, the problem can be met by the release of heat no longer inside an engine combustion chamber but externally to the airframe.

The first of the three problems noted above (transonic and supersonic base drag) has been addressed, for example, by Townend & Reid (1964) and by Billig & Schetz (1995), and is not considered here. The second and third problems are considered in §§ 4 *b*, *c*, but first of all, § 4 *a* presents some basic illustrations of external heat addition and in particular considers lift enhancement, with external heat addition treated as a variation in the design of high speed aerofoils.

Table 4. *Lift and drag at flight Mach number 6 (see also figure 5).*

		C_L	$(L/D)_p$	$(L/D)_v$
1	Corda	0.030	13.61	*ca.* 10 (say)
2	wedge flow	0.062	8.13	*ca.* 6 (say)
3	Pike	0.030	20.9	*ca.* 13 (say)
4	Pike	0.062	10.84	*ca.* 8 (say)
5	Broadbent	0.062	28.2	*ca.* 14 (say)
6	Broadbent	0.124	16.36	*ca.* 12 (say)
7	Broadbent	0.248	8.18	*ca.* 6 (say)

(*a*) *Lift enhancement*

As originally studied by Oswatitsch (1959), external heat addition was applied above and below a symmetric aerofoil and was shown to offer the prospect of external thrust at supersonic speeds. Subsequent theoretical work and various experiments demonstrated that external heat addition applied to one side of an aerofoil (see, for example, Billig 1993; Dorsch *et al*. 1959) would offer both a drag reduction (or a potential thrust) and lift enhancement. These experiments used pyrophoric fuels such as aluminium borohydride or triethyl aluminium, but Kallergis (see Quick 1968) demonstrated the external combustion of hydrogen and generated an external pressure thrust at $M_0 = 4$. Theoretical flow models were elaborated, for example, by Zierep (1966) and Baldwin (1959), but one of the more detailed analytic methods was that due to Broadbent (1976), who enabled the details of streamline shape, streamline pressure distributions and other properties throughout the flow to be predicted. In all cases, the flame was preceded by at least one oblique shock and was followed by a supersonic expansion, so that the three essential elements of the scramjet cycle were included; as a result, external combustion came to be classified as 'propulsion' and was criticized on the grounds of low specific thrust and low specific impulse. As a further result it was often reserved for applications such as control rather than propulsion but the fact is that, if correctly applied (as a variation in aerofoil design) a low thrust (or simply a reduction in drag) can be valuable and, if obtained by external heat addition, it can offer specific impulses that compete with scramjet values (a range of data is presented in § 4 *b*).

A conventional wing designed for $M_0 = 6$ (in fact, a case calculated by Corda (1990) for a conical flow waverider) gives the top line of data in table 4. In viscous flow the L/D might be 10 or so, but in practice, a wing could be designed for lower values of $(L/D)_p$ and $(L/D)_v$, and so for a higher C_L; see, for example, line 2 (which is for a 7° wedge). It is most unlikely that 'viscous optimization' would introduce significantly better performers; on the other hand (1) some complexity in lower surface shaping ('geometrical optimization' as achieved by Pike (this issue)) potentially offers an extra 30% in $(L/D)_p$ (see lines 3 and 4); and (2) the introduction of additional changes to the shape and of external heat addition below modified wings gives lines 5, 6 and 7.

The conclusions are that (1) for $(L/D)_p$ around 15 ± 1, external heat addition offers a quadrupling in C_L, and (2) for $(L/D)_p$ around 8.1 ± 0.1, external heat addition again offers a quadrupling in C_L. In more general terms, external heat addition may allow

