

4. Technology Challenges

A number of technologies are important for determining whether development of a TAV is feasible. Minimizing vehicle empty weight is highly desirable for any vehicle concept and critical for SSTO concepts. This will require the integration of lightweight composite materials into the vehicle airframe and subsystems. Rapid turnaround between missions, cost effective operation, and high payload mass fraction characteristics will also require development of lightweight, robust, and durable TPS materials.

Another key technology is propulsion. The vehicle's rocket engines will have to operate as efficiently as possible and provide the required delta-V to reach orbit. Rapid turnaround and low-cost operations also require that engines be durable, damage tolerant, easily inspectable, and capable of rapid and safe shutdown.

If the orbital vehicle is air launched or aerial refueled it becomes desirable (and perhaps necessary) to have highly throttleable engines and to maximize propellant density. With the use of high-density propellants, the size of the orbital vehicle and its empty weight can be minimized, permitting air launch using existing transport aircraft or horizontal launch from existing runways and aerial refueling with existing tanker aircraft.

In this section, we explore these technology issues by reviewing the four technology areas of primary importance for the development and design of TAVs: propulsion, materials and structures, thermal protection systems, and systems integration. This review is based upon discussions that took place at the RAND TAV workshop, on recent technical publications, and on material made available to RAND by aerospace contractors.

Propulsion

For propulsion systems, high efficiency (i.e., high specific impulse) and high thrust-to-weight (T/W) are primary performance goals. Cryogenic LOX-LH₂ propulsion provides a specific impulse in the vicinity of 460 seconds. Higher efficiency may possibly be gained by adding high energy density materials, although the LOX-LH₂ propellant combination is the most efficient one used in existing operational launch systems. An additional advantage of LOX-LH₂ is

that the exhaust product is water, which is environmentally attractive for routine operations.

A related consideration is the handling characteristics of TAV propellants. Cryogenic fuels are more energetic, but introduce fuel handling and storage problems that may limit the number of launch bases a vehicle can operate from. The use of cryogenics requires appropriate logistics support to be in place, including the ability to store and process LH2 at very low temperatures. The complications involved in handling cryogenic fuels may affect TAV responsiveness and limit turnaround time even when in a quick-reaction or alert status mode. The most difficult problem concerning cryogenic propellants is considered to be handling LH2 in an operational environment. Hydrogen is difficult to contain, and leaks pose an explosion hazard. Another design complication arises from the very low boiling point of liquid hydrogen (20° K). This introduces material compatibility issues for both the vehicle LH2 tank and the pipes that deliver liquid hydrogen to turbopumps or rocket engines, as some metals lose strength and become brittle at such low temperatures.

On the other hand, past DoD research on the handling of cryogenic fuels demonstrated quick-reaction fuel-loading times on the order of an hour to a few hours. It can be argued that a two-hour LH2 loading time may not introduce too great a time delay for TAV military missions. In addition, extensive experience has been gained in handling cryogenic propellants for expendable rocket systems.

Nevertheless, the use of LH2 presents additional design challenges for some military TAV concepts. If liquid hydrogen is used, propellant tank and overall vehicle size may be significantly larger than they would be if high-density propellants were used. It may be more difficult to design and build an air-launched LH2 powered TAV. Consequently, it is important to keep the trade space open between cryogenic and noncryogenic fuels, especially for air-launched concepts (methane and liquid oxygen may be an inexpensive propellant combination well suited for air-launched systems).

The maturity of proposed rocket engines is also an important design consideration for TAVs if up-front development costs are to be minimized.

Rocket engines can be classified as conventional bell-nozzle rocket engines or as aerospike engines. We next review their capabilities and a few representative designs.

Conventional Rocket Engines

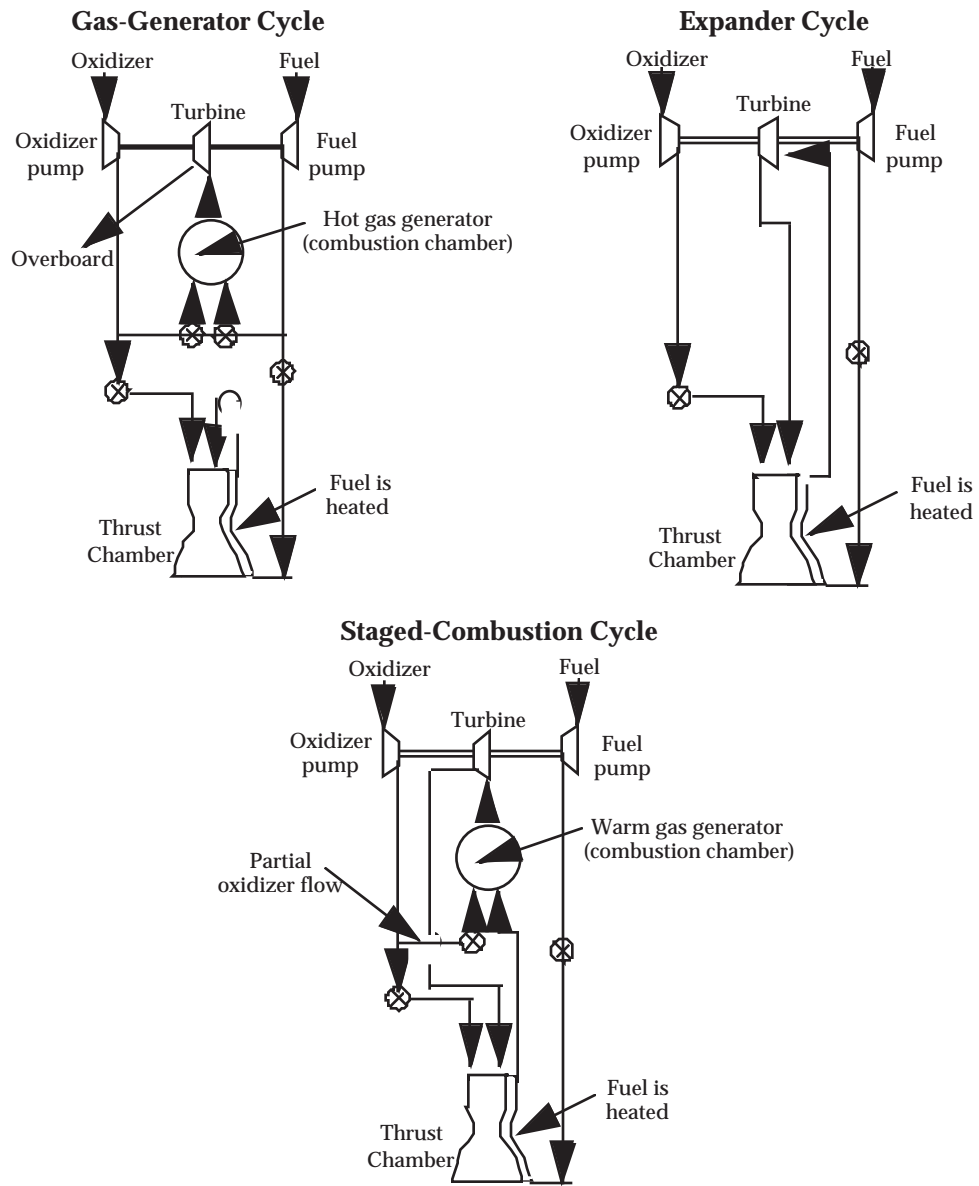
There are a number of conventional rocket engines that are either off-the-shelf products or that could be used to propel TSTO TAVs. Several highly capable conventional bell-nozzle engines are currently in production but would require modification for use in an SSTO vehicle (specifically, T/W values for these engines would have to be increased; see below). An example of such an engine is the Space Shuttle Main Engine (SSME), which Rockwell and McDonnell Douglas proposed to use in their X-33 designs.

