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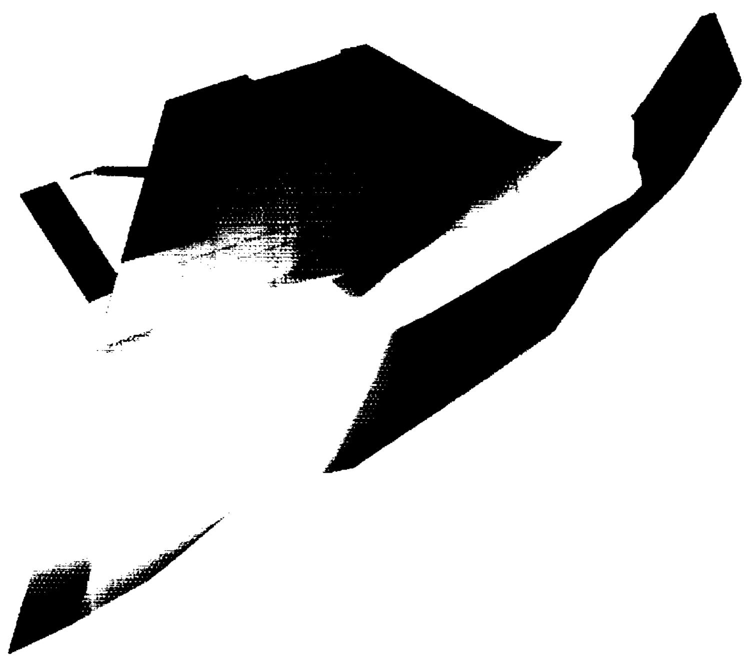
Oxidizer Selection for the **ISTAR** Program (Liquid Oxygen versus Hydrogen Peroxide)

Jason Eugene Quinn
NASA Marshall Space Flight Center, Huntsville, Alabama



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OXIDIZER SELECTION FOR THE ISTAR PROGRAM (LIQUID OXYGEN VERSUS HYDROGEN PEROXIDE)

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Abstract

This paper discusses a study of two alternate oxidizers, liquid oxygen and hydrogen peroxide, for use in a rocket based combined cycle (RBCC) demonstrator vehicle. The flight vehicle is baselined as an airlaunched self-powered Mach 0.7 to 7 demonstration of an RBCC engine through all of its air breathing propulsion modes. Selection of an alternate oxidizer has the potential to lower overall vehicle size, system complexity / cost and ultimately the total program risk. This trade study examined the oxidizer selection effects upon the overall vehicle performance, safety and operations. After consideration of all the technical and programmatic details available at this time, 90% hydrogen peroxide was selected over liquid oxygen for use in this program.

Introduction

The Advanced Space Transportation Program at NASA Marshall Space Flight Center has assembled a government / industry team to conduct the system development and ground test of a RBCC propulsion system (and potentially flight). Government team membership includes participants from several NASA centers: Dryden Flight Test Center, Glenn Research Center, Langley Research Center, Marshall Space Flight Center and Stennis Test Center. The primary industry team member is the Rocket Based Combined Cycle Consortium (RBC^3), which includes Boeing Rocketdyne, Gencorp Aerojet and United Technologies Pratt & Whitney[1]. Additional industry team members include Boeing for vehicle activities and several other companies.

Current plans for the propulsion system development and ground test of the RBCC engine system are funded through NASA MSFC as the Integrated System Test of an Airbreathing Rocket (ISTAR) program. The primary ISTAR program objective is to develop a propulsion system, which would be capable of powering a flight demonstrator vehicle from launch off a B-52 aircraft (approximately Mach 0.7) up to scramjet speeds of about Mach 7. This flight velocity range would demonstrate all of the RBCC engine operational modes: Air Augmented Rocket (AAR), Ramjet and Scramjet. Although only the ISTAR propulsion system development and the initial ground test program is currently funded, due to the tightly integrated nature of a hypersonic vehicle, initial demonstrator vehicle design is being performed to define the propulsion system requirements.

Initial conceptual design of hypersonic vehicles requires the designer to assume many system characteristics such as propellants, thrust, and vehicle size to satisfy mission objectives. The designer uses their experience on previous similar vehicles to decide what to select. Although these initial "guesses" are often quite good, they can limit the ability of the design if one does not go back and check their assumptions. For example, one of the first items to decide upon is the propellant combination that the system will use. Typically the designer would select the highest ISP combination propellant, but this might not be the best selection for non-orbital missions. A lower performing (in terms of ISP) propellant combination that is more dense or non-cryogenic could result in a smaller, lower cost, more reliable or more operable vehicle system.

The ISTAR program has conducted several trade studies on the "right" propellant combination to select for this program. Selection of a hydrocarbon (HC) as the fuel for the ISTAR program over liquid hydrogen, although an interesting trade study, will not be discussed in this paper. Once the decision to use a hydrocarbon fuel was made, there were still at least two oxidizers that could potentially meet the system requirements: liquid oxygen (LOX) and hydrogen peroxide (HTP). The purpose of this paper is to detail the trades that the ISTAR program went through in selecting between these two oxidizers.

General Considerations

Most of the rocket industry moved away from the use of hydrogen peroxide in the early 1960's for several reasons: large all rocket propulsion systems went to LOX due to the increase in performance (ISP), long term storage users went to NTO/MMH and mono-propellant customers switched to Hydrazine. The recent resurgence of interest in HTP in applications where LOX would typically be used is in part due to the realization that higher ISP is not the entire story. Higher propellant density and storability can, in some cases, make HTP a better choice than LOX. This is especially true for systems that have aerodynamic drag losses or are severely volume constrained[2]; ISTAR has large aerodynamic drag (relative to an all rocket) due to its airbreathing trajectory and severe volume constraints due to its slender hypersonic shape.

Hydrogen Peroxide has physical properties very similar to water (i.e. density, color, viscosity, etc.)[2]. The primary exception is that the molecular structure is only meta-stable and will exothermically decompose from H_2O_2 (basically a water molecule with an extra oxygen atom attached) at some rate to a water molecule, oxygen and energy. This hot steam and oxygen (1300 °F) can then be expanded out a nozzle as a monopropellant thruster or the hot oxygen can be combusted with a fuel. In either case the hydrogen in HTP is tightly bound in a water molecule and is <u>not</u> combusted. When the concentration is quoted as 90% that means 90% Hydrogen Peroxide in solution with 10% water and traces of stabilizers. In a compatible storage container 90% HTP has been observed to decompose at less than 1% per decade but in the presence of a catalyst HTP can be caused to decompose extremely rapidly[7]. HTP has been used as a turbomachinery drive gas in many systems: V-2, X-1, Redstone, Jupiter, Centaur, Viking, X-1 and the X-15 [2,4]. HTP has also been used as the primary oxidizer (bi-propellant) in many propulsion systems: Me 163 Komet, Gamma 201/301, AR series of rocket engines, LR-40 and others[2,4,5,6] (as the majority of these flight systems used 90% HTP it was the first concentration considered). HTP physical properties of interest are compared to LOX in Table 1 and density compared to several propellants in Figure 1.