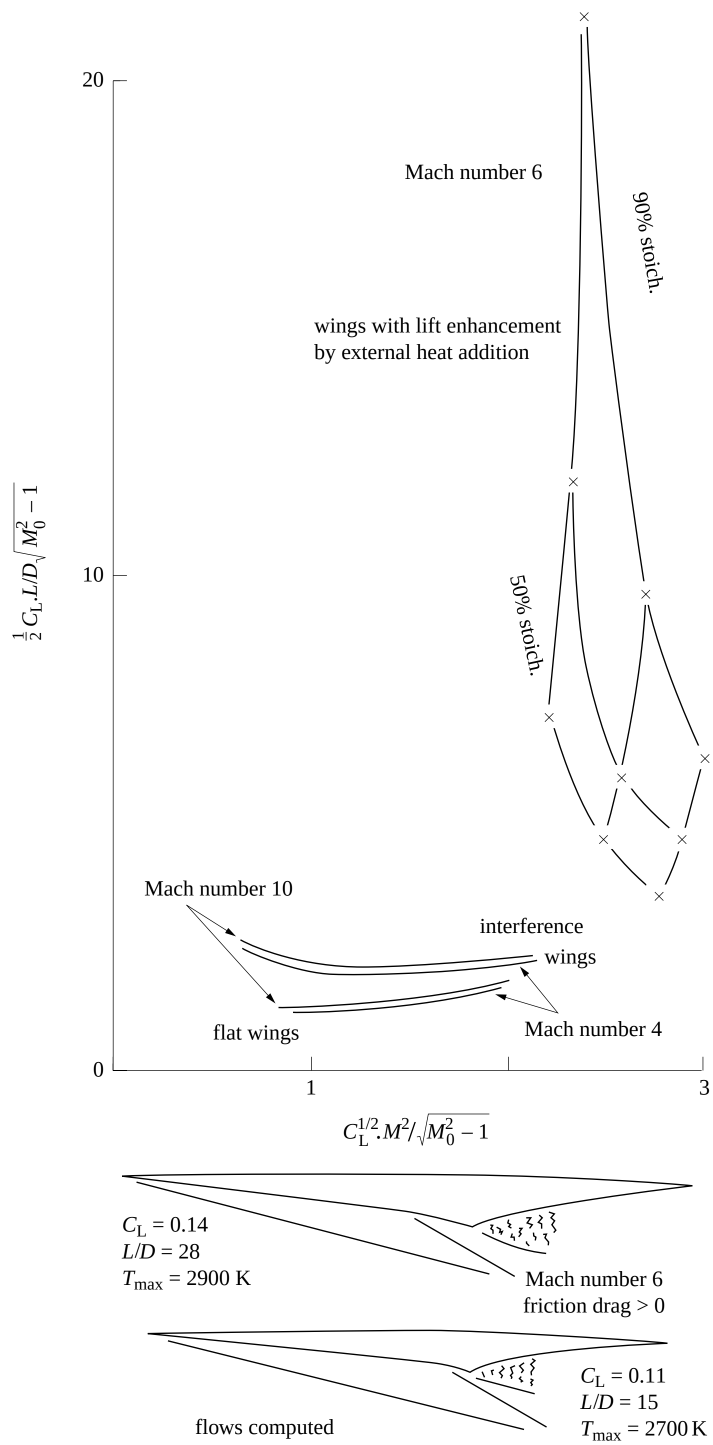


Figure 5. Wings with external heat addition.

PHILOSOPHICAL TRANSACTIONS OF THE ROYAL SOCIETY A MATHEMATICAL, PHYSICAL & ENGINEERING SCIENCES

Table 5. *External heat addition giving pressure thrust*

(Much higher SI-values result if a pressure drag is permissible (see Broadbent data, figure 6.)

M_0	C_L	$C_\mathrm{L}^{1/2}M_0^2/\surd(M_0^2-1)$	$\frac{1}{2}C_\mathrm{L}(L/D_\mathrm{p})\surd(M_0^2-1)$	SI (s)	$\Delta f/g$
6	0.125	2.151	−4.71	2608	0.25
9	0.064	2.298	−4.12	1832	0.18
12	0.04	2.412	−4.05	1403	0.13

(a) significant increases in wing loading (and possible reductions in structure mass);

(b) the use of direct lift control and substantial C_L-enhancement at constant angle of attack: thus a scramjet vehicle can gain altitude or pull a high-g turn without driving the scramjet intakes off-design by altering vehicle attitude, which could be useful for rapid adjustments to reconnaissance altitude, for other types of evasive manoeuvre or for first stage return; and

(c) useful performance without giving an external net thrust; in table 4, $(L/D)_\mathrm{p}$ is high, but it is not a negative number as it would become if pressure-drag had become a pressure-thrust.

(*b*) *Drag reduction and thrust*

Where lift and component planform areas are already selected (the wings perhaps by take-off or landing considerations, and the cowl by intake width and scramjet combustor length), there is a need to retain (rather than to enhance) the C_L, and to use external combustion to cut the drag. A conventional wing (or unmodified cowl) giving $C_\mathrm{L}^{1/2}M_0^2/\surd(M_0^2-1)$ equal to 3 would give a value of $\frac{1}{2}C_\mathrm{L}(L/D_\mathrm{p})\surd(M_0^2-1)$ no higher than 3 (as it happens) and thus a C_{D_p} of about 0.058. The C_L would be 0.243 (and the L/D_p would thus be about 4.2). A modified wing with external heat addition could still give a C_L of 0.243, but C_{D_p} would fall to 0.028, effectively by one-half.