Highly responsive TAVs will require engines that can be maintained easily and that have high reliability. The SSME, while mature, is not designed for the routine maintenance and rapid turnaround of TAVs.

Conventional high-performance LOX/LH₂ engines such as the SSME utilize a staged-combustion cycle. Figure 4.1 reveals schematically the differences in rocket combustion cycles. In a staged-combustion cycle, either the fuel or oxidizer is channeled completely through a “precombustion chamber,” where it is mixed with a portion of its counterpart to run the turbopumps. The products are then channeled to the thrust chamber, where they are joined with the rest of the fuel/oxidizer counterpart to complete the combustion process.

The gas-generator cycle, which is used in the aerospike engine (see the next subsection), channels part of the fuel and oxidizer to a generator that runs the turbopumps and then expels the products of combustion. The remainder of the fuel and oxidizer is channeled straight to the main thrust chamber. In the staged-combustion cycle, therefore, all of the fuel and oxidizer are used as the propellant, whereas the gas generator cycle “wastes” some of the propellant mass by expelling it overboard.

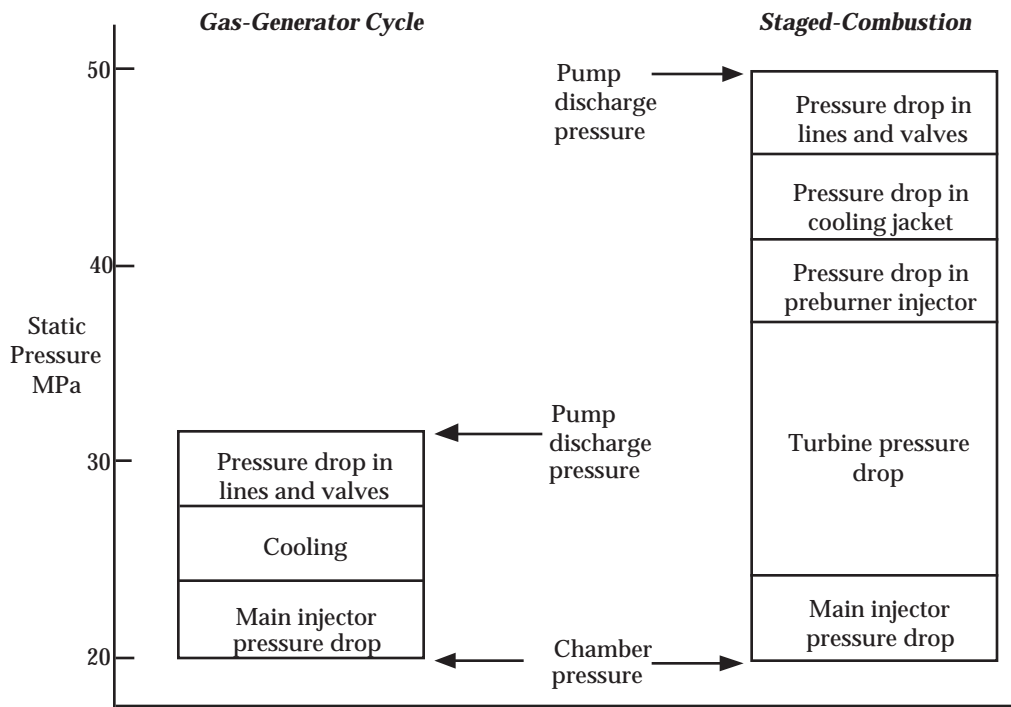
However, since the staged-combustion cycle’s precombustion chamber is essentially connected in series with the main thrust chamber, the pump pressure necessary to maintain a given thrust chamber pressure is higher than the pump pressure in a gas-generator chamber, where the precombustion chamber and the main thrust chamber are linked in parallel (see Figure 4.2). Therefore, staged-combustion engines generally exhibit higher Isp values than similar gas-generator engines (2–5 percent), but durability is lower because of higher pump pressures and reliability is lower because of increased complexity.



Source: R. Humble, G. Henry, and W. Larson, *Space Propulsion Analysis and Design*, McGraw Hill Inc., 1995.

Figure 4.1—Rocket Engine Combustion Cycles

The performance characteristics of two staged-combustion engines, the SSME and the Russian RD-120 engine, are listed in Table 4.1. Note the high chamber pressures in both cases.



Source: R. Humble, G. Henry, and W. Larson, *Space Propulsion Analysis and Design*, McGraw Hill Inc., 1995.

Figure 4.2—Comparison of Pressure Levels for Gas-Generator and Staged-Combustion Cycles

Table 4.1
Comparative Performance of LOX/LH₂ Engines

	SSME Block II (1)	RD-0120 (1)	Aerojet/Lyulka D-57
Sea-level thrust (lb)	395,000	330,000	NA
Vacuum thrust (lb)	470,000	441,000	89,600
Isp (sec) in vacuum	453	455	456.5
Chamber pressure	3200	3170	1603
Mixture ratio	6.0	6.0	5.8
Area ratio	77.5	85.7	143
T/W	51	43	48
Propellants	LOX/LH ₂	LOX/LH ₂	LOX/LH ₂
State of development	in production	in production	in production

SOURCES: (1) National Research Council, *Reusable Launch Vehicle Technology Development and Test Program*, Washington, D.C., 1995; (2) J. D. Elvin, "X-33 Ascent and Reentry Trajectory Analysis," Lockheed Martin Skunk Works interdepartmental communication; (3) A. Wilson (ed.), *Jane's Space Directory*, 11th Edition, UK, 1995.