Figures 2 & 3 show the ideal specific impulse and density specific impulse of LOX and several concentrations of HTP with hydrocarbon fuel. Density specific impulse is the density of the propellant combination (i.e. bulk density or $\rho_{mixture}$) multiplied by the specific impulse. It is often used to select propellants when volume considerations are taken into account. 90% HTP is only 15 lb/ft^3 denser than LOX, but the peak ISP occurs at a much higher mixture ratio (typically 2.6 for LOX/HC versus 7 for HTP/HC). This results in a much higher bulk density for HTP/HC over LOX/HC. The argument can be made that for a sufficiently low mission velocity change (ΔV), a lower performing (in terms of ISP) denser propellant combination would yield a smaller propellant volume required. Typically we assume that the mission cost is roughly proportional to vehicle size – thus the lower performing propellant would be a "better" choice.

Generic Comparison

In order to determine if our vehicle would be "better" with a particular propellant combination we can calculate the propellant volume required for our given mission ΔV and assumed mission averaged effective specific impulse (I*)[3].

$$Volume = \frac{\left(e^{\frac{\Delta V}{g \cdot I^*}} - 1\right) Mass_{final}}{\rho_{mixture}}$$

In the case of ISTAR our initial vehicle utilized the rocket thrusters only for AAR Mode which is Mach 0.7 (air drop) to Mach 3.0-4.0 where the rocket is to be turned off (see Figure 10) and the engine transitioned to Ramjet mode. The vehicle is assumed to perform identically for either oxidizer above Mach 3.0-4.0 as Ramjet/Scramjet modes only use tanked HC fuel and air. For a first cut analysis assume that the mass of the vehicle is the same at the end of AAR mode (Mach 3.0-4.0) for either oxidizer (which approximately corresponds to ΔV of 2000 to 4000 ft/sec). With this assumption we can plot, Figure 4 which shows the propellant volume required with either oxidizer for the Air-Augmented Rocket (AAR) mode of operation. The point made by Figure 4 is that below a mission ΔV of approximately 4000 ft/sec HTP/HC will require less propellant volume than LOX/HC for the I* value assumed. These I* values were generated from our three degree-of-freedom trajectory model. If you assume the same total propellant volume is available with either oxidizer, then one can plot Figure 5 which shows the I* required for each propellant combination at several values of mission ΔV. For example, Figure 5 shows that for a ΔV of 2000 ft/sec if the I* for the LOX/HC vehicle is 240 sec than the I^* of the HTP/HC vehicle needs to be 196 sec or higher to have the same or less propellant volume required. From these graphs we can conclude that below ~ 4000 ft/sec ΔV a 90% HTP/HC thruster will require less tank volume than a LOX/HC thruster including the effect of the additional mass of the more dense propellant provided the I* of the HTP/HC vehicle in AAR mode exceeds the I* of the LOX/HC value given in Figure 5.

Background - Initial System Definition

In order to understand the conclusions drawn from the trades performed some familiarity with the ISTAR vehicle (X-43B) and engine system is needed. The ISTAR vehicle was designed around an existing hypersonic vehicle shape designed for liquid hydrogen which was then modified for our particular mission and propellant combination. Figure 6 shows three views of the current configuration with the propellant tank location / volume emphasized. Generically this configuration is a lifting body with external forebody compression, moving cowl flap inlet with fixed internal geometry and external aft expansion.

A generic mission (shown in Figure 7) begins when the ISTAR vehicle is dropped from the NASA B-52 at Mach 0.7 and 40,000 ft, accelerates to approximately Mach 7 at 90,000 ft before shutting down the engine and gliding back for re-use. After being dropped from the B-52 vehicle the vehicle freefalls for a few seconds before starting the rocket thrusters and accelerating in Air-Augmented Rocket (AAR) mode. In this propulsion mode the rocket thrusters are firing at full thrust and additional fuel is injected to burn with the incoming air. As the vehicle accelerates through Mach 3-4 the rocket thrusters are turned off and the vehicle continues to accelerate in Ramjet mode. Upon reaching Mach 6-7 the vehicle transitions fully into Scramjet mode before shutting down - completing the demonstration of all the airbreathing propulsion modes and the transitions between them for an RBCC engine.

The ISTAR team baselined a simple engine system based upon the type used in the X-15 (and many other previous systems) using a high-pressure tank of HTP to drive the fuel and oxidizer pump. The functional schematic for this type of system with either LOX or HTP as the primary oxidizer is shown in Figures 8 & 9. This type of system was selected to keep the development costs of the engine system as low as possible and still allow the RBCC multi-mode operation. The ISTAR program is focused on engine flowpath performance throughout the mission trajectory, especially in mode transitions, and not in engine system development. Due to flight experience with many previous programs, 90% HTP was baselined to be used for the turbine drive gas rather than a higher concentration of HTP, which might have increased performance (and increased risk).

Figure 8 shows the LOX/HC engine system and all of the functional components required to operate in the engines different modes. As previously stated the system uses a high pressure HTP tank to provide hot gas to the turbine drive (yellow). As this tank is not linked to the main oxidizer tank we can run just the fuel pump (no rocket thruster oxidizer needed in Ram / Scramjet modes). Figure 8 shows the vehicle systems in the top left portion of the figure with the main fuel tank, high-pressure HTP tank and main LOX tank along with the gray purge gas tanks. The top right portion of the figure shows the systems required on the carrier aircraft (B-52) primarily the LOX top off tank (note the purge gas supply was the same for both the HTP and LOX systems and is not shown). Finally the bottom left side of figure 8 shows the engine system components including the ignition system (baselined combustion wave ignition [CWI] for the LOX/HC system). The components highlighted with a red background are those that are different between the LOX and the HTP system – note that the primary difference between figures 8 and 9 is the LOX top-off system and the ignition system (CWI for LOX/HC and catalyst packs for the HTP/HC system)

Figure 9 shows the HTP/HC engine system. This system is very similar to the LOX system but replaces the CWI system with a catalyst pack arrangement to provide auto-ignition in the rocket thrusters and doesn't require LOX top-off. These two changes reduced the number of fluids on the vehicle and the complexity of the functional schematic considerably.