Analysis shows that specific impulse is a strong function of the value of $C_\mathrm{L}\cdot L/D$ before and after the introduction of external heat addition. This parameter $(C_\mathrm{L}\cdot L/D)$ also arises in wing optimization theory (see Pike, this issue) and allows consideration of various cases whether external heat addition provides a drag reduction or a thrust; it eases the process of linking the disciplines of lift production and thrust production (or drag reduction), which are implicit in external heat addition and are essential to its evaluation. It is also a dominant term in $\Delta f/g$, where $\Delta f/g$ is a measure of the additional acceleration potentially achievable with external heat addition.

For stoichiometric external combustion of hydrogen, injected at a fuel Mach number of 2 and fuel temperature of *ca.* 950 K, pressure-drag can become pressure-thrust, as shown in further data due to Broadbent (see table 5) and specific impulse values are comparable with those of conventional scramjets (see figure 6).

The values of $\Delta f/g$, which measure the *gains* in acceleration due to external combustion, equate roughly to $\Delta f = 2.45\ \mathrm{m\,s^{-2}}$, $1.8\ \mathrm{m\,s^{-2}}$ and $1.3\ \mathrm{m\,s^{-2}}$, and they are based on a vehicle design in which external combustion is spread across the whole span of the underside.

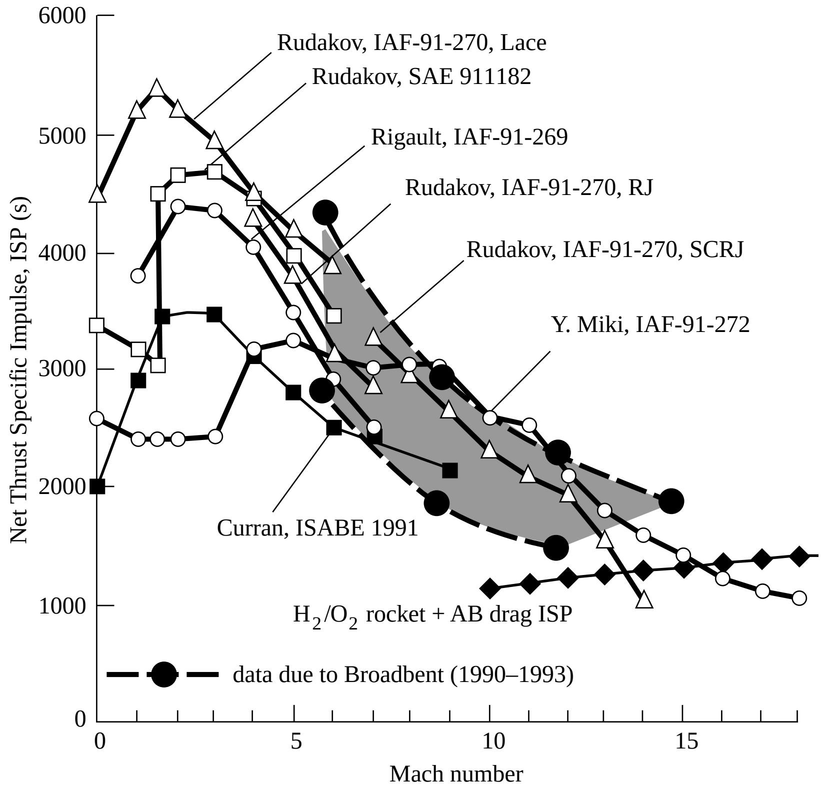
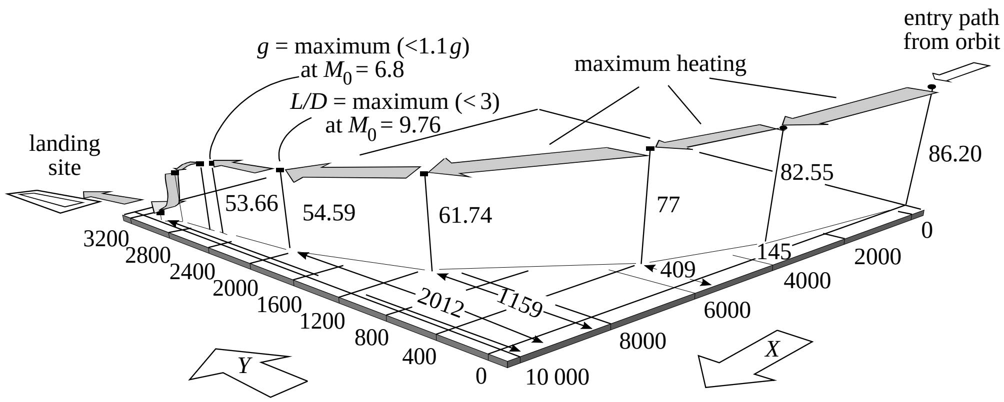