The SSME has proven to have excellent performance and reliability on space shuttle missions. However, it requires extensive turnaround time for inspection and refurbishment of components, and its thrust-to-weight ratio is inadequate for SSTO missions. The National Research Council reported that an SSTO RLV would require a T/W value between 75 and 80. Programs to upgrade the shuttle are projected to improve the T/W value to 58 for the Block II+ version and near 70 for the Block III version. New turbopumps, heat exchangers, valves, and combustion chamber would enable the projected performance enhancement, as well as reduce the turnaround time required for maintenance. The current SSME must be pulled between flights to replace the turbopumps. The Block II version is projected to be launched 10 times before the engine must be pulled. As it is, however, the performance of the SSME is adequate for the TSTO missions of the space shuttle. Therefore, it might not be necessary for a TAV designed for TSTO to have engines with T/W values as high as 75-80. Improvements in reliability alone may be adequate, then, to allow the SSME to meet the needs of a TSTO TAV.

The Russian RD-0120 has similar performance to the SSME. It is desired for its deep throttling capability (25-114 percent, compared to 65-104 percent for the SSME). However, it has an even lower T/W. Improvements similar to those for the SSME are planned for upgraded versions of the RD-0120.

As shown in Table 4.1, the Aerojet/Lyulka (Russia) D-57 engine, proposed for use in the Northrop Grumman TAV concept, has an Isp and a T/W that are comparable with the two larger engines discussed above. Because the D-57 is a much smaller engine, it produces less thrust, but it also weighs much less, even though it is based on late Soviet 1960s materials technology (this engine was originally developed for the Soviet lunar program).

Depending on the design of the TAV and its ascent trajectory, the high thrust capacity of engines targeted toward SSTO missions may not be necessary for TSTO missions. If this is the case, smaller, simpler propulsion systems may provide adequate performance for such vehicle designs. In his TAV proposal, Ken Hampsten of Phillips Laboratory suggested two existing LOX/kerosene engines for the propulsion system of an air-launched TAV. Although these engines have lower performance than LOX/LH2 engines, they do not necessitate the high fuel storage volumes and complex cryogenics of LH2 storage tanks and feed lines. The use of LOX/kerosene engines results in a relatively simple, easy-to-maintain propulsion system.

The performance characteristics of the two LOX/kerosene engines suggested are shown in Table 4.2.

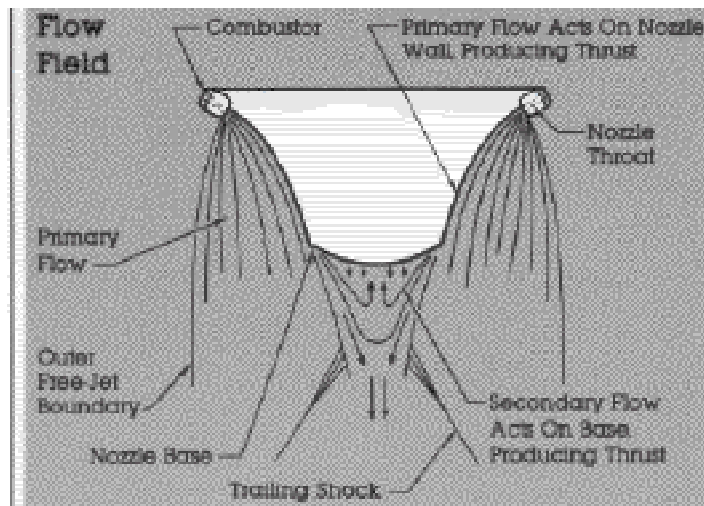
Table 4.2
Comparative Performance of LOX/Kerosene Engines

	Aerojet-TRUD NK-31	P & W D-58N
Sea-level thrust (lb)	NA	NA
Vacuum thrust (lb)	90373	18700
Isp (sec) in vacuum	355	353
Chamber pressure	1334	1125
Mixture ratio, engine	NA	NA
Mixture ratio	2.6	2.6
Area ratio	114	184
T/W (calculated)	59	27
Engine cycle	staged combustion	staged combustion
Propellants	LO ₂ /kerosene	LO ₂ /kerosene
State of development	in production	in production

SOURCE: Ken Hampsten, "An Air-Launched, Highly Responsive Military TAV, Based on Existing Aerospace Systems," Briefing at RAND, January 22, 1996.

Linear Aerospike Engines

The linear aerospike engine has been the subject of research for many years. This type of engine has many potential advantages and is now planned as the main propulsion system for the X-33 technology demonstrator. The X-33 aerospike engine configuration is composed of small, side-by-side combustion chambers (thrusters) that exhaust onto a common exterior surface (nozzle), in contrast to the single combustion chamber exhausting into a bell-shaped nozzle found in traditional rocket engines. The general design and exhaust flow field for an aerospike engine are shown in Figure 4.3.



SOURCE: Aerojet Corp., *Aerospike*, Brochure, Pub.no., 671-M-87.6, 7 November 1987

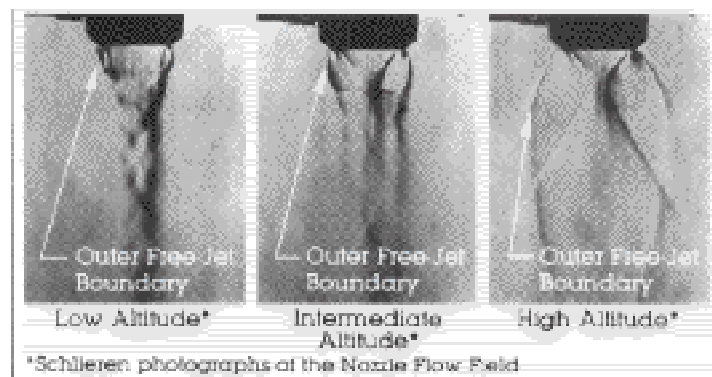
Figure 4.3—Aerospike Engine Nozzle Operation

Exhaust gas expands against the nozzle on one side and into the atmosphere on the other side. The expansion characteristics of the propellant flowfield are influenced by the ambient pressure, since the outer surface of the flow is a free jet boundary.

At high altitudes, the free jet boundary expands to the Prandtl-Meyer turning angle. At low altitudes, the high ambient pressure compresses the primary flowfield, increasing the static pressure on the nozzle wall, as shown in Figure 4.4. Thus, the nozzle wall may be designed with a high area ratio for optimum high-altitude (or vacuum condition) performance without experiencing significantly decreased performance at low altitudes (high pressure condition). In contrast, a bell-shaped nozzle designed for high I_{sp} in a vacuum would have an overexpanded jet, and relatively poor performance, at low altitudes.