Detailed Oxidizer Trade

The ISTAR engine system was originally baselined to be LOX/HC as these propellants are familiar and were believed to provide enough performance without the severe volume penalty of LH2. Prior to the initial formation of the RBC[3] team a conceptual trade study on the propellant selection indicated that replacing LOX with 90% HTP would allow a smaller propellant volume to complete the mission (Mach 0.7 to 7). This rudimentary study results and several discussions between NASA and RBC[3] provided motivation to complete a comprehensive trade study examining in detail the system impacts of switching the oxidizer from LOX to HTP.

The oxidizer trade study team brainstormed a detailed list of the important criteria that were judged to affect the entire system design. Table 2 shows these criteria grouped into five sections: Safety, Programmatic, Mission Success/ Engine System Design, Mission Success/ Vehicle Integration and Operations. Within these five sections the criteria were also grouped into categories (i.e. vehicle design impacts). Each of the five sections was assigned a weighting factor, which attempted to capture program management's preference or importance level for that section. Each criteria was to be assigned a 1, 3 or 9 score for how beneficial an oxidizer was to the system on this criteria. Scores were then combined with the weighting to produce a single ranking for each oxidizer.

Each of the criteria was assigned a criteria owner(s) who was responsible for investigating the criteria, selecting a score and providing a group presentation to back up that score. The criteria owner(s) then presented the score to the entire trade study team for discussion and the trade team selected a final consensus score.

The process of assigning a team consensus score required all of the team members to become educated about the criteria being considered. This educational process enhanced the team objectivity and the entire trade study. Criteria owners were selected from the RBC^3 team members, NASA centers and Boeing. The entire team leaned heavily on the recent operational experience with both LOX and HTP at NASA STENNIS and the flight operations experience (both recent and historical) at NASA DRC.

Very early in this trade study the trade study team explored the potential for use of 98% HTP rather than 90% HTP. 98% HTP was considered as there was additional performance over 90% HTP. However, for this low ΔV mission, the additional unknowns and development risk with 98% HTP was judged to be not worth this small performance boost. The remainder of the trade study was performed comparing 90% HTP to LOX only.

Safety

This category was assigned the heaviest weight at 25% (see Table 2) but was the least controversial in team discussions. Peroxide was judged to be safer overall than LOX primarily due to the need for a LOX top-off system to transfer LOX from the B-52 to the X-43B vehicle (similar to how the X-15 worked). This LOX top-off system would have had to operate on the manned carrier aircraft during the entire flight up to X-43B drop. HTP has a higher degree of risk with leaks and spills in the engine / vehicle system but was considered safer than LOX when considering the entire propellant handling process. These two criteria balanced one another out in the scoring leaving HTP scoring safer than LOX for the ISTAR project.

Programmatic

The next three categories were considered to be of equal importance and were all given the same weight of 20%. The programmatic criteria required much discussion and work before the team could agree on a consensus score in the three areas of Schedule, Cost and Risk. After significant discussion the schedule for either oxidizer was determined to be equivalent provided additional money was made available and / or more risk was accepted for HTP. The initial development cost with HTP was considered to be significantly more (on the order of several million dollars) primarily due to the need to develop the HTP cooled thruster. Considering the entire system HTP would save significant dollars ($3-5 million) due to not needing a LOX top-off system but the additional cost for facility modification influenced the team to only allot a slight advantage to HTP in facilities / test costs. The final criteria of programmatic risk determined that selecting HTP would add more overall risk to the program. This risk would be primarily in the beginning of the project during the thruster development while selecting LOX would potentially delay the risk of the LOX top-off system to the flight phase of the program.

Mission Success / Engine System Design

The single most controversial criteria in this oxidizer trade study is the technology readiness level and associated risk in developing the rocket thrusters with either oxidizer including oxidizer cooling. Due to the need to cool the significant surface area of the duct with the fuel being fed to the rocket both the LOX/HC and HTP/HC were assumed to use oxidizer cooling of the rocket thrusters.

Selection of LOX allows us the comfort level as the thruster would be similar to existing experience base (combustion chamber parameters) but the requirement for LOX cooling introduces a number of system issues which increase mission success risk. NASA GRC has test fired LOX cooled thrusters, which addressed the majority of the issues with LOX cooling, but they did not address the closed loop issues (cooling LOX was not injected and burned in this test series). The closed loop issues like the two phase regenerative cooling during the start transient is risky and potentially would not be discovered until late in the engine system development. Other potential risks with LOX cooling include freezing of the HC in the lines and the difficulty of dealing with fuel temperature changes as the vehicle accelerates through the Mach number range.

Selection of HTP limits the material selection of the thruster as many metals (including copper) catylize the decomposition of HTP and are typically not considered for use in HTP systems. After several iterations NASA and RBC^3 developed the design for two different rocket thrusters (using 347 stainless steel) each of which was capable of performing the mission with some film cooling. Other difficulties introduced by the selection of HTP is the difficulty of packaging the catalyst packs, and a lower thermal margin in the thruster design. After several meetings on this criterion the team determined that LOX was preferred over HTP in the thruster design.

Pump and Ignition Systems were judged to be of equal level of difficulty. The inherent risk in the system due to the startup / shutdown transient and mixture ratio of the engine is higher for the LOX system (more difficult) primarily due to the need to start the thrusters

LOX rich and pass through stoichiometric and how the LOX density changes during the start.

The engine system design criteria favoring LOX over HTP are: less known issues with the oxidizer that might require development effort (known unknowns), a better current design experience base, higher rocket ISP (small advantage) and likely longer hardware life (primarily due to higher thermal margin in the LOX system). Criteria favoring HTP over LOX are primarily due to the lower system complexity. Selection of HTP significantly reduced the engine system complexity in terms of number of propellants and complexity of the ignition system as can be seen in Figures 8 & 9. The LOX/HC system shown conceptually in Figure 8 needs an ignition system (CWI) and a LOX top-off system. While Figure 9 shows that the HTP/HC system uses a catalyst pack for ignition and doesn't require a LOX top-off system. Also HTP is likely to require less purge gas and no chill preconditioning of the oxidizer hardware.