Figure 6. Specific impulses. Data correlation (Czysz) plus data points for external heat addition (Broadbent).

(*c*) *The drag-relieved re-entry craft*

For really high crossrange, a lifting re-entry glider needs to provide an L/D of perhaps 3.5 throughout re-entry (see Love 1964), but the associated C_{L} is low and the heating rates are greatly increased by the high kinetic pressures that result. Thus in practice, high C_{L} is called for during the heat pulse (which occurs at flight Mach numbers 20 ± 3, say) and L/D at these conditions will be less than 1.5. With this constraint, a trajectory can resemble that of figure 7, but the gliding crossrange is reduced to some 3000 km. In principle, external combustion could be introduced at flight Mach numbers 12 to 6 (say) so that L/D would increase, maximum deceleration would be delayed, crossrange would be greatly extended, and heating of the structure by the flame would occur well after the conventional heat pulse was past.

A typical re-entry trajectory at $L/D = 1.5$ would allow a glider to cover some 10 000 km (downrange) while decelerating to $M_\infty \approx 12$. In so doing the vehicle could also turn through some 90° in heading and, from $M_0 \approx 10$ (say), direct contributions to crossrange would accrue at *ca.* 800 km per unit L/D. Thus if external combustion were used at flight Mach numbers 10 down to 3 (say), to secure a value of L/D of 10, the total crossrange covered would be some 8000 km, plus the crossrange already provided during the hypersonic turn (*ca.* 1000–2000 km). It appears that with external combustion, global crossrange may be achievable even though airbreathing combus-



Typical SLEEC re-entry trajectory (downrange, 10 000 km; crossrange, 3000 km)

t (s)	altitude (km)	M_0	X (km)	Y (km)	L/D	g
0	86.20	28	0	0	1.19	0.205
450	82.55	25.13	3264	143	1.29	0.299
800	77	21.6	5543	409	1.35	0.531
1350	61.74	14.06	8335	1159	2.47	1.046
1700	54.59	9.76	9329	2012	2.96*	1.046
1900	53.66	6.83	9597	2490	1.72	1.085*
1950	50.90	6	9640	2587	1.86	1.077
2150	40.81	3.64	9721	2885	2.41	1.054
	0	0	10 000	3000		

*Maxima in L/D and g.

Figure 7. SLEEC22 re-entry.

tion would be needed only below $M_0 \approx 10$. This reasoning effectively combines some cruise and re-entry studies (Townend 1962, 1966, 1978; Cuadra & Arthur 1966) in the 1960s, with recent APECS assessments of a Space Station ambulance.

East (this issue) and Nonweiler (this issue) have reported on two designs of Space Station ambulance (see figure 8). Of these, the more innovative was SLEEC22 (see Nonweiler's figure 2, p. 2158, this issue), a well-streamlined glider having a low wing loading and a length of 9 m. The basic SLEEC22 trajectory shown in figure 7 would be ideal for a Space Station ambulance re-entry craft (because the deceleration nowhere exceeds 1.1g, whereas injured astronauts with broken limbs and internal bleeding would risk embolism at higher values of g). With external combustion added, both the crossrange and the re-entry 'window' would increase, and with them, the versatility of SLEEC. It is in fact possible that an enlarged SLEEC with external combustion might qualify as a military aerospaceplane that would be parked in orbit and offer a simplified version of the Lawrence Livermore Hypersoar concept.

5. The anti-ballistic missile (ABM)

A current concern is to intercept ballistic missiles having (say) 100–200 miles range but operating from mobile launchers and offering the ability to reach launch readiness

Figure 8. Space Station ambulance.

quite rapidly. Interception by missiles such as Patriot or Patriot PAC implies a 'last-ditch defence'. It would certainly be safer to complement Patriots with Boost Phase Interceptors, so that the ballistic missile could be destroyed within a minute or two of launch (say, within 30 miles of its launcher). This implies the need for an ABM of at least 175 miles range, but very short flight time and final speeds of Mach number 20 or so. If the ABM is to be of restricted launch mass, an all-rocket system will not provide sufficient range or final Mach number. A scramjet could be used for the Mach number range from 6 to some much higher value at which a rocket would again be used; this implies a minimum of three or four stages, of which the penultimate is scramjet powered. Thus, most of the trajectory is constrained by the need for moderate lateral accelerations, and for very high longitudinal accelerations.