There are some potential drawbacks to the aerospike engine. The aerospike engine has been found to exhibit a thrust loss through the transonic regime. Nevertheless, overall aerospike engine performance is superior to that of a conventional bell nozzle equipped rocket engine. There are also some concerns regarding vehicle stability during the ascent flight regime at max Q or when the vehicle experiences maximum dynamic pressure. The thrust vectoring control scheme (see discussion below) will have to be designed to accommodate stressing reaction control dynamics during this period.

Performance characteristics of the J2S aerospike engine planned for the X-33 are included in Table 4.3. It will have a relatively low T/W of 35 because relatively old gas generators (composed of older materials) will be used from the Saturn J-2 upper stage (from the Apollo program). Note, however, the low chamber



SOURCE: Aerojet Corp., *Aerospike*, Brochure, Pub.no., 671-M-87.6, 7 November 1987

Figure 4.4—Aerospike Engine Flow Fields

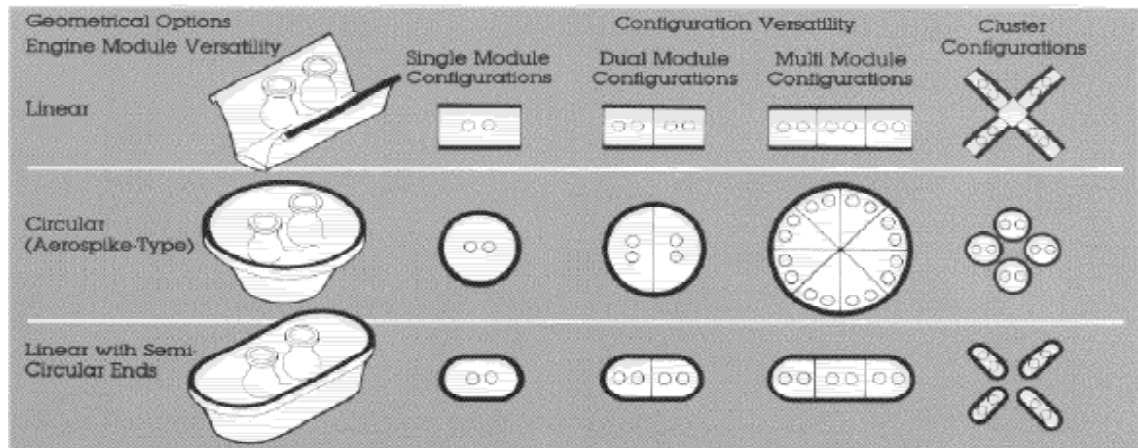
Table 4.3
J2S Aerospike Engine Performance

Sea-level thrust (lb)	202,480
Vacuum thrust (lb)	270,045
Chamber pressure (psia)	883
T/W	35
Mixture ratio, engine	5.5
Mixture ratio	5.84
Area ratio	87
Propellants	LOX/LH2
Total propellant flow rate (lbm/sec)	608.71

SOURCE: J. D. Elvin, "X-33 Ascent and Reentry Trajectory Analysis," Lockheed Martin Skunk Works, interdepartmental communication.

pressure of 883 psia planned for the X-33. The full-scale aerospike engine planned for the full-scale Lockheed-Martin RLV, the RS-2200, would have a higher chamber pressure of 2,250 psia, still significantly less than that of the SSME.

Aerospike engines are designed to be modular. Several modular units can be arranged in a specific configuration to form a propulsion system that can accommodate the shape of the vehicle and therefore minimize drag and optimize other aerodynamic performance characteristics. Several modular aerospike engine configurations are shown in Figure 4.5. On the X-33, the aerospike engine has two to three modules located side-by-side, each module having thrusters on opposite sides of a central truncated spike. In addition to fitting the rear profile



SOURCE: Aerojet Corp., *Aerospike*, Brochure, Pub.no., 671-M-87.6, 7 November 1987

Figure 4.5—Aerospike Engine Configurations

of the vehicle, this configuration allows thrust vectoring by throttling individual thrusters in the different modules, eliminating the need for mechanical gimbals and all the components associated with traditional thrust vectoring.

Although the concept of the linear aerospike engine dates back to the 1960s, the engine has never been flown and no flight-weight engines have been tested. Lockheed Martin plans to begin flight testing of the engine in October 1997, using an SR-71 as a supersonic testbed. The results of this test series will be correlated with ongoing wind tunnel testing at various facilities to verify analytical predictions.

Hydrogen Peroxide (H₂O₂) Propulsion

As mentioned in Section 2, hydrogen peroxide (H₂O₂) has been proposed as either an oxidizer or a monopropellant for TAVs. It has the advantage of being noncryogenic and having a relatively high density. It is not, however, without drawbacks. H₂O₂ presents a storage problem because it may not be stable in some situations, especially if contaminated. It can slowly decompose and evolve oxygen gas. At high temperature (e.g., if heat transfer in flight occurs), contamination can lead to an explosion hazard. It has been argued that contamination could occur during in-flight transfer.¹

The high concentration of H₂O₂ required for propellants makes the substance very different from peroxide found in the household. The concentrated form can burn skin and poses a fire hazard to many organic materials. The vapors could pose a problem if there were faulty design, maintenance, or operations.

Nevertheless, its high density may make it an attractive propellant choice for air-launched TAV concepts, if it can be used safely. Unfortunately, no H₂O₂ rocket engines are available today. A new engine development program would be required.

A more attractive high-density propellant combination for air-launched TAV concepts may be LOX and methane, because propellant handling and storage may be easier on board the aircraft. Unfortunately, no existing LOX/methane rocket engines are available today.

¹An argument has also been made that LOX can be contaminated and lead to potential hazards. It is not clear how this risk compares to the complications of in-flight refueling of peroxide.

Materials and Structures

The development of advanced composite materials may provide the needed structural or thermal performance that were limiting factors in earlier TAV and SSTO concepts. These materials include carbon-carbon, boron carbon, and graphite epoxy composites; and titanium-based metal matrix, ceramic matrix, and copper matrix composites. These are useful for high-speed vehicle structures subject to air friction without the weight penalty encountered by materials previously used in aircraft and TAV programs, such as aluminum alloys and the nickel alloy Inconel. At the RAND workshop, Vincent Weldon of Boeing reported that Inconel in a Ti-Inconel structure drove costs up significantly, and suggested that titanium be given serious consideration as a honeycomb structural material in metallic TAV or X-33 designs.

