High Speed Propulsion Cycles

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ACRONYMS

ALS	- air liquefaction system
AspiRE	- aspirating rocket engine
ATRDC	- deeply cooled air turbo rocket
ATREX	- expander air turbo ramjet
CIAM	- Central Institute of Aviation Motors
DC	- direct cooling
DCTJ	- deeply cooled turbojet
GG	- gas generator
GTOW	- gross takeoff weight
ISAS	- Institute of Space and Astronautical Science (Japan)
KLIN	- thermally integrated deeply cooled turbojet and rocket engine
LRE	- liquid rocket engine
MIPCC	- mass injection precompressor cooling
MTB	- MIPCC test bench
NEPP	- NASA Engine Performance Program
RASCAL	- Responsive Access, Small Cargo, Affordable Launch
RBCC	- rocket based combined cycle
SC	- staged combustion
SFC	- second fluid cooling
SLS	- sea level static
SSTO	- single stage to orbit
T/W	- thrust-to-weight ratio
TBCC	- turbine based combined cycle
TFC	- third fluid cooling
TJ	- turbojet
TRL	- technology readiness level
TSTO	- two stage to orbit

1.0 INTRODUCTION: OPPORTUNITIES PROVIDED BY DIFFERENT CYCLES

The present lecture relates to high-speed aircraft and space-launch vehicle propulsion, specifically to methods of the performance and thrust enhancement of turbojet engines and combined-cycle engines (when they are used in such vehicles). Methods for enabling these engines to operate effectively at higher speeds and higher altitudes will be examined.

Missions for trans-atmospheric vehicle include high-speed, long-range transports, military strike and reconnaissance aircraft, as well as orbital space transports, Reference 1. These extreme missions place severe demands on propulsion systems. They must deliver very high performance to efficiently achieve high velocities. They must also function from very low velocity during takeoff at sea level, to orbital velocities beyond the atmosphere.

Trans-atmospheric vehicles generally use a combination of air-breathing and rocket propulsion. Air-breathing systems are valuable since they gather a significant fraction of their propellant from the atmosphere. This reduces the quantity of propellant that must be stored onboard and increases overall vehicle efficiency. Consequently, air-breathing propulsion is often used to the maximum extent possible before exiting the atmosphere, where acceleration to final velocity is under rocket power.

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Turbojet engines are attractive for such applications due to their high efficiency, as well as their operational flexibility. They are particularly valuable during takeoff and landing where their high efficiency at low speeds is critical. However, conventional turbojets are limited in their ability to operate at the high speeds and altitudes associated with trans-atmospheric flight. To extend the velocity and altitude that can be reached using air-breathing engines, a series of combined-cycle approaches have been suggested. These cycles combine the positive attributes of turbojet engines with rocket engines or other air-breathing cycles, including ramjets and scramjets. Some of these cycles are considered in this lecture.

Usually, air-breathing cycles are characterized by relatively low thrust-to-weight ratios. This is acceptable for missions where propellant economy during long periods of atmospheric cruise is important. However, trans-atmospheric and space-launch missions are generally dominated by acceleration requirements where high thrust is often more advantageous than specific impulse. This is due to the increase in gravity and drag losses during extended acceleration periods. Consequently, an increase in engine thrust, even at relatively low specific impulse, can result in decreased overall propellant consumption since acceleration time decreases out of proportion to the increase in propellant flow.

To address the problem of low engine thrust-to-weight, several concepts have been proposed which utilize pre-cooling to increase the density of the inlet air. This increases the engine's power density and permits it to operate at higher Mach numbers. These engines generally use liquefied hydrogen for fuel. Before entering the engine, the cold hydrogen is circulated through heat exchangers ahead of the turbojet inlet to cool the incoming air. A broad class of propulsion systems utilizing air precooling in air/hydrogen precoolers is presented in this lecture.

The propulsion systems addressed in the previous paragraph are those generally using liquid hydrogen fuel. On the other hand, turbine engines based on hydrocarbon fuels can utilize very efficient thrust augmentation concepts which do not require significant modifications of the basic turbomachines, such as MIPCC and rocket-augmented turbine engines.

This lecture also covers examples of other synergetic cycles, namely, second fluid-cooled scramjet engines and third fluid-cooled liquid rocket engines. The common feature of all these synergetic cycles is that the working fluids do more than one job and/or hardware is adjustable for more than one operating mode.

The next section gives a brief overview of eight different synergistic cycles and Section 3 provides greater details of the MIPCC engine.

2.0 OVERVIEW: SYNERGISTIC PROPULSION CYCLES

This lecture covers eight original synergetic cycles including four TBCCs (ATREX, ATRDC, MIPCC, rocket augmented turbine), two RBCCs (KLIN, AspiRE), scramjets and rocket engines. Four of these cycles must have liquid hydrogen as a fuel (ATREX, ATRDC, KLIN, AspiRE), while the others are intended for hydrocarbon fuels or are not specific to the fuel option. Six cycles have been invented or introduced by the author (ATRDC, KLIN, AspiRE, MIPCC, SFC, TFC). The author has been also involved in ATREX cycle development.

2.1 ATREX Cycle

The ATREX cycle was introduced in Japan by Prof. N. Tanatsugu in the 1980s as a propulsion system for the TSTO space plane. ATREX is an original combined cycle, performing like a turbojet at low speed and like a fan-boosted ramjet at hypersonic flight. The ATREX engine is intended as the propulsion system of the fly-back booster of a TSTO space plane. ATREX works as an expander cycle utilizing in turbomachinery the thermal energy regeneratively extracted in both the precooler installed behind the air inlet and the heat exchanger in the combustor. The engine is able to produce effective thrust at flight conditions from SLS up to Mach 6 at 35 km altitude, Reference 2. When the fan inlet temperature is lowered to 160K with a pressure recovery factor of 0.9 in the precooler, the thrust and specific impulse of the ATREX engine increase by factors of 2 and 1.5, respectively, at SLS conditions compared to the non-precooled engine.

ATREX employs a tip turbine configuration in order to reduce turbomachinery weight and size. Figure 1 shows an engine schematic.

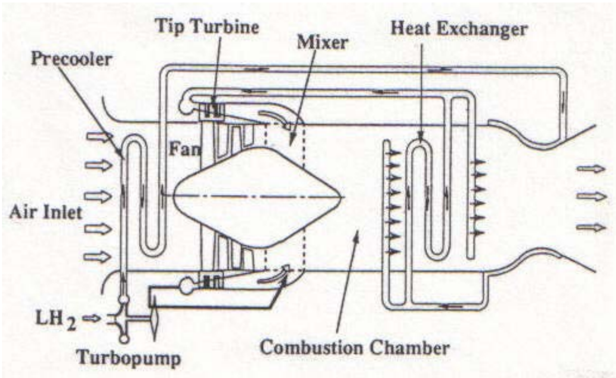


Figure 1: Schematic of the precooled ATREX engine, Ref. 2.

Turbomachinery with the combustor incorporating a hydrogen heater and upstream precooler were successfully demonstrated in various assembled configurations, as well as in separate units, such as air inlet and plug nozzle. Results have been reported in numerous papers. Figure 2 shows a precooled ATREX engine on the test stand.