<u>Mission Success / Vehicle Integration</u>

Unlike the Engine System Design criteria discussed above <u>all</u> the Vehicle Integration criteria trades favored the selection of HTP over LOX. Due to the wedge shape of hypersonic vehicle resulting in a low available volume for propellant, the propellant tanks are typically required to be integral (the tank is the vehicle). Integral cryogenic LOX tanks have never been developed and were judged to be more difficult than the material compatibility issues with integral tanks for HTP. With these considerations HTP was obviously preferred from a structural criteria.

Examining the engine system packaging and propellant feed system issues introduced via each oxidizer came down to the non-cryogenic nature of HTP removing/reducing the need for vacuum jacketed lines, cryogenic insulation, with the potential for removal of the boost pump (considered part of the vehicle). HTP would still require some thermal management but the non-cryogenic nature was felt to be much easier to deal with.

As previously discussed the higher density of HTP/HC in a volume limited, low ΔV vehicle like X-43B compensates for the lower ISP of the rocket thrusters. This assumes that the same volume is available for the propellant and the vehicle weight won't change with the higher propellant weight. While the assumption that vehicle weight is relatively constant with propellant weight is valid for this vehicle, the assumption of the same propellant volume available is not. LOX would likely require some amount of cryogenic insulation to prevent the formation of ice on the vehicle (also would reduce the amount of top-off needed from B-52). Assuming an insulation thickness of 1 to 2 inches yields an amount of insulation equivalent to 10-30% of the total LOX volume. HTP was preferred over LOX for the vehicle integration performance criteria due to this higher potential volume for propellant.

The remaining vehicle integration criteria dealt with the carrier aircraft. The oxidizer selection was felt to heavily influence the impacts on the carrier aircraft on terms of consumables required to be in-flight transferred. HTP would require only nitrogen (or helium) while LOX was assumed to require a higher amount of nitrogen as well as LOX for top-off and chill down. Additionally the avionics / control onboard the B-52 would be much more critical as it would be a manned system with LOX venting. All of the above considerations resulted in the entire section of the trade matrix associated with vehicle integration to be heavily weighted toward the selection of HTP.

<u>Operations</u>

Operations had the lowest weighting factor at 15%. This lowest weight does not mean that the team did not consider operations important (it is) but this is an engine system for an X-vehicle only meant for ~25 flights. Most of the criteria considered in the operations area favored HTP over LOX with the exception of operational procedures being well established. NASA has had recent experience at STENNIS in the E3 test stand with both LOX and 90% HTP but LOX procedures are simply better know and more established. HTP was considered to have slightly easier handling than LOX (NASA STENNIS & DRC experience), easier ground operations with less equipment and less vehicle ground operational difficulties (i.e. simpler setup/servicing).

Summary and Conclusions

A trade study considering two alternate oxidizers, liquid oxygen or 90% hydrogen peroxide, for a rocket based combined cycle demonstrator vehicle was completed. This trade study considered the overall system performance from both a technical and programmatic viewpoint, to select the lowest risk solution. Given the limited energy requirement (ΔV) of the demonstrator vehicle (Mach 0.7 to 7), the higher density and mass ratio of 90% hydrogen peroxide yielded similar vehicle performance when compared to LOX. Additionally, hydrogen peroxide provided system simplification, increased flight safety and packaging advantages. After consideration of the technical and programmatic details, 90% hydrogen peroxide was selected over liquid oxygen for use in the ISTAR program.

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Table 1: Oxidizer properties compared[7,8].

Fluid Properties	90% HTP	LOX
Boiling Point, °F (extrapolated at 1 atm for H_2O_2)	286.7	-297.4
Freezing Point, °F (1 atm)	11.3	-362
Bulk Decomposition Temperature, °F (red line)	275	NA
Density, g/cc (H_2O_2 @ 77 °F, 14.7 psia; LOX @ -297 °F, 14.7 psia)	1.387	1.14
Density, lbm/ft^3 (H_2O_2 @ 77 °F, 14.7 psia; LOX @ -297 °F, 14.7 psia)	86.6	71.2
Heat Sink, BTU/lbm (77 °F to 250 °F for H_2O_2; -300 °F to 140 °F for LOX)	114.7	178
Critical Pressure, psia (estimated for H_2O_2)	3556	730.4
Critical Temperature, °F (estimated for H_2O_2)	833	-181.8
Cost, $/lbm	3-4	.042-.068

Table 2: Liquid oxygen versus hydrogen peroxide criteria trade matrix.

Section	Category	Criteria	Score HTP	Score LOX
Safety	Engine/Vehicle	Test Personnel/Facilities	**1**	**1**
		ISTAR Engine/Vehicle	**1**	**3**
	Handling	Personnel/Hardware	**3**	**1**
Weighting Factor: 25%	Carrier Aircraft	B-52 Crew/Aircraft	**9**	**3**
		4	14	8
Programmatic	Schedule	SSC Ground Test	**9**	**9**
	Cost	Development	**3**	**9**
		Facilities Modifications/Test	**3**	**1**
Weighting Factor: 20%	Risk	Cost & Schedule (Confidence in)	**3**	**9**
		4	18	28
Mission Success / Engine System Design	Technology Development Level/Risk	Thrusters	**1**	**3**
		Pumps	**9**	**9**
		Ignition System	**3**	**3**
		Transient Operations	**9**	**3**
		Known Unknowns	**1**	**9**
	Experience Base	Expert Knowledge/Experience	**3**	**9**
	Performance	Engine System Performance	**1**	**3**
		Design Complexity (No. Values, Pumps, etc)	**9**	**1**
		Hardware Life	**3**	**9**
	Resources	Purge Requirements	**3**	**1**
Weighting Factor: 20%		Chill & Conditioning	**9**	**1**
		11	51	51
Mission Success / Vehicle Integration	Vehicle Design Impacts	Structural	**3**	**1**
		Engine System Packaging	**3**	**1**
		Propellant Feed System	**3**	**1**
	Performance	Fuel Margin Remaining (FMR)	**3**	**1**
	Carrier Aircraft	Consumables (Propellant, purge transfer)	**9**	**3**
Weighting Factor: 20%		Avionics/Control Interface	**9**	**3**
		6	30	10
Operations	Handling	Known Hazards	**1**	**1**
		Operational Knowledge/Procedures	**1**	**3**
		Transportaion, Handling, Storage	**3**	**1**
	Ground Operations	Test Operations	**3**	**1**
		Vehicle Ground Operations	**3**	**1**
Weighting Factor: 15%	Logistics	GSE, Expendables, Soft Goods, etc	**3**	**3**
		6	14	10
Raw Score		31	**127**	**107**
Safety			3.5	2.0
Programmatic			3.6	5.6
Mission Success/Engine System Design			10.2	10.2
Mission Success/Vehicle Integration			6.0	2.0
Operations			2.1	1.5
Weighted/Normalized Score			**126**	**106**