The space shuttle load-bearing structure is composed of aluminum alloy. Metals may still be used in current concepts for some structures, such as lightweight aluminum-lithium for oxidizer tanks internal to the structure that would not have to carry high heat loads. Composite material might also be used for the oxygen tank, although in the past there have been concerns over leakage. Where possible, tank structures are generally designed as integral parts of the structure.

Key properties of some of these materials are shown in Table 4.4. Advanced composites like boron epoxy and graphite epoxy exhibit high strength and stiffness. These materials are not in general unidirectional; they tend to be stiff in only one direction and less stiff in others. These limitations can be overcome by using multilayer laminate construction techniques.

Perhaps their most important property for TAV applications are their high strength-to-weight ratios and stiffness-to-weight ratios compared to standard

Table 4.4
Material Properties

Material	Specific Gravity	Tensile Strength (GPa)	Tensile Modulus (GPa)	Specific Tensile Strength (GPa)	Specific Tensile Modulus (GPa)
Boron/epoxy	2.0	1.49	224	0.73	110
Graphite/epoxy	1.6-1.5	0.93-1.62	213-148	0.58-1.01	133-192
Aramid/epoxy	1.45	1.1.38	58	0.95	40
Glass/epoxy	1.9	1.31	41	0.69	22
Steel	7.8	0.99	207	0.13	27
Aluminum alloy	2.8	0.46	72	0.17	26
Titanium	4.5	0.93	110	0.21	24

SOURCE: B. C. Hoskins and A. A. Baker, *Composite Materials for Aircraft Structures*, AIAA Education Series, New York, NY, 1986.

metals. Boron epoxy has four times the stiffness of an equivalent weight of steel, and graphite epoxy is five times stronger than the same weight of L65 aluminum alloy.

An autoclave is an oven that is used to cure composite parts. Because autoclave systems can be expensive, the design of TAVs based on composite parts must consider the cost of fabrication systems needed during vehicle construction.

Thermal Protection Systems (TPS)

Rockwell's space shuttle orbiter is the only current example of a TAV. The shuttle's aerodynamic load-bearing structures are composed of aluminum and titanium alloys and can not withstand the severe thermal loads experienced on reentry into the earth's atmosphere. Thus, an external thermal protection system is needed to isolate these aerodynamic load-bearing structures, as well as the internal structures of the vehicle, from the thermal loads.

Early TPS were originally expandable materials that heated and ablated under friction to protect the underlying structure. The shuttle uses relatively delicate silica glass tiles over much of the vehicle surface. However, its TPS must be carefully inspected and repaired after each flight—a major factor limiting shuttle responsiveness. Consequently, this type of TPS would not be well suited for a military TAV for which highly responsive operations would be a requirement.

Thermal loading on a TAV is dependent on the vehicle flight trajectory, as well as the aerodynamic design of the vehicle itself. The temperature profiles for various TAVs have been obtained from several sources. A summary is given in Table 4.5.

The first two columns contain predictions obtained from a Hypersonic Technology report by the National Research Council (NRC) on the National Aerospace Plane (NASP) and from a briefing by Boeing on its proposed Reusable Aerodynamic Space Vehicle (RASV); the last column contains data

Table 4.5
Selected TAV Temperature Distributions

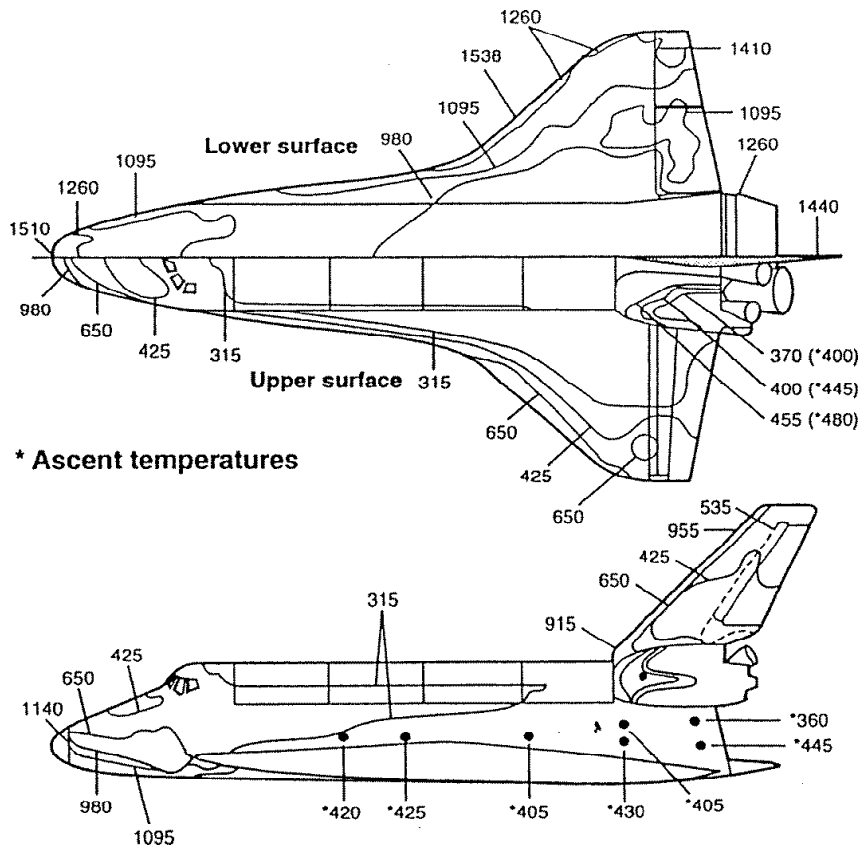
	NASP ¹	RASV ²	Space Shuttle ³
Nose, leading edges	1650-2200 C	980-1520 C	1260-1540 C
Lower fuselage/wings	N/A	650-980 C	980-1260 C
Upper fuselage/wings	N/A	480-650 C	320-980 C

SOURCES: 1: RASV briefing, Boeing Aerospace Company, no date; 2: National Research Council, *Hypersonic Technology for Military Application*, Washington, D. C., 1989; 3: D. Curry, *Space Shuttle Orbiter Thermal Protection Systems Design and Flight Experience*, First ESA/ESTEC Workshop on Thermal Protection Systems, Netherlands, May 5-7, 1993.

taken from a Rockwell report on space shuttle missions. Similar temperature profiles were predicted for the pending X-33 technology demonstrator, as shown on the Lockheed Martin website.

The predicted thermal load on the NASP is much more severe than that predicted for the RASV or measured on the space shuttle. The NASP, however, is an air-breathing vehicle with design parameters distinct from the other vehicles. Temperature profiles of the latter two vehicles are similar and could be similar to those of a military TAV with similar hypersonic characteristics.

A detailed profile of peak temperatures on the space shuttle is shown in Figure 4.6. The profile indicates where thermal loads are highest and shows that these areas are relatively small compared to the entire surface area of the vehicle. However, because of the TPS used on the shuttle, these thermal loads are not distributed over the surface of the vehicle (the ceramic tiles are good insulators), resulting in the sharp temperature gradients shown in the figure.



SOURCE: D. Curry, *Space Shuttle Orbiter Thermal Protection Systems Design and Flight Experience*, First ESA/ESTEC Workshop on Thermal Protection Systems, Netherlands, May 5-7, 1993.

Figure 4.6—Space Shuttle Peak Surface Temperature Profile

Current TPS Materials

A summary of the materials used on the space shuttle, the locations of their applications on the vehicle, and the maximum operating temperatures is given in Table 4.6.

Reinforced carbon-carbon (RCC) composites, having excellent strength and thermal properties, are used in peak thermal load regions. The RCC nose cap and wing leading edges also serve as aerodynamic load-bearing structures. They are attached to the shuttle frame via Inconel 718 and A-286 stainless steel fittings bolted onto flanges formed on the RCC components. The outer surface of the RCC panels are protected from oxidation with an silicon carbide coating. However, as Tables 4.5 and 4.6 reveal, on a typical mission the nose and leading edges of the space shuttle are subjected to temperatures exceeding the maximum operating temperature of RCC for 100-mission life. This suggests that the RCC panels have to be replaced frequently. On the space shuttle, these parts are indeed replaced after every mission.

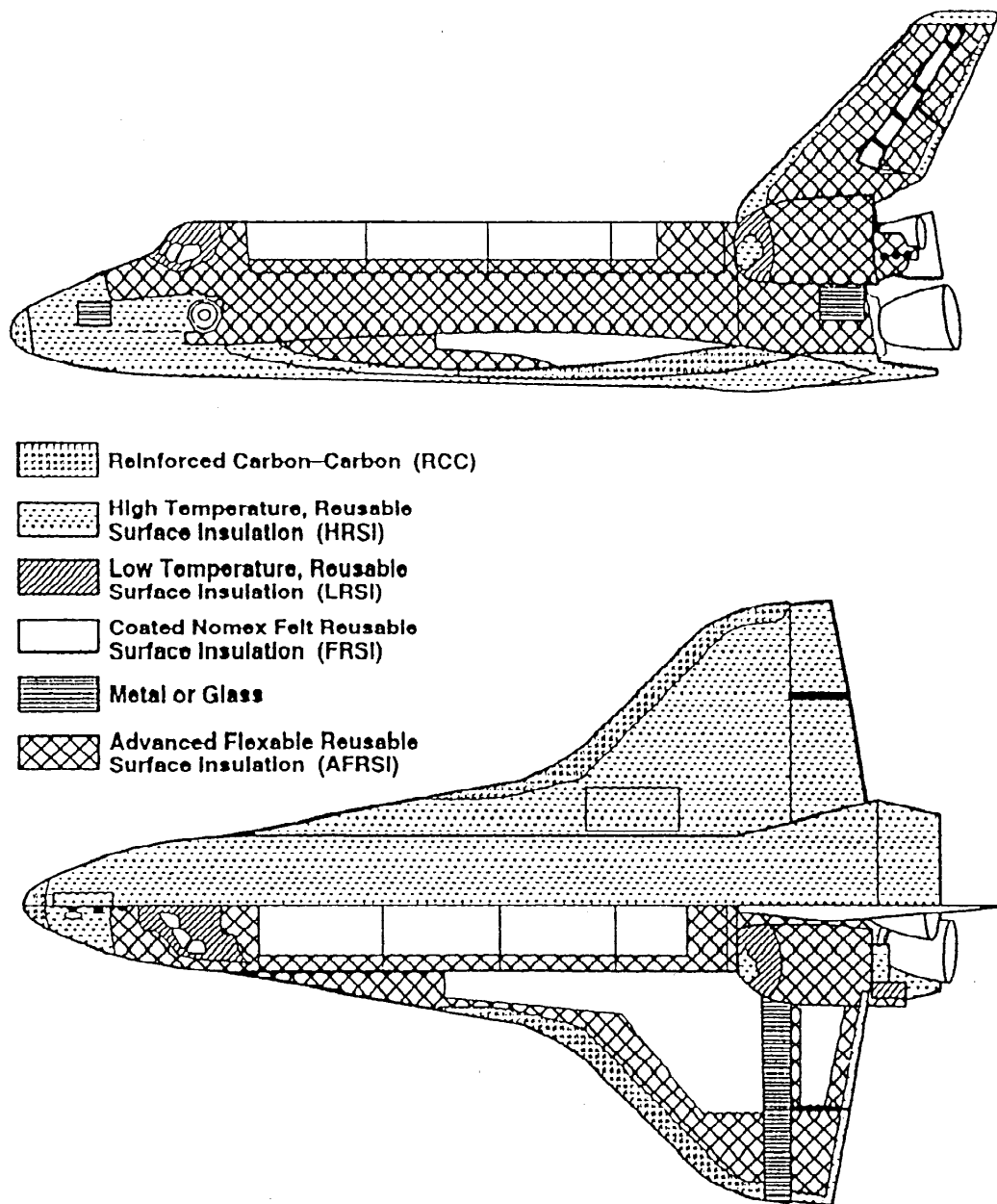
The specific locations of space shuttle TPS components are shown in Figure 4.7.

Table 4.6
Space Shuttle TPS Materials

Material	Location on Shuttle	100-Mission Life	Single-Mission Life
Reinforced carbon-carbon (RCC)	Nose, leading edges	1482 C	1816 C
High temperature reusable surface insulation (HRSI)	Lower fuselage/wings	1260 C	1427 C (LI-900) 1482 C (LI-2200)
Fibrous refractory composite insulation (FRCI)	Selected HRSI regions	1260 C	1427 C
Low temperature reusable surface insulation (LRSI)	OMS ^a pod frontal area, area surrounding window panels	1093 C	1149 C
Felt reusable surface insulation (FRSI)	Upper fuselage/wings (low-temp regions)	399 C	482 C
Advanced flexible reusable surface insulation (AFRSI)	Upper fuselage/wings (high-temp regions)	816 C	982 C

SOURCE: D. Curry, *Space Shuttle Orbiter Thermal Protection Systems Design and Flight Experience*, First ESA/ESTEC Workshop on Thermal Protection Systems, Netherlands, May 5-7, 1993.

^aOMS = Orbital Maneuvering System.



SOURCE: D. Curry, *Space Shuttle Orbiter Thermal Protection Systems Design and Flight Experience*, First ESA/ESTEC Workshop on Thermal Protection Systems, Netherlands, May 5-7, 1993.

Figure 4.7—Location of Space Shuttle TPS Components

HRSI and LRSI refer to ceramic tiles bonded to the aluminum skin of the space shuttle.

FRSI refers to a new ceramic composites that have improved strength and durability relative to HRSI tiles. FRSI tiles have been used on recent shuttle

missions in place of HRSI tiles in areas susceptible to high impact damage. FRCI tiles have a higher energy density than HRSI tiles, however.

FRSI refers to flexible Nomex felt blankets used on the low-temperature regions of the shuttle body. These blankets are lighter than ordinary shuttle ceramic tiles, and require less time and labor to manufacture, inspect, and refurbish.

AFRSI refers to a new, silicon-based flexible blanket that has better thermal properties than FRSI. Because it is lighter and less complex than tile TPS, AFRSI has been used to replace a majority of the original LRSI tiles on the space shuttle.

Ceramic composites are excellent insulators and have a relatively low density, but have very low strength and ductility. Thus, it is necessary to isolate the ceramic insulation from aerodynamic stress loads. Independent motion of the tiles accommodates the flexing, expanding, and contracting motions of the metallic skin under aerodynamic and thermal loads. The main disadvantage of using ceramic tiles is system complexity (attachment to the aluminum substrate requires high-temperature adhesives, strain-isolation pads, filler bars, and precise gaps between the tiles). Ceramic tiles are also susceptible to impact damage because of the low strength of ceramic, and are water resistant to a temperature of only 570°C. The resulting TPS therefore requires rigorous inspection and refurbishment. Space shuttle inspection and refurbishment of the TPS require 17,000 man-hours per flight.

New TAV TPS Goals

To prepare the space shuttle for flight, months of inspection and refurbishment are needed between flights. Drastically increased responsiveness is desired for commercial launch operations, and is essential for military purposes. Plus, a reduction in mission costs would make transatmospheric missions more viable. These two goals imply that the following improvements must be made to future TAVs:

- Increased reliability: more durable and reliable equipment, requiring fewer inspections and refurbishments between missions.
- Increased simplicity: equipment requiring fewer parts and fewer man-hours for maintenance.

TPS need improvement for these new goals to be met. The disadvantages of the current shuttle TPS mentioned above must be overcome before significant advances can be made. The above TPS requirements can be met if materials can be manufactured that are stronger, lighter, have better thermal properties, and

can be used as multi-function (i.e., aerodynamic and thermal load-bearing) structures. First introduced in the RASV program, advanced metallic alloy TPS may provide the reliability and simplicity needed for future TAVs.

Metallic TPS Materials

Metallic alloys offer several significant advantages over ceramic alloys. First, they are more resistant to impact damage. Also, they may be welded or fastened with rivets, potentially providing more flexibility in installation. Furthermore, because metals are conductive, they may redistribute heating loads over the body to reduce severe thermal gradients in peak heat-loading regions. Lastly, because they have greater strength and toughness than ceramics, metallic alloys can be used as aerodynamic load-supporting structures. This dual function of thermal and aerodynamic load support may eliminate the need for adhesives, strain-isolation pads, and gap fillers that are necessary with ceramic tiles. Lighter, more fragile internal insulation may also be used to enhance heat isolation.

The disadvantages of metallics are (1) higher density than ceramic tiles, (2) higher thermal conductivity, and (3) possibly greater difficulty in refurbishment if metallic panels are incorporated into load-supporting structures. The last disadvantage, however, is primarily a design problem associated with using the materials in a multi-functioning capacity.

Titanium aluminides are “advanced” metallic compounds developed for the NASP. The two classes of aluminides, alpha (TiAl) and beta (Ti₃Al), offer the advantages of a high maximum operating temperature (800°C) compared to other titanium compounds (500-700°C), oxidation and creep resistance, and, because of the high aluminum content, relatively low density. However, because the manufacturing processes have not yet been optimized, these materials have relatively poor room-temperature properties: they have low ductility, which results in low fracture toughness. In other words, they are more brittle than common titanium compounds at room temperature.

Commercially available advanced titanium alloys such as Ti-1100 and IMI-384 offer reasonable room-temperature ductility and toughness. They also possess high strength and creep resistance. However, a maximum operating temperature of 700°C limits their range of usefulness as a TPS material. On the space shuttle, for instance, these titanium compounds may be used only on the upper fuselage. They also have higher densities than the titanium aluminides.

Nickel-based alloys such as Inconel 617 have significantly higher maximum operating temperatures (e.g., 1100°C for Inconel 617), allowing use on lower

fuselage surfaces and other high-temperature areas. They are also creep and oxidation resistant. They have significantly higher density than titanium alloys, however.

Performance-Enhancing Processes. Several manufacturing processes have been developed that allow “tuning” of material properties to enhance specifically desired characteristics. For example, heat treating enhances the strength of the material. Dispersion strengthening, in which particles of a different material are dispersed within the alloy matrix, enhances stiffness and thermal stability. These dispersion-strengthened materials are called metal matrix composites (MMCs). Finally, various coatings provide improvements in properties such as oxidation resistance, water resistance, and thermal emittance. These performance-enhancing processes allow the designer to improve thermal and structural characteristics of the material specifically where most needed.

X-33 TPS Materials. The Lockheed-Martin X-33 technology demonstrator uses a metallic TPS. According to representatives from Rohr (the subcontractor building the TPS), “the metal is more durable and lighter than normal shuttle insulation, but not as good an insulator.” The lifting-body design of the X-33 allows reentry with a less severe heat-loading profile than the space shuttle, affording the use of the metallic TPS. In fact, temperatures on most of the bottom fuselage surfaces are predicted to be lower than 1300°F (700°C) and may be protected by titanium. Higher-temperature areas will have Inconel 617, and peak heating areas such as the nose and leading edges will be protected with RCC.

Several questions remain regarding the X-33 TPS. The first is the effect of the lifting-body design on the maneuverability and stability of the vehicle. Second, is the TPS fastened with rivets or bonded with adhesives? Are thermal sealants used in the gaps between the panels? Does the metallic TPS redistribute heating loads on the body surface? Also, are the RCC components designed to be replaced after every mission as they are with the space shuttle?

Observations on Thermal Protection Systems

Our review of TPS materials indicates that it should be feasible to design an alternative TPS from advanced metallic alloys, provided the reentry path and aerodynamic design of the vehicle body result in reentry temperatures that are less severe than those found on the space shuttle. Although peak temperature locations would probably still require RCC to withstand reentry thermal loads at selected areas like the nose, many TAV designs may be protectable by strategic combinations of metallic panels. Although metallic panels are of higher density than ceramic tiles, the metallic TPS may be lighter and simpler by eliminating the

need for a complex adhesive system like that used on the space shuttle. The panels may also serve as aerodynamic load-bearing structures, eliminating the necessity for an underlying airframe.

Additional processing of the materials may enhance specifically desired properties to improve TPS performance.

Finally, by optimizing the vehicle's aerodynamic design it may be possible to reduce the thermal loads on the vehicle, thereby decreasing the degree of thermal protection required. These improvements and the other improvements mentioned above could result in a TPS that is more reliable and less expensive to maintain than that of the current space shuttle, thereby enabling development of a TAV that could responsively carry out military missions.

Vehicle Integration

To maximize orbital vehicle performance, it is necessary to efficiently integrate the engine, the airframe, and propellant tankage in a way that allows for good vehicle hypersonic characteristics as well as a minimum amount of added weight for subsystem integration interfaces (such as the TPS system and internal load-bearing structures or payloads).

If the system has two stages, they must be carefully and efficiently integrated as well. For either an aerial-refueled system or for air-launched system concepts, this implies careful attention to the mechanical interfaces and operational procedures during the staging maneuver. In either case, there is increased operational complexity and hazard introduced relative to an SSTO system.

Concepts in which the first-stage launch aircraft carry the TAV either under or on top of the aircraft must consider the drag effect that such structures have on the performance of the launcher platform, which must itself reach altitudes and speeds suitable for the system launch envelope. The separation dynamics of different TAV and aircraft combinations must be carefully considered, and for safety reasons subsonic air-drop configurations may be preferred.²

From the rocket equation for a SSTO design, typical delta-V required for orbit combined with an Isp of 460 seconds implies that 88 percent of the gross lift-off weight (GLOW) will be accounted for by propellant. SSTO and other TAV designs (including those using different fuels and lower specific impulse engines) must have a mass fraction sufficient to accommodate not only the

²The easiest approach is a subsonic air-drop, such as is used in Pegasus.

payload but also the structure, undercarriage and landing gear, propellant for delta-V maneuvering on-orbit, avionics, TPS, and other systems.

Mass fractions are more difficult to estimate for rocket vehicles than for conventional aircraft, especially because design experience for SSTO concepts is so limited. This introduces a significant uncertainty regarding weight of the mainframe and other results computed using aircraft-based models. Consequently, there is significant uncertainty regarding the TAV mass fractions predicted using these models.

Another vehicle integration and design issue is body shape, and the advantages or disadvantages of a design with regard to the encapsulation of propellant tanks and payload or crew areas. The role of wings versus lifting bodies must also be considered in terms of aerodynamics as well as weight goals. Lifting bodies have low lift-to-drag ratios (L/D) at subsonic speeds and moderate to high L/D at supersonic speeds, but do not have wings, which may introduce stability concerns on landing. On the other hand, wings are a parasitic weight when rocket propulsion is dominant over lift, as it is during the ascent phase. However, a design with small wings may imply high landing speeds, and this factor must be considered if the TAV design landing mode is horizontal.

Maneuvering in space is a fuel-consuming process, so the question is raised as to whether or not it is advisable to plan for a strategy of dipping into the atmosphere and using aerodynamics to achieve a cross-range capability. If this is done, then the effect on the lifetime of the vehicle must also be considered. Would a metallic design be more robust? What feedback effect would it have on demands for the propulsion system—would there be a significant savings over the life cycle of a vehicle if an atmospheric cross-range capability were built in?

Although LOX-LH2 offers high Isp, liquid hydrogen has a very low density, which in turn requires large tanks to contain it. This adds to the structural weight and volume of the vehicle. Such volumetric concerns are important system integration issues for air-launched TSTO TAV design concepts.

During the workshop sessions, it was indicated that a 1.5 percent weight ratio, significantly lower than for any other landing gear, was designed for the B-58 on display at the Wright Patterson Museum. The implication was that one could possibly trade landing gear weight for increased engine weight and perhaps gain increased propulsion performance.

TAV Design Scalability

Past experience with the design and development of aerodynamic vehicles demonstrates the value of starting with a subscale X (research) vehicle as part of an evolutionary development path toward a bigger, more capable Y (prototype) vehicle. An exploratory first step might be followed by an intermediate step to gather information on the effects of the environment, followed by a third step or expansion to the full-up operational vehicle. There were concerns expressed at the workshop that prototypes too frequently end up looking like the end item, and that the intermediate step is never really taken. The implication was that if R&D and user groups generally cannot agree on one design, vehicle design suffers and the potential scalability of the prototype vehicle design is reduced. Furthermore, if the need later develops for specialized vehicle applications, it may not be possible to accommodate those applications with a vehicle based on the original prototype concept. The capability to directly scale TAVs using subsystems that would be common and need only repackaging (i.e., no new designs) would be an ideal feature.

Special and unique scalability challenges apply to air-launched and aerial-refueled TSTO TAV design concepts. These challenges and their implications were discussed in the previous subsection and relate to the limits on the size of the orbital vehicle the first-stage carrier aircraft can carry to altitude, and practical limits as to how much propellant can be transferred to an orbital vehicle (and how quickly). Some and perhaps all of these scalability limitations can be overcome by developing new and larger transport or tanker aircraft, but this would raise the overall development for the design concepts significantly (anywhere from \$2B to \$5B for development alone of a new carrier aircraft) and may make them unattractive from a budget standpoint to SSTO RLV alternatives. Thus, it appears that air-launched or aerial-refueled TSTO TAV concepts would be attractive from a budget standpoint only if they are designed for small- to medium-sized payloads.

X-33 Vehicle Scalability Challenges

Two major scalability design challenges in going from an X-33 to an RLV that were mentioned by the working group were the need for the development of (a) high temperature composite thermoplastic types of materials and (b) extremely high thrust to weight ratio engines.

The discussion above on aerospike engines illustrates the scalability challenge faced by Lockheed Martin and Aerojet in the propulsion area. Similar

propulsion scalability and performance challenges would face the other X-33 competitors in trying to scale up their vehicle designs from suborbital X-33 vehicles to full-scale SSTO RLVs capable of delivering sizable payloads to orbit.