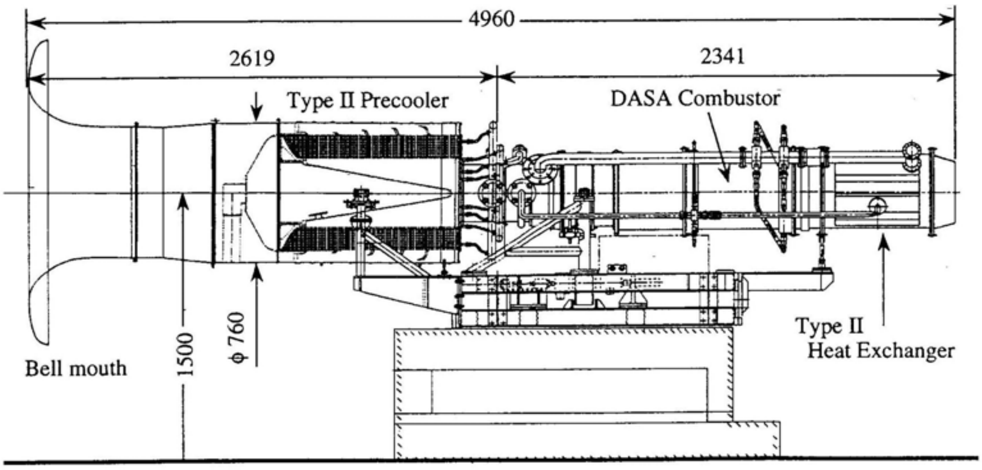
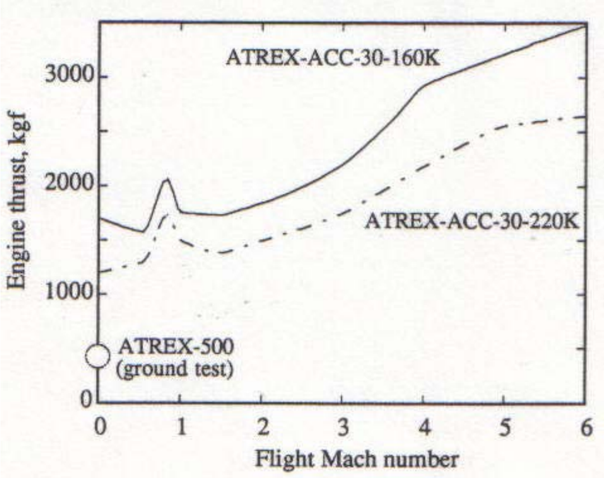


Figure 2: ATREX engine at the test stand, Ref.2.

The effect of air precooling on the engine's flight performance is shown in Figure 3 as a comparison of two levels of air precooling at SLS conditions, namely, T_2=220K and T_2=160K. Performance of the ground tested ATREX-500 engine without inlet air precooling is also shown.



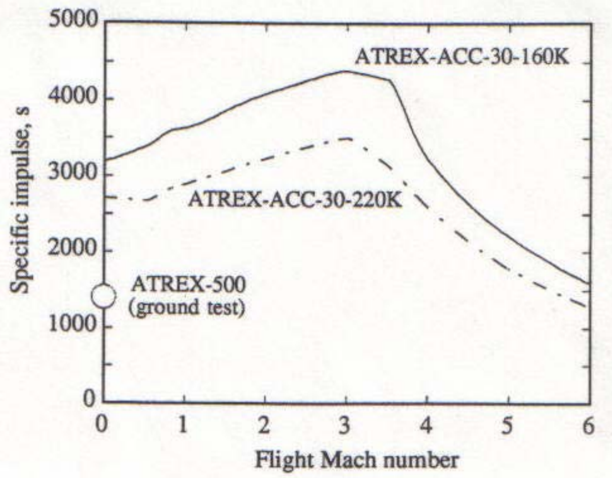


Figure 3: Flight performance of the precooled ATREX engine, Ref. 2. ATREX-ACC-30-160K – advanced ATREX engine using carbon-carbon as a fan structural material, fan diameter 30 cm, precooled to 160K at SLS conditions; ATREX-ACC-30-220K – engine precooled to 220K; ATREX-500 – non-precooled engine with nominal thrust of 500 kgf.

The ATREX engine air/hydrogen precooler is a unique component which was built for the first time on this project. One of the early configurations is shown in Figure 4. One of the issues specific to precooled cycles is precooler icing prevention. Reference 2 provides a discussion of the subject and Reference 3 describes measures the design team successfully demonstrated to prevent icing, namely, spraying a small amount of alcohol in front of the precooler.

Figure 4: Assembly of the quarter of the BARABAN-type precooler. (Picture acquired from the ISAS website http://atrex.isas.ac.jp/).

Major features of the ATREX engine as well as other cycles are summarized in Table 3 in the concluding Section 4.

2.2 ATRDC Cycle

Air precooling permits a significantly higher compression ratio than that utilized in the ATREX cycle, especially if the overall cycle is fuel rich. A good example of such a cycle was examined in CIAM, Russia, Reference 4. This is a deeply cooled air turborocket engine (ATRDC), Figure 5. The engine employs deep air precooling by the use of hydrogen fuel. In addition to its use as a fuel, hydrogen is performing two other jobs, namely, it precools the air and it drives the turbine. Hydrogen is used in an amount significantly higher than required for stoichiometric combustion, characterized by an equivalence ratio $\varepsilon \cong 2$. This permits much greater air cooling than in other precooled cycles and much easier air compression (typical equivalence ratios for ATREX cycle is ε=1.3-1.5). The combustion chamber of the engine operates near stoichiometric since nearly half of the hydrogen flow rate is used to drive the turbine and is then exhausted without combustion.

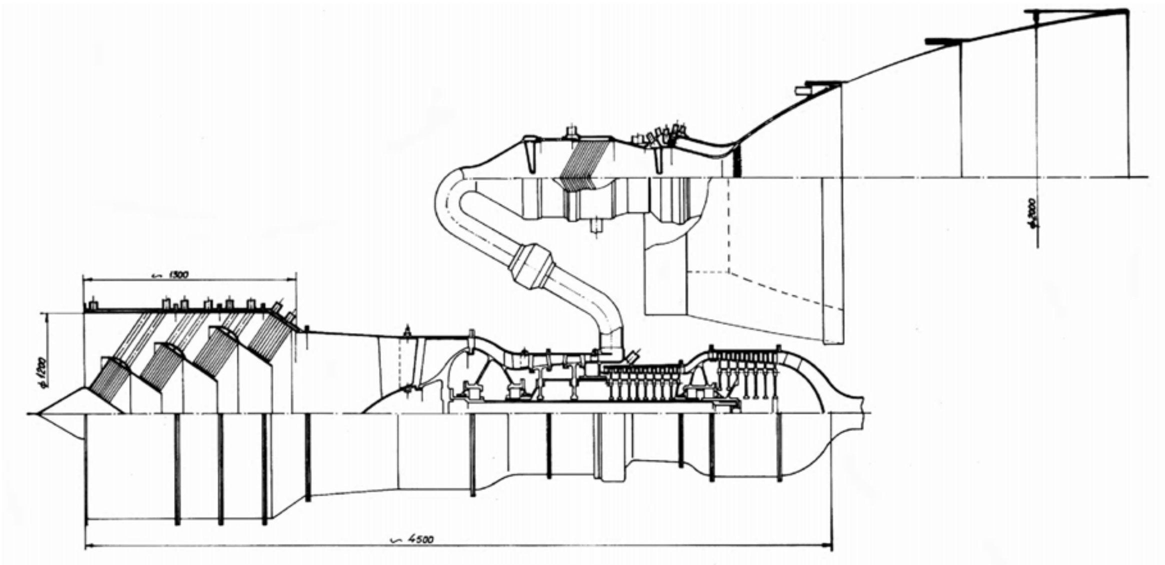


Figure 5: ATRDC engine of 30 ton thrust class, Ref.4.

The ATRDC engine consists of two units which may be located in different parts of the vehicle. The first unit includes the inlet air precooler and turbocompressor, the second includes a double-duct combustion chamber with a two-position bell nozzle and a hydrogen heater located between the two combustion zones. The inner chamber duct operates in airbreathing mode, and the external duct in rocket mode.

At an air temperature upstream of the compressor of T_2=98-112K, the achievable compressor pressure ratio is π_C=20-40. At a pressure ratio of π_C=40, ATRDC provides an average specific impulse in the range of Mach=0to 6 of I_{SP}=2500 s. T/W ratio for the engine constructed with advanced materials has been estimated as high as 18-22. The air precooler is the bulkiest component of the ATRDC cycle accounting for 40% of the total engine weight (without the air inlet).

Since the turbine exhaust is pure hydrogen, the ATRDC engine can be integrated with a ram/scramjet for hydrogen utilization. Such a system is a good example of a synergistic cycle where the hydrogen does four jobs, namely, air precooling in the ATRDC, turbine driving, combustion in the ATRDC chamber, and combustion in the ramjet chamber. Such a propulsion system has better performance as a whole in terms of I_{SP} than the individual components separately. The total specific impulse of the ATRDC and ramjet running simultaneously but which are not thermodynamically connected is :

$$I_{SP} = (1-\xi) I_{SP}^{ATRDC} + \xi I_{SP}^{RAM} , \quad (1)$$

where I_{SP}^{ATRDC} and I_{SP}^{RAM} are values of ATR and ramjet I_{SP};
ξ - fraction of the fuel feeding ramjet.

In the case where ramjet fuel is used for air cooling before the ATRDC engine compressor and to drive the turbine, the total specific impulse is :

$$I_{SP} = I_{SP}^{ATRDC} + \xi I_{SP}^{RAM} - \Delta I , \quad (2)$$

where ΔI is a value of turbine exhaust specific impulse in the case where it is separately exhausted. Depending on flight altitude and speed, the value of ΔI has been estimated to be 2.5-9% of the total ATRDC specific impulse.

When the integrated ATRDC and ramjet are running simultaneously, the total enthalpy of the turbine exhaust is transferred in full to the ramjet flow and ΔI=0. The thrust of the integrated propulsion system is characterized by an equation of the same structure as Eq.2.

Figure 6 shows performance of the separately running ATRDC and ramjet, simultaneously running ATRDC and ramjet without thermodynamic integration as given by Eq.1 at ξ=0.5 and of the propulsion system consisting of the thermodynamically integrated cycles per Eq.2. Total specific impulse of the latter is higher than ramjet I_{SP} by 15-30%.

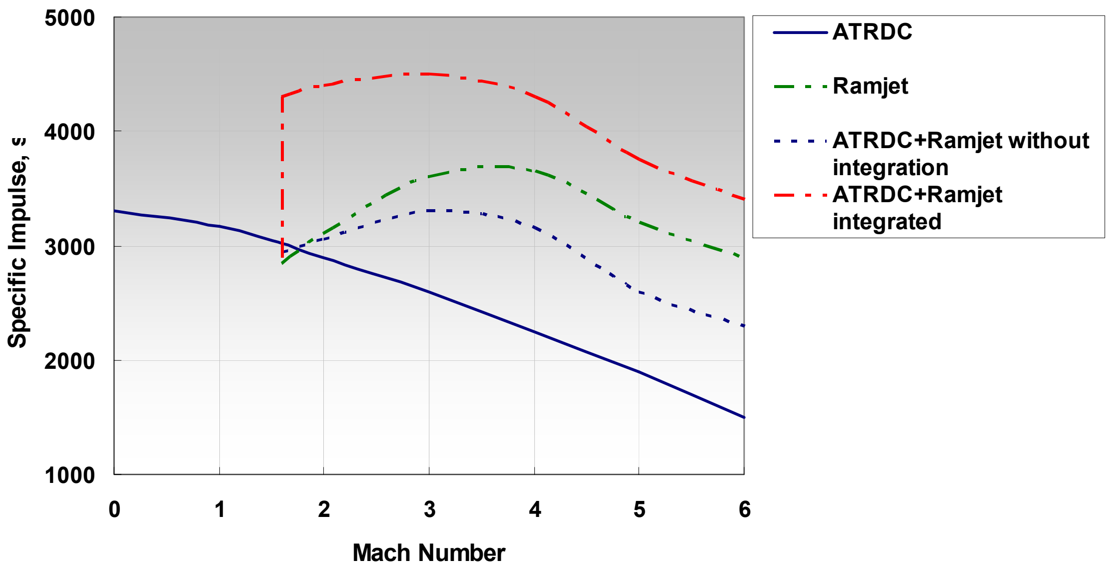


Figure 6: Performance of the separately and simultaneously running engines, Ref.4.

Cycle similar to ATRDC was intended for the British SSTO HOTOL back in 1980s.

2.3 KLIN Cycle

The KLIN cycle is a **thermally integrated,** deeply cooled turbojet (DCTJ) and liquid rocket engine, Reference 5. Thermal integration means that liquid hydrogen fuel for the rocket and turbojet engines is used to deep cool inlet air to 110K at SL and 200-250K at Mach 6. High pressure ratio is attainable with simple and lightweight turbomachinery. This results in high performance and exceptional thrust-to-weight ratio, Reference 6. Schematic of the KLIN cycle is shown in Figure 7.

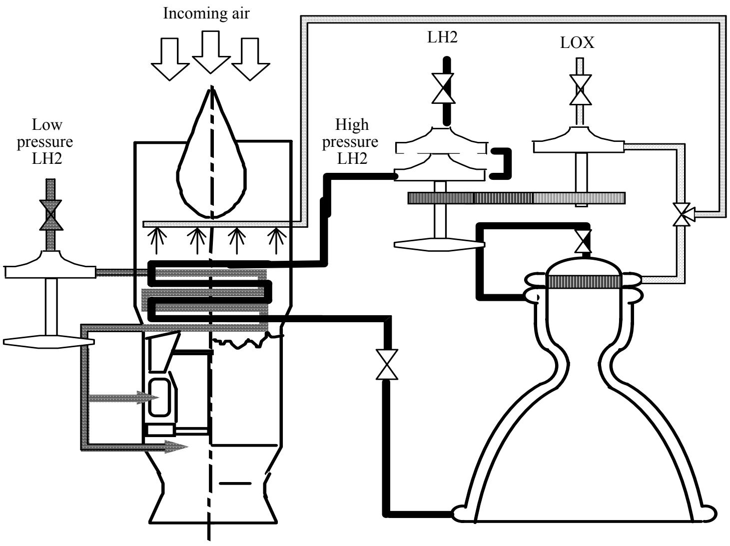


Figure 7: Basic configuration of the KLIN cycle, Ref.6.

The KLIN cycle incorporates several rocket and DCTJ units. All DCTJ units and all or part of rocket units operate from take off. The rocket units may be throttled or even cut off after initial acceleration, returning to full usage when the DCTJ are cut at Mach 6. The DCTJ units will be newly designed turbomachines incorporating a lightweight compressor optimized for low-temperature operation.

For a small launcher, the high performance reliable family of RL10 rocket engines is an appropriate choice. Lowcycle pressure and some features of configuration make the RL10 an ideal candidate for integration into the KLIN Cycle. The RL10 engine uses an expander cycle; therefore, it can be naturally integrated into the KLIN Cycle benefiting from additional hydrogen heating in the DCTJ.

The KLIN Cycle offers very flexible performance characteristics and represents a unique compromise between engine weight and fuel efficiency that provides a high payload capability for the vertical takeoff launcher.

Major parameters characterizing the KLIN cycle are :

KA - the ratio of the airflow and total hydrogen flow (KA=4-12), **KA_0** corresponds to KA at SLS conditions;

ξ - the ratio of the TJ's hydrogen and total hydrogen flow, the hydrogen distribution factor (ξ =0.15-0.4)

DOL_0 - fraction of the TJ thrust in total thrust at SLS (DOL_0=0.25-0.65)

With the KLIN Cycle, various launchers (e.g., SSTO, TSTO with fly back booster), as well as different takeoff and landing scenarios (horizontal or vertical, including a powered descent and landing), are possible.

Advantages of the KLIN Cycle can be summarized as follows:

• simple configuration—ideas such as an air/oxygen heat exchanger for additional air cooling, helium closed loop, or the bypass turbojet were rejected from the beginning of the concept analysis in order to maintain a simple design. Turbomachinery of the simplest possible configuration was considered, and a single spool design with no variable geometry for the compressor was employed. The addition of these features may improve turbojet parameters, but they will also add mass and complexity;

• light weight structure due to the high efficiency of air processing (high specific thrust), "excess" of cooling hydrogen and a low temperature compact compressor;

• high engine thrust-to-weight ratio;

• two to three times higher I_{SP} than for an LRE depending on the mission; and

• known solution for icing problem (LOX injection in front of precooler).

A small reusable vertical takeoff/horizontal landing SSTO launcher that delivers a 330 lb payload to a 220 nmi, 28.5-degree inclination orbit was selected as the reference launcher in the Reference 6 study. The launcher was sized at TGOW 62 tons and a dry weight of 12 tons. The propulsion system configuration for the launcher was defined. It includes seven RL10-type engines with 7.7 tons SLS thrust each and four DCTJs with 6.5 tons SLS thrust each.

The optimum KLIN Cycle corresponds to an initial air-to-hydrogen ration of K_{AO}=6. The thrust-to-weight ratio of this engine was estimated at 33.1, if it is based on an LRE with T/W=43.6. The most simple LRE control law (with full thrust) is proven to be the most efficient for the reference KLIN Cycle, as it provides the highest profile of effective specific impulse for K_{AO}=6.

Figure 8 shows how parameters of the optimized KLIN cycle change along the trajectory and optimum mode sequence. There are three operational modes indicated by circled numbers in Figure 8a :.

Mode 1 (from takeoff to Mach=0.8) corresponds to simultaneous operation of all DCTJ units with oxygen augmentation and all the LRE units;

Mode 2 (in the range of Mach=0.8–6.5) begins after oxygen injection is cut off;

Mode 3 (from Mach 6.5 to orbital speed) a pure rocket mode begins after DCTJ cutoff, and LRE sole operation continues.

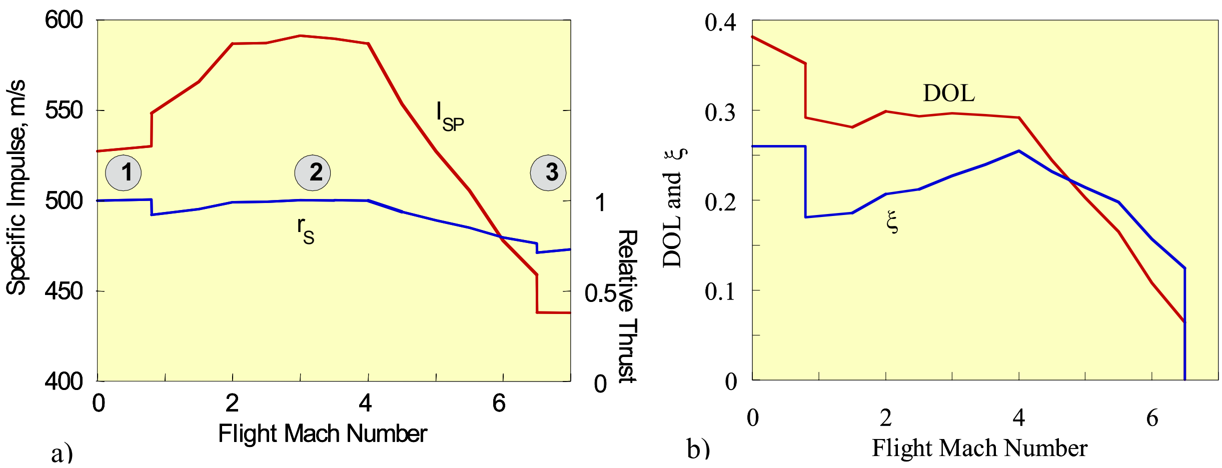


Figure 8. Performance of the reference KLIN cycle, Ref.6.

Optimized operation and performance are shown in Figure 8 where optimization of the launch vehicle is assumed; i.e., minimum GTOW and dry weight at a given payload. A summary of results is shown in Figure 9.

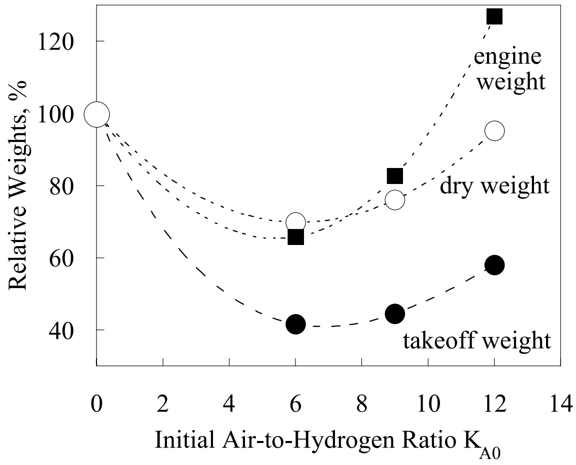


Figure 9. Comparison of the weights of all-rocket launcher and the KLIN Cycle launcher as a functions of the air-to-hydrogen ratio.

The relative GTOW, dry weight, and propulsion system weight are plotted vs. air-to-hydrogen ratio. Corresponding weights of the all-rocket launcher are taken as 100%. As Figure 9 shows, the minimum GTOW would be approximately 40%, and the dry weight and engine weight would be 70% of the corresponding weights of an all-rocket system. These minimums are rather gentle, and one may see that in the range of K_{AO} =5–8, the KLIN Cycle launcher masses do not change much. The minimum launcher dry weight corresponds to the DCTJ contribution in SLS thrust, DOL=30%–45%.

Figure 10 is a schematic of the DCTJ sized for the optimized KLIN cycle. It is a dimensional scheme with major units sized for an SLS thrust of 7 tons. Table 1 gives DCTJ parameters in the main stations as shown in Figure 10, Reference 6.

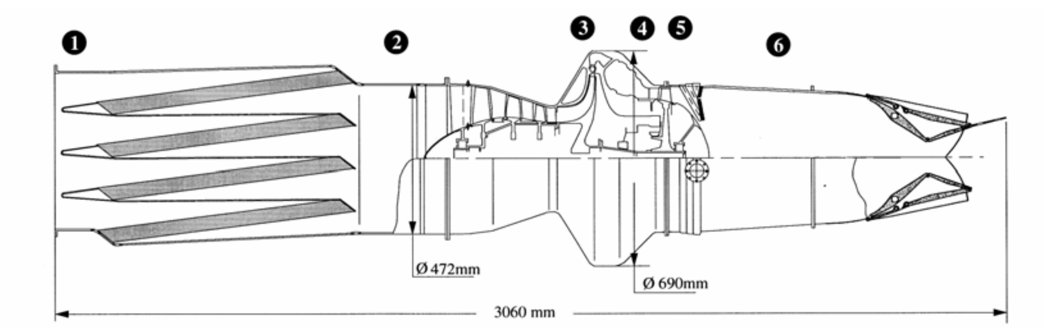


Figure 10. Dimensional scheme of the 7-ton thrust DCTJ, Ref.6.

Table 1. DCTJ parameters at SLS.

Station in Figure 10	Temperature, K	Pressure, bar
1	243	0.92
2	110	0.782
3	331	23.5
4	1451	22.3
5	1229	12.2
6	2505	11.8

2.4 AspiRE Cycle

The KLIN cycle requires development of a new turbocompressor that is a costly and long-duration effort. The next cycle is a logical extension of the KLIN cycle which does not require a turbocompressor. The AspiRE is an original combined cycle, Reference 7, that is capable of operating in two different propulsive modes. For the first air-breathing mode, part of the onboard oxygen is replaced by liquefied intake air prepared in an air/hydrogen heat exchanger/condenser and air/oxygen mixer. At a high Mach number, the engine changes to the second mode, which is a conventional oxygen/hydrogen rocket engine. Both modes use common hardware, namely the fuel and oxidizer turbopumps and the combustor/nozzle assembly. The air inlet and air liquefaction system (ALS) are the only attributes of the airbreather. This synergy is possible because in the combined-cycle mode, three fluids (LH_2, LOX, and Lair) are used in a combination that provides sonic flow in the throat of the nozzle, which is designed for the rocket mode when two fluids (LH_2 and LOX) are used. This results in high cycle performance and an extremely high thrust-to-weight ratio for the combined cycle. Figure 11 shows the AspiRE cycle based on the expander rocket engine.

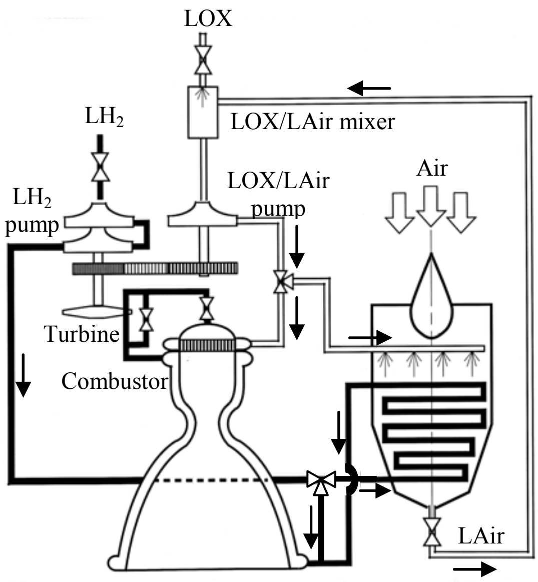


Figure 11. AspiRE cycle schematic.

The AspiRE approach inherits some features of the KLIN cycle and has an even higher degree of cycle "rocketization." In fact, the AspiRE is a more rocket dominated cycle than any other RBCC.

The main issue with operating a combustor/nozzle in two different modes is the matching of the gas flows through the nozzle critical section (throat) and fluid flows through the pumps. References 7 and 8 show how it can be done for the AspiRE cycle. All devices providing high pressure (pumps and their drivers) are used in both the AspiRE and all-rocket modes and this results in a significant increase of the engine thrust-to-weight ratio compared to prior art concepts.

Figures 12 and 13 show a comparison of the I_{SP} and relative thrust for the AspiRE cycle with different initial air liquefaction ratios KA_0 (air liquefaction ratio goes down during acceleration) and pure rocket propulsion. It is seen that the AspiRE cycle I_{SP} is always higher (in sea level conditions, it is 30% higher) than that of a rocket engine until it is switched to the pure rocket mode at Mach 6.5. Substantial thrust deterioration along the trajectory is a typical weakness of the air-breathing accelerators, which results in engine oversize. In the case of the AspiRE cycle, thrust is high at sea level conditions and always higher and nearly constant during acceleration (Figure 13). A combination of these two parameters provides a very favorable effective Isp, which along with an exceptional engine thrust-to-weight ratio, explains the high launcher efficiency. These simulations were conducted with real RL10 engine constraints.

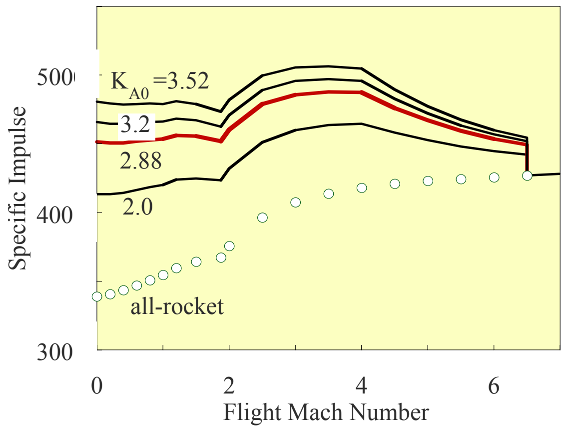


Figure 12. I_{sp} comparison, Ref. 8.

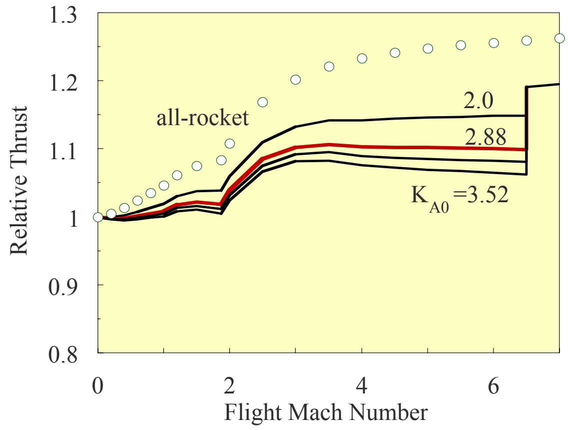


Figure 13. Relative thrust comparison, Ref.8.

The AspiRE cycle makes feasible systems that are not feasible with all-rocket propulsion (small or mid-size reusable SSTO launchers). Small military spacecraft and also quick response suborbital vehicles having global-reach capability are feasible with this technology. In the commercial arena, an AspiRE cycle-based launcher can create a new capability for on-demand, small payload launch services similar to Federal Express® or United Parcel Service®. The AspiRE cycle is also an attractive propulsion option for an International Space Station (ISS) resupply vehicle, Ref.8.

Figure 14 shows a comparison between an AspiRE cycle launcher and an all-rocket launcher in terms of relative GTOW and dry weight (corresponding parameters of the all-rocket launcher are taken as 100%), as a function of the initial air liquefaction ratio K_{A0}.

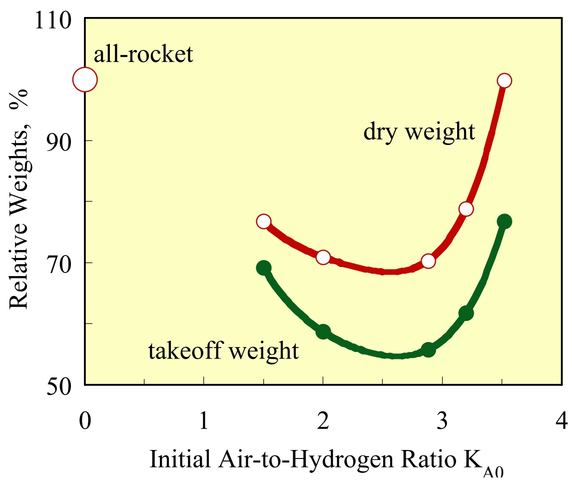


Figure 14. Comparison of the weights of an all-rocket launcher and the AspiRE launcher, Ref.8.

According to Figure 14, the GTOW and dry weight of a small launcher (330-lb payload to 220 nmi orbit) using the AspiRE cycle could be reduced by 45% and 30%, respectively, as compared to the all-rocket launcher. The best launcher efficiency corresponds to an initial air liquefaction ratio K_{A0}=2.0–3.0. At smaller K_{A0}, I_{SP} improvement allows only insignificant system improvement. With K_{A0} increase beyond the indicated range, the disadvantage of the bulky ALS cannot be mitigated by better I_{SP}. Due to low optimal K_{A0}, the ALS/air inlet system is expected to be rather compact.

The lightweight, moderate I_{sp}/high thrust AspiRE cycle provides exceptional efficiency for both vertical and horizontal takeoff small SSTO launchers. Other attractive features include:

- The AspiRE cycle is fully within the current manufacturing capability of the aerospace industry;
- The AspiRE cycle can be based upon existing expander rocket engines of the RL10 class. Incorporation of these engines and their derivatives into the AspiRE cycle can allow them to be used as boosters for small launchers;
- SLS operation of the AspiRE cycle is the most important mode, therefore, feasibility and efficiency of the technology can be proven in the ground demonstration;
- The ALS and triple mixture (air/oxygen/hydrogen) combustor/turbopump assembly can be separately developed and demonstrated.

Figure 15 shows an AspiRE cycle based on the RL50 engine (RL50 configuration from Pratt&Whitney web site).

Figure 15. AspiRE cycle based on RL50 engine, Ref.8.

2.5 MIPCC Engine

The main feature of the previously-discussed cycles is precooling of the incoming air using the heat sink capability of the liquid hydrogen fuel. Recently, hydrogen-fueled propulsion concepts are not as popular as they were back in 80s and 90s, especially for the booster stages of launch vehicles. The next cycle does not require hydrogen fuel. Moreover, this technique is readily applicable to existing turbine engines.

Mass Injected Pre-Compression Cooling (MIPCC) propulsion systems featurea conventional turbojet or turbofan engine as their core propulsion unit with a specially designed fluid injection system that sprays water and/or liquid oxidizer into the engine inlet, Reference 9. MIPCC reduces the incoming air stream temperature and delivers additional mass to the system. It results in an increase in the density of the air stream, permitting more capture. It allows operation at higher than core engine design Mach number and provides enhanced thrust levels at elevated Mach numbers. When properly scheduled, MIPCC allows the engine to operate within its normal operating envelope and use its existing control systems.

Water is an excellent coolant due to its high latent heat of vaporization, therefore, it was chosen for the baseline MIPCC concept. The main purpose of the oxidizer injection is to avoid afterburner blowout at depleted oxygen concentration due to water addition and high altitude. Cryogenic LOX and Lair as well as stable N_2O_4 may also serve as additional coolants. H_2O_2 and N_2O are subject to the exothermic reaction of decomposition which will add enthalpy in the AB, but is not appropriate in front of compressor. A MIPCC engine schematic with all listed injectants is depicted in Figure 16, Reference 8.

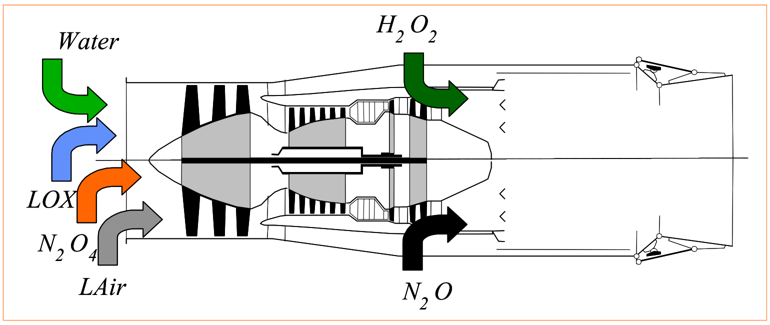


Figure 16. Injection of the different coolants/oxidizers in the MIPCC engine, Reference 8.

Reference 1 emphasizes use of the oxidizer as a coolant in front of turbine engine. The addition of oxygen to the inlet air flow allows the engine to operate at higher altitudes by preventing flameout due to decreasing oxygen.

If liquid air is used as a coolant, inlet temperature in front of the fan/compressor of the regular turbine engine can be kept rather low without any changes in gas composition, which provides comfortable conditions not only for compressor components but also for combustion devices. In addition, cryogenic fluid is easier to evaporate. However, use of the cryogenic oxidizer as a sole coolant leads to rather low I_{SP} at high Mach numbers.

Greater details of the MIPCC cycle as well as a progress in engine development and demonstration are discussed in Section 3.

2.6 Rocket Augmentation for Turboaccelerator

Further turbine engine thrust increase can be provided by oxidizer addition to the afterburner. It provides direct mass addition and also, if hardware permits, can be used for temperature increase in the AB. In this case more fuel will also be consumed. This type of turbine engine modification has been discussed in Reference 8 (see Figure 16). Reference 10 provides further details, Figure 17.

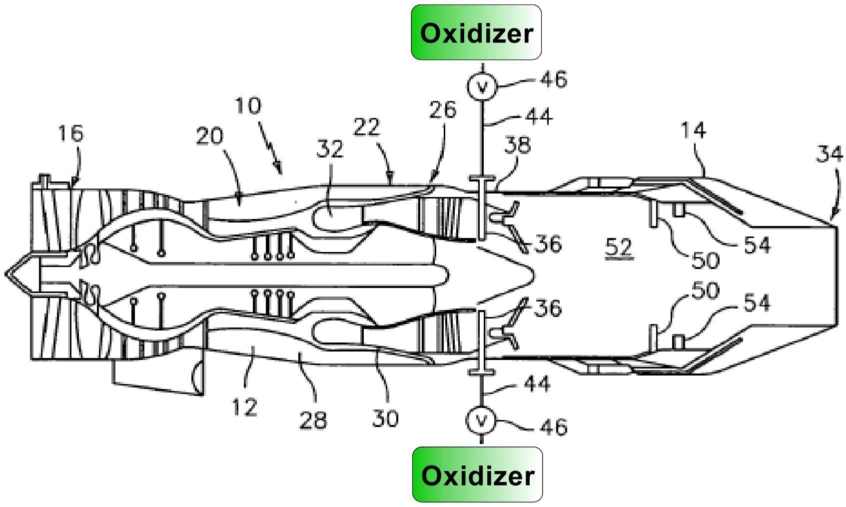


Figure 17. Rocket augmentation for turbine engine, Ref. 10.

According to Reference 10, advanced reusable hypersonic vehicles have a number of challenges to overcome in order to achieve their mission objectives. One of the most significant is the ability to meet the vehicle thrust requirements while also achieving aggressive size/weight/volume goals for the propulsion system. The combination of meeting propulsion thrust requirements while meeting severe packaging restrictions has contributed to the failure of earlier hypersonic vehicle designs. The propulsion system is especially tasked at the transonic pinch point. The transonic pinch is caused by the steep increase in vehicle drag as it transitions from subsonic to supersonic flight. Thus, what is needed is a way to increase engine thrust while holding engine size so that the vehicle could then reach closure (i.e. complete the mission). One of the means of such thrust increase is rocket augmentation of the turbine engine, i.e., injecting oxidizer into hot gases in the augmenter so as to significantly increase the oxygen available for combustion resulting in a significant increase in the thrust force generated by the engine. Hydrogen peroxide is a preferred oxidizer of the authors of Reference 10.

Figure 18 illustrates the impact on the engine thrust with the injection of hydrogen peroxide into the augmenter of the engine. According to Figure 18, 70% thrust increase using rocket augmentation requires eight times increase of the consumables through the engine. This results in eight times lower I_{SP} compared to the basic turbine engine.

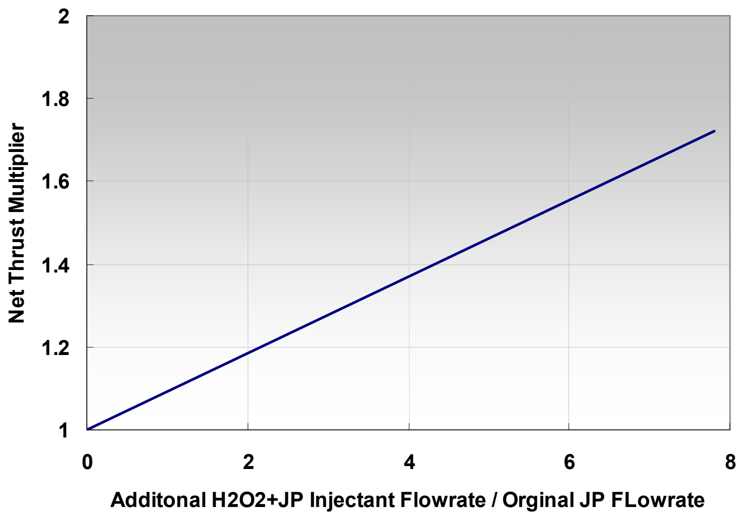


Figure 18. Rocket augmentation impact on the engine thrust, Ref. 10.

2.7 Second Fluid-Cooled Scramjet

A second fluid cooling (SFC) system is the system intended for use in a scramjet engine fueled by heavy hydrocarbons (US Patent pending). Hot elements are cooled with a second non-reactive fluid (N_2, He, etc.), that permits a much higher hot wall temperature, resulting in reduced heat flux to the coolant. This permits an extension of coke-free engine operation to higher Mach numbers. Less heat is transferred to the fuel leading to comfortable and controllable thermal conditions in the SF/fuel heat exchanger. The SF forms a closed-loop Brayton cycle pumping the second fluid through the system, along with pumping of the main fuel and additional optional power generation, Reference 11.

Major advantages of the SF system over a direct cooling (DC) system are extended Mach number, higher I_{SP} (no overfueling) and/or higher thermal margins.

The second fluid of the cooling system travels in a closed Brayton loop. A compressor pumps the second fluid which enters the combustor wall. Within the combustor wall, the second fluid absorbs the heat generated by the combustion process. The heated second fluid exits the combustor wall and is expanded in a turbine which is used to drive the compressor and a fuel pump and provides additional power for the high-speed vehicle. The second fluid then enters a heat exchanger wherein heat is transferred from the second fluid to the fuel. The second fluid then returns to the compressor, closing the Brayton loop. The heated fuel travels from the heat exchanger to the combustor, where it utilized to propel the high-speed vehicle. A high temperature material combustor wall is essential for the SFC concept. Figure 19 shows SFC scramjet architecture.

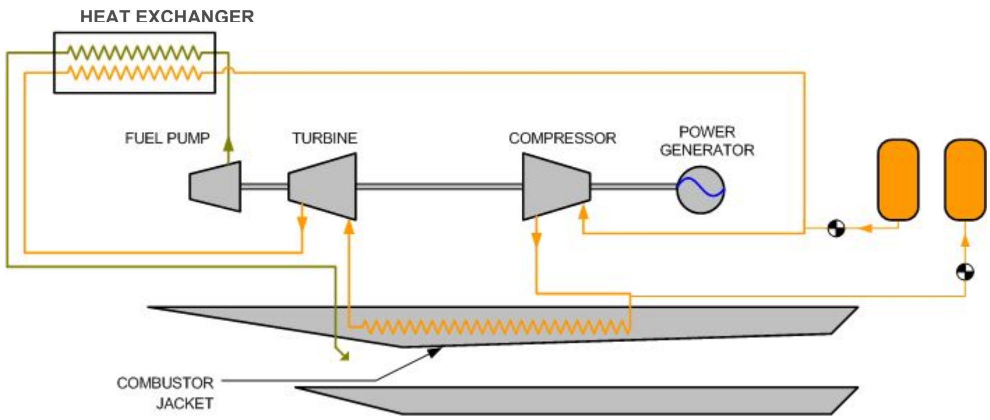


Figure 19. SFC cycle architecture, Ref. 11

In both conventional, direct-cooling techniques and in the SFC technique, fuel ultimately acts as the end heat sink. The ability of the fuel to absorb heat may be described by the "heat sink margin". Before the bulk fuel temperature reaches the coking limit and the wall temperature reaches its limit, the heat sink margin indicates how far the fuel heat sink is from the maximum possible heat sink at the coking limit.

If the fuel temperature at stoichiometric condition reaches the fuel coking limit, or if the wall reaches its material limit, extra fuel should be added for cooling purposes, even if it is excessive for the combustion process. The heat sink margin will be negative to reflect the need for engine overfueling. Without engine overfueling, engine operation is not permitted for long periods of time, as fuel coking will occur. With the use of overfueling, operation is possible at the expense of engine fuel efficiency.

Thus, fuel heat sink margin may be presented in two forms:

- before temperatures hit their limits at stoichiometric fuel/air ratio, fuel heat sink margin is

$$\delta = 1 - \frac{Q_X}{Q_{\max}}\varepsilon \quad (\delta \geq 0) \qquad (3)$$

- after temperatures exceed their limits at stoichiometric conditions

$$\delta = 1 - \varepsilon \quad (\delta < 0) \qquad (4)$$

where δ - fuel heat sink margin;
Q_X - heat absorbed by the fuel;
Q_{max} - maximum heat to be absorbed by the fuel at coking limit;
ε - equivalence mixture ratio.

As an illustrative example, Figure 20 shows a comparison of the fuel heat sink margin for direct cooling with endothermic hydrocarbon fuel and for second fluid cooling, where the same fuel is the end heat sink but nitrogen is used as a second fluid.

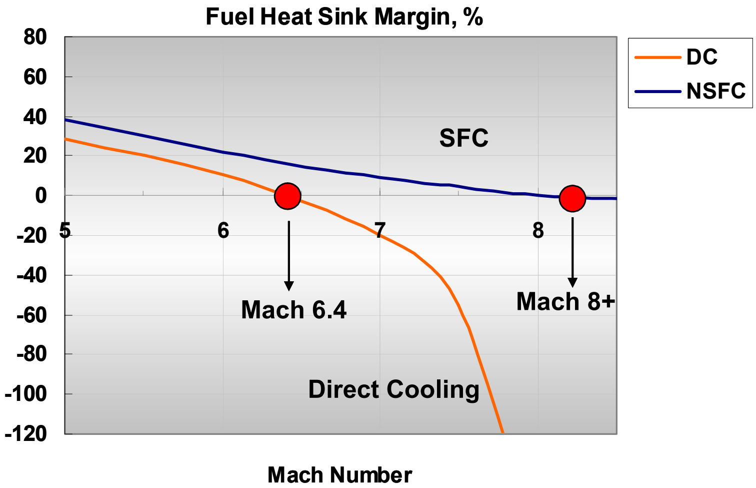


Figure 20. Fuel heat sink margins for DC and SFC methods, Ref. 11.

It is seen that direct cooling can provide scramjet operation up to Mach 6.4 with positive heat sink margin, i.e., without overfueling. Prohibitive overfueling, characterized by a negative fuel heat sink margin of less than minus 100%, is required to reach Mach 8. An SFC system extends the stoichiometric operation to Mach 8+ and narrow positive fuel heat sink margin is still available at Mach 8, as shown. The gentle slope of the second fluid cooling curve, as compared to the direct cooling curve, shown in Figure 20, enables further flight velocity increase to speeds over Mach 8.0 with moderate engine overfueling.

In this manner, efficient cruise flight engine operation at Mach 8, where the fuel flow rate required for combustion is lower than during acceleration, may be enabled by SFC technology.

2.8 TFC Rocket Engine

The TFC cycle is a logical extension of the SFC cycle approach on rocket engines, which also utilizes fluid other than fuel and oxidizer (third fluid) to cool the combustion chamber and drive the turbopump. Unlike other cycle presented here, both SFC and TFC cycles minimize the amount of work done by engine consumables through the introduction of the special fluid which is doing all internal work while forming a closed loop.

The TFC concept is a novel expander rocket engine that uses a third fluid as the combustor coolant and the turbine driver, Reference 12. The third fluid forms a closed-loop Rankine cycle permitting much higher available turbine expansion ratios than other closed rocket engine cycles.

The TFC rocket engine combines advantages of all three major pump-fed cycles: high available turbine pressure ratio of the gas generator cycle, full flow through the chamber typical for the staged combustion and expander cycles, high chamber pressure typical for the staged combustion cycle, and no preburner as with the expander cycle.

This unique set of the features permits the TFC cycle to be more efficient compared to the gas-generator and expander cycles in terms of I_{SP} and thrust-to-weight ratio, to exceed or match the staged combustion cycle in terms of thrust-to-weight ratio and to be more reliable than the staged combustion cycle due to significantly lower maximum cycle pressure.

The TFC cycle is applicable to both LOX/LH2 and LOX/HC rocket engines. Here, LOX/LH2 engine will be briefly discussed. More details can be found in Reference 13.

In modern LOX/LH2 rocket engines, hydrogen serves as the combustor coolant and sole turbine driving fluid (in the expander cycle) or part of the turbine gas (in gas generator, tap-off, and staged combustion topping cycles) prior to entering the combustor. To develop higher thrust in a rocket engine, higher pressures in the combustor are required, since output thrust is directly related to combustor pressure and this, in turn, requires higher propellant flow rate.

In both coolant and turbine driver applications, the hydrogen flow loses a significant amount of the pressure generated by the pump. In both the staged combustion SSME and expander RL10 engines, hydrogen pressure downstream of the pump is more than two-fold higher than the pressure in the combustor. However, hydrogen is the most difficult liquid to pump due to its very low density (approximately 70 kg/m3). This leads to lower than desired combustor pressures and explains the complexity of liquid hydrogen turbomachines, which particularly include the number of pump stages and very high mechanical load on feeding systems that reduces engine reliability and, finally, prevent the development of truly reusable engines.

The problem of oversized LH2 turbopumps can be resolved when a third fluid is employed as a coolant and turbine driver.

The TFC configuration, per Reference 12 and Figure 21, includes a typical engine assembly constructed of an injector 1, combustor 2, and nozzle 3. The combustor 2 and nozzle 3 form a nozzle and combustor assembly 4. Fuel, such as liquid hydrogen, and an oxidizer, such as liquid oxygen, are fed from supply tanks 5 and 6, respectively, to the injector 1. These fuel and oxidizer components, referred to as propellants, are mixed and fed to the combustor 2 wherein they are burned to produce hot gas which is ejected from the nozzle 3 to propel the vehicle. The fuel is fed to the injector 1 by a turbine-driven fuel pump 7, while the oxidizer is fed to the injector 1 by a turbine-driven oxidizer pump 8.

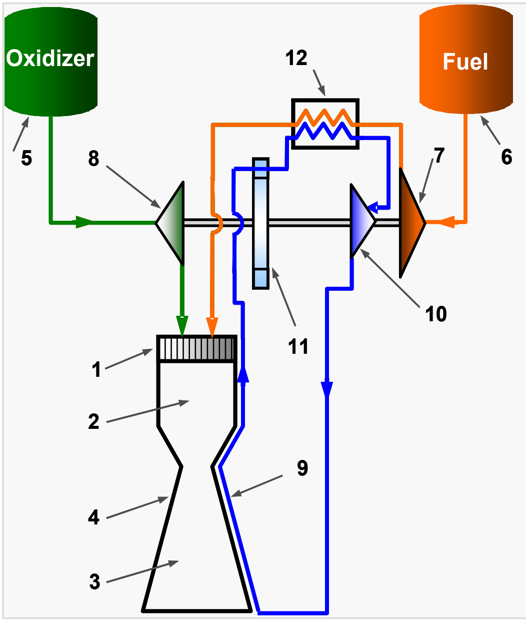


Figure 21. TFC engine flow diagram, Ref.12.

In the TFC engine, the nozzle and combustor assembly 4 is cooled by a circulating coolant such as water, methanol, ethanol, or liquid having equivalent properties, or mixtures thereof. The coolant is circulated through a jacket 9 enclosing the nozzle and combustor assembly 4 by a turbine-driven coolant pump 10. As the coolant circulates through the jacket 9, it is heated and vaporized forming steam or a vapor or gaseous-phase fluid. This vaporor gaseous-phase fluid is fed to the turbine 11 for driving the oxidizer pump 8, coolant pump 10, and the fuel pump 7.

The coolant vapor expands and is partially condensed in the turbine 11 and the turbine temperatures are reduced. The work of driving the turbines is produced by the expansion, temperature reduction, and partial condensation of the coolant vapors. The condensation process is completed in a heat exchanger 12 for exchanging heat between the coolant vapor and the incoming propellant, such as the liquid fuel or oxidizer or both. The coolant vapor condenses to heat the propellant, thereby returning the heat removed by the coolant from the combustor to the propellant fed to the injector 1. The fuel pump, oxidizer pump, water pump and turbine are mounted on one shaft in the particular case shown in Figure 21.

A third-fluid closed loop comprising turbomachinery, combustor jacket (shown as a heat exchanger), and third fluid/fuel heat exchanger is shown in Figure 22.

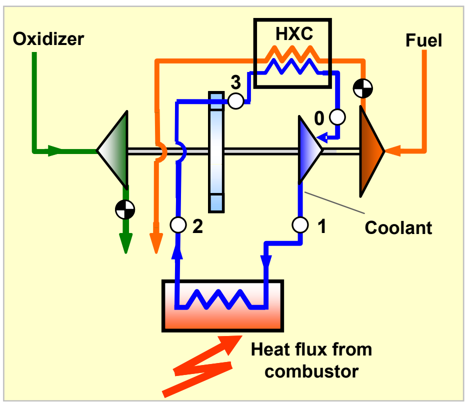


Figure 22. Closed-loop Rankine cycle.

A well-known thermodynamic cycle, the Rankine cycle, for the flow of coolant such as water, is shown in Figure 23. The diagram shows the stations of the water flow path shown previously in Figure 22; namely water pump inlet 0, water pump exit 1, combustor jacket exit 2, turbine exit 3, heat exchanger exit 4 with the same parameters as in pump inlet 0. Line S defines the water saturation line. Points a and b correspond to intermediate stages of the water heating; point a corresponds to the beginning of the water evaporation in the combustor jacket, point b corresponds to the complete water evaporation in the combustor jacket. Point 3s corresponds to the ideal process of isentropic steam expansion (S=const) in the turbine. The solid lines in Figure 23 correspond to the following stages in the process:

0-1 - water pumping;
1-a-b-2 - water heating, evaporation, and steam heating in the combustor jacket;
2-3 - steam expansion in the turbine (2-3s - ideal expansion);
3-0 - steam condensation in the heat exchanger.