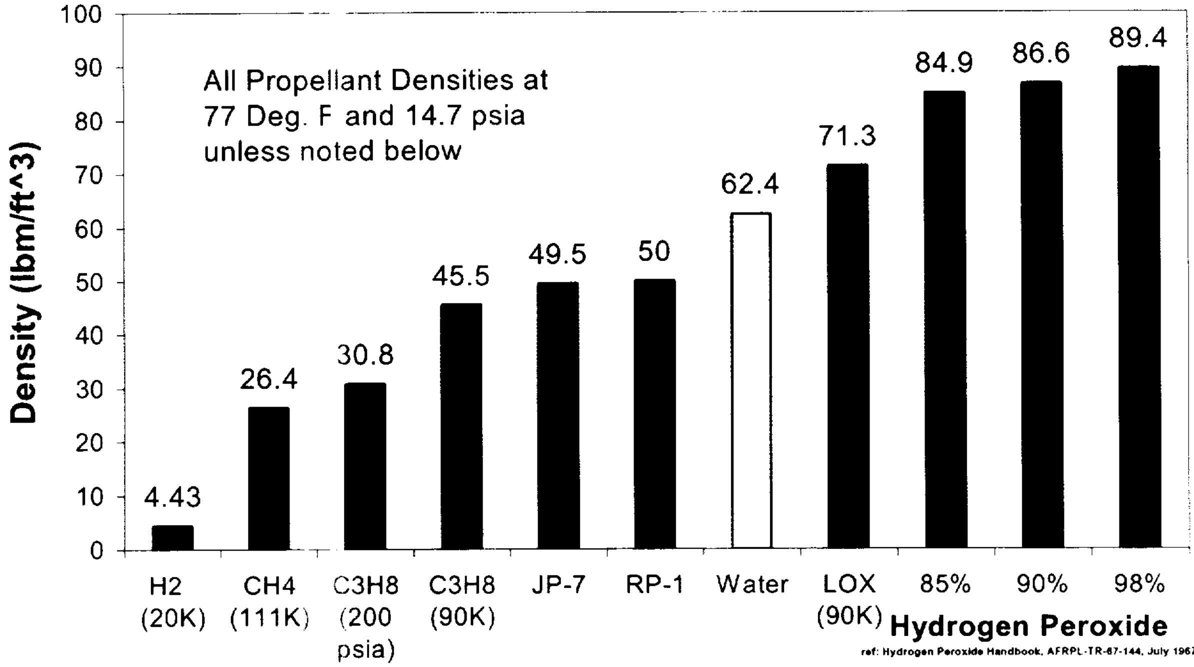

Figure 1: Densities of various propellants compared[7,8].

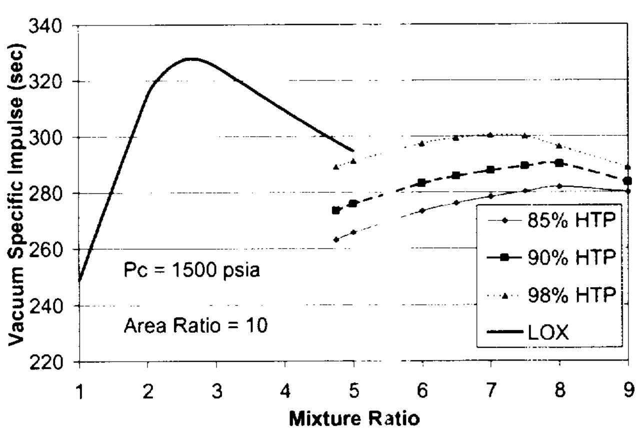

Figure 2: Ideal vacuum specific impulse for hydrocarbon (HC) fuel and various oxidizers.

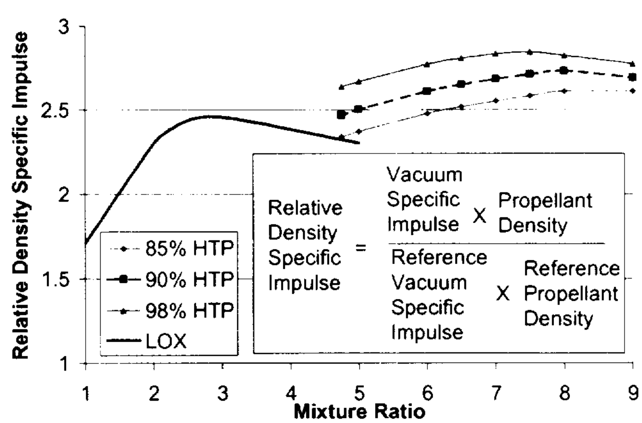

Figure 3: Ideal relative density specific impulse for HC and various oxidizers, (note reference ISP = 417.2 sec, reference density = 20.3 lbm/ft^3).

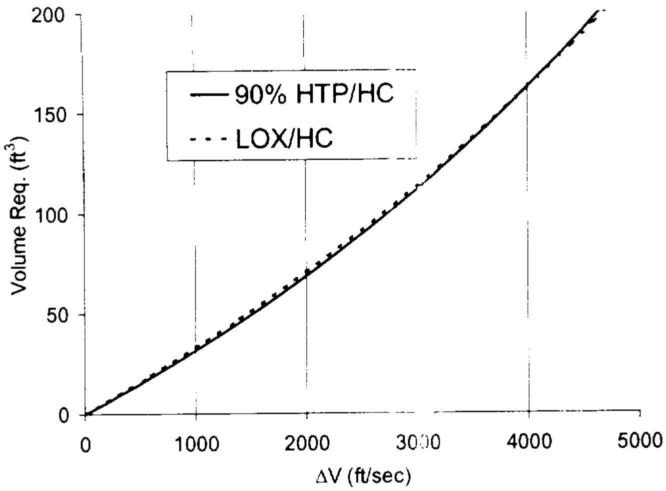

Figure 4: Propellant volume required on X-43B for oxidizer selected plotted against mission ΔV.

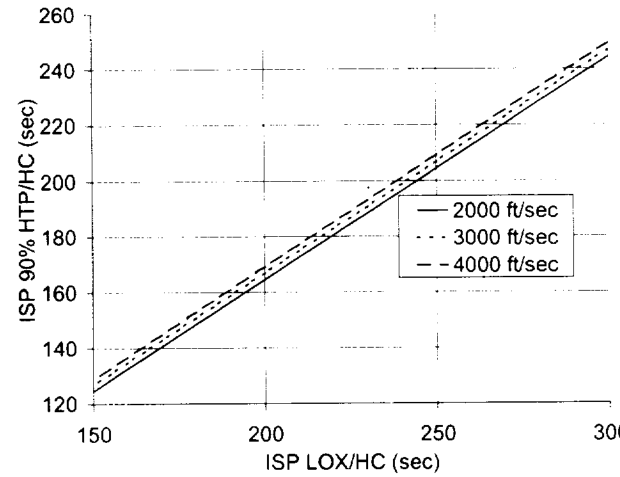

Figure 5: ISP required at given ΔV for equal propellant volume with either oxidizer.

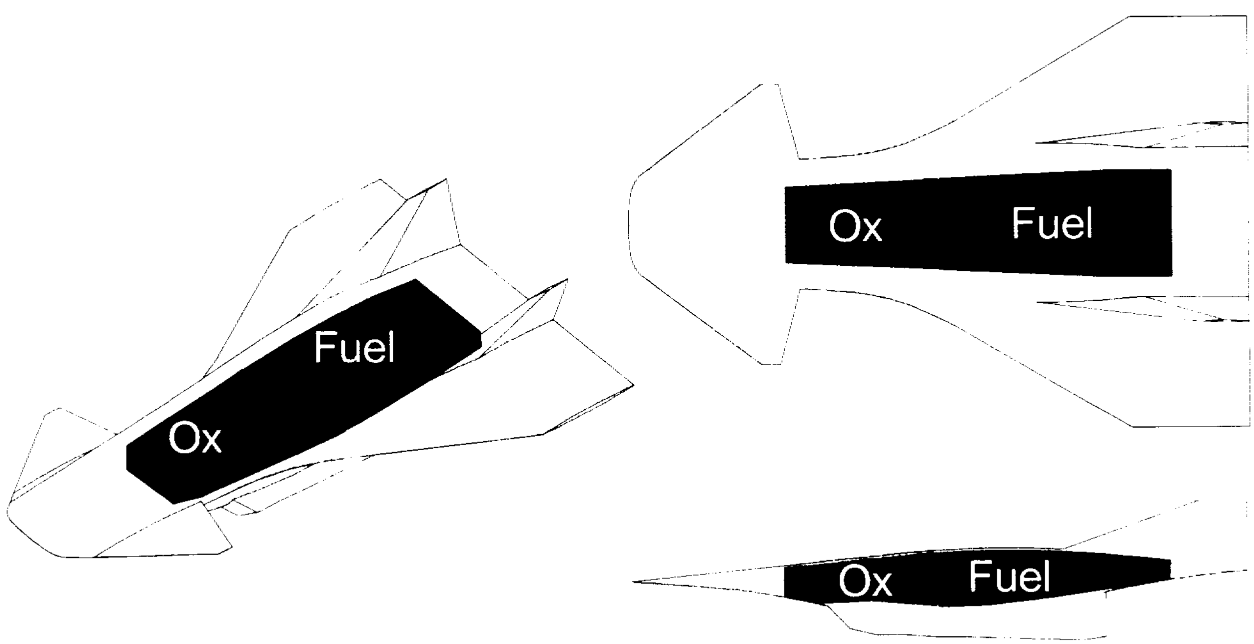

Figure 6: X-43B RBCC vehicle with oxidizer and fuel tank volume / locations shown.

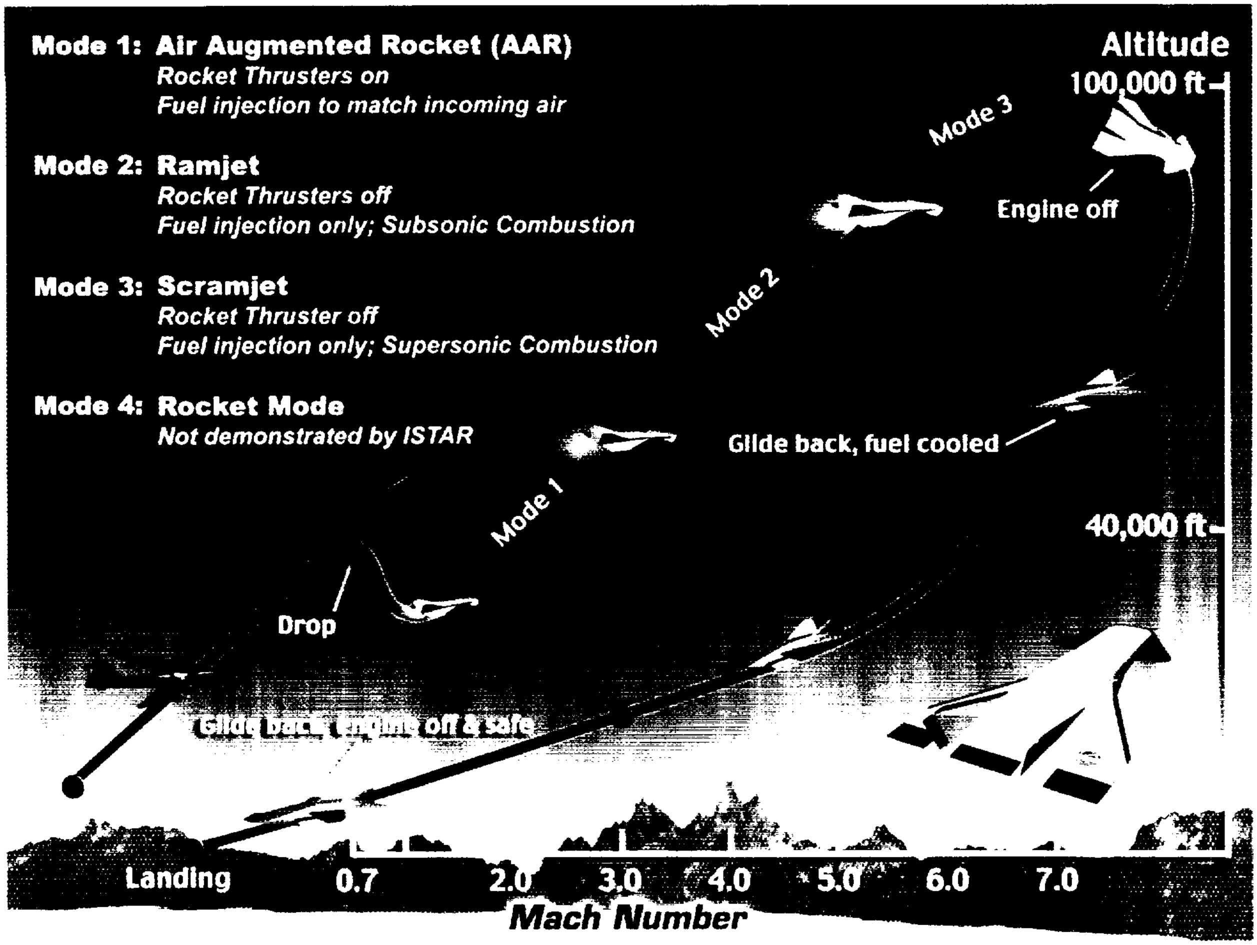

Figure 7: ISTAR Mission Profile.

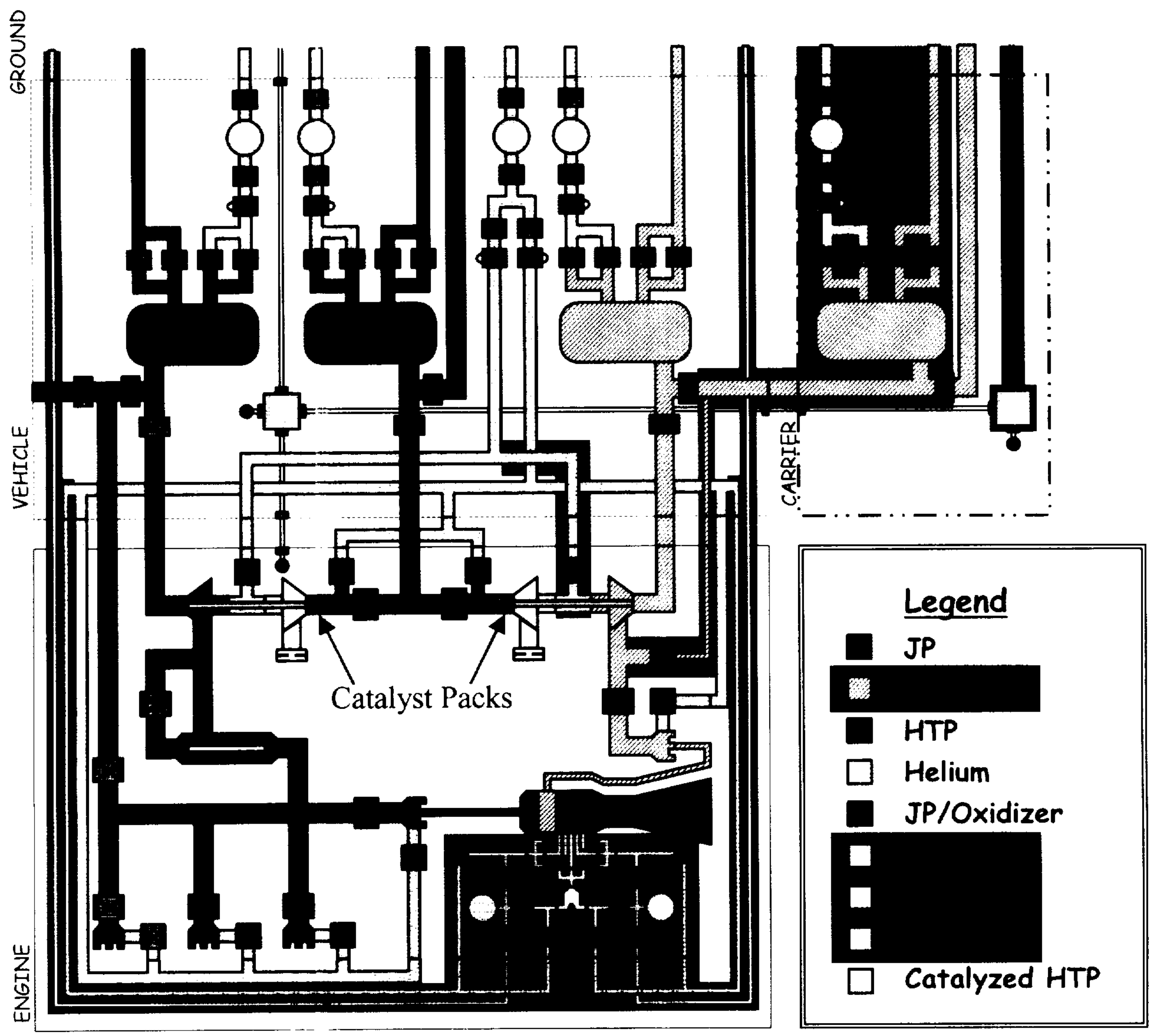

Figure 8: JP-7 / LOX propulsion system including the vehicle and carrier aircraft (B-52) note the red shaded areas are the additional components required by the system for this oxidizer.

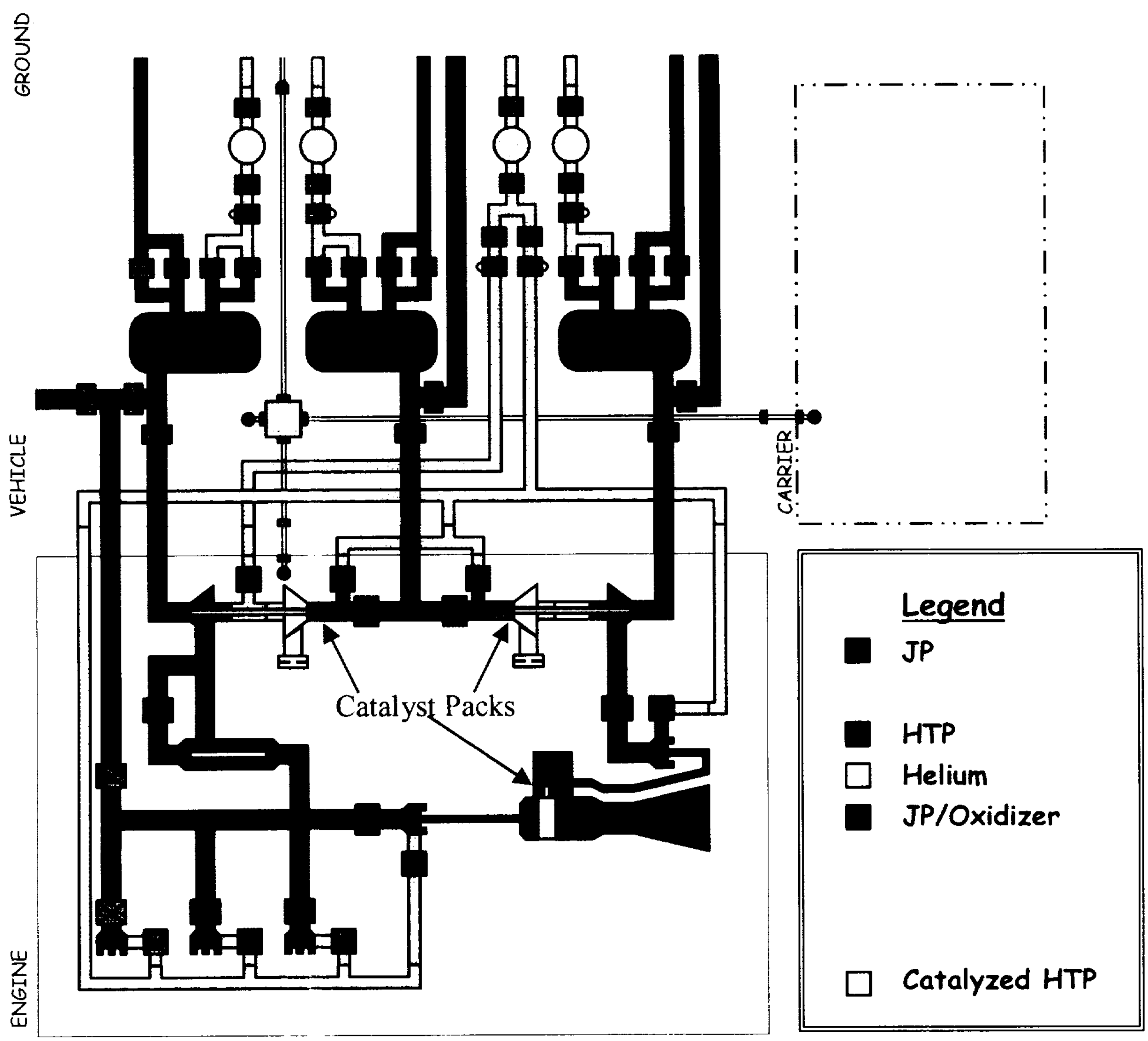

Figure 9: JP-7 / Hydrogen Peroxide propulsion system including the vehicle and carrier aircraft (B-52) note the red shaded areas are the additional components required by the system for this oxidizer.

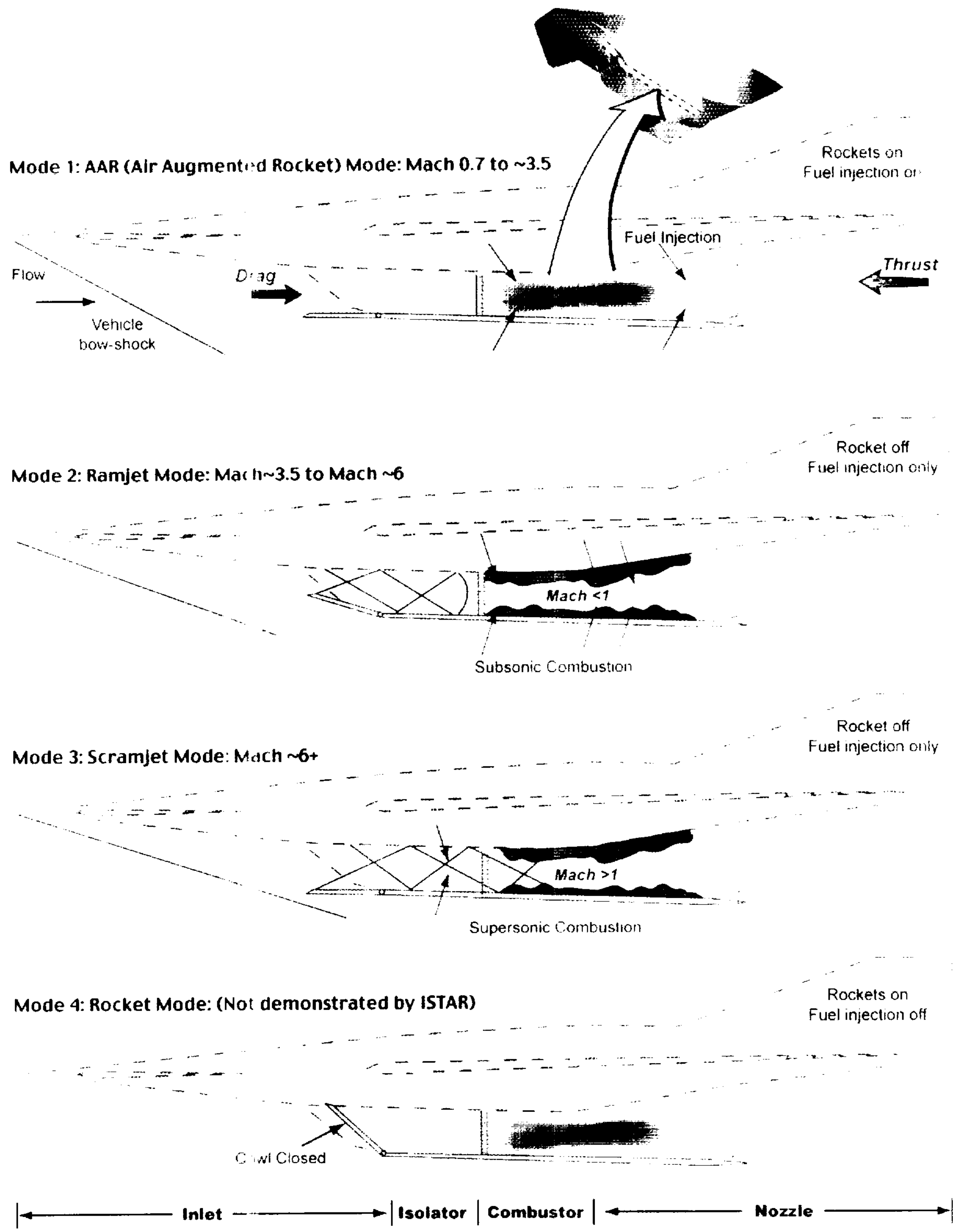

Figure 10: ISTAR engine operating modes.