APECS data due to Nonweiler show that some overlap between the launch rocket boost and the first few seconds of scramjet power would be helpful. With rocket launch, the ABM would reach Mach number 3 in perhaps 3 s, the scramjet would overlap for another 3 s (by which time longitudinal acceleration would be approaching $50g$) and the scramjet alone would then continue to accelerate the ABM (at $10g$ to $20g$) for another 15 s or so. At this point, some 30 miles from the ABM launch, the final rocket stage(s) would accelerate the warhead or projectile to Mach numbers exceeding 20 so as to intercept some 150 miles downrange from the ABM launch. The Patriots would wait for any missile that survived the BPI.

The production of a scramjet ABM for Boost Phase Interception would depend on compactness and thus the use of a hydrocarbon (such as kerosene) or of a solid fuel, but with either, airbreathing combustion would need to remain secure under $50g$ acceleration and the vehicle must survive the panel loadings due to kinetic pressures of many tons per square foot. The scramjet would lead an active life of less than 20 s, but it would not lack for excitement (Curran *et al*. 1996).

6. Conclusions

Since the work described has been exploratory rather than definitive, conclusions are tentative.

1. A large SSTO for commercial use will need not only advanced structures but careful optimization of propellants because, at given take-off mass and payload mass, hydrocarbons for the airbreather can offer smaller airframe size.

2. A small SSTO for military uses (cruise and space access) will benefit from exclusive use of hydrocarbons in both the airbreather and rocket, since take-off mass remains acceptable and size can be lower than for a hydrogen-fuelled vehicle.

3. An even smaller vehicle may be achievable by using hydrocarbon-fuelled scramjets on the second stage of a TSTO, subject to chemical constraints on hydrocarbon combustion.

4. Scramjet performance (especially specific thrust at supersonic and 'low' hypersonic Mach numbers) could be improved by the injection of evaporative coolants into the intake or the airflow upstream.

5. External combustion should be used as a means to achieve low-drag lift rather than to attempt significant net thrust. In other words, external combustion serves best, not as an engine, but as a modification to aerofoil design, and can then return specific impulses that rival scramjet values.

6. The most urgently needed hypersonic vehicle is a re-entry glider, serving as a low-g Space Station ambulance (SLEEC), but the addition of external combustion to an enlarged SLEEC offers a small, simple, space-based military aerospaceplane.

The author thanks the British Ministry of Technology, the United States Air Force, the McDonnell Douglas Corporation, Boeing Rocketdyne, the European Space Agency and the British National Space Centre for the use of data funded partly by these organizations. In addition, his thanks are due to the special contributions of Ted Broadbent, Robin East, Terry Nonweiler and Jack Pike, to Olga and Whitney Allan for preparing the text, to Ann and John Parker for generating the illustrations, and finally to the International Society for Air Breathing Engines for their invitation to give this, the First Frederick S. Billig Lecture in Florence, Italy, 1999.

Nomenclature

ABM	anti-ballistic missile (missile)
A_{i}	intake capture area
BPI	boost phase interception
D	drag
ECHO	European compact hypersonic orbiter

f	acceleration
Δf	gain in acceleration
g	acceleration due to gravity
h	specific enthalpy
ISP	specific impulse
Ke	kerosene
L	lift
L/D	lift-to-drag ratio
$(L/D)_{\mathrm{p}}$	L/D calculated from pressure forces
$(L/D)_{\mathrm{v}}$	L/D calculated from pressure lift and the sum of pressure and friction drags
LH_2	liquid hydrogen
LOX	liquid oxygen
M	Mach number (pull-up Mach number is the flight Mach number at which a vehicle starts the climb to orbit using rocket power rather than airbreathing propulsion)
M_{pc}	Mach number after combustion
n	number of shock waves in scramjet intake (figure 3)
p_{∞}	ambient pressure
RBCC	rocket-based combined cycle
SLEEC	slender lifting entry emergency craft
SSTO	single stage to orbit
t	time
T	thrust
TSTO	two stages to orbit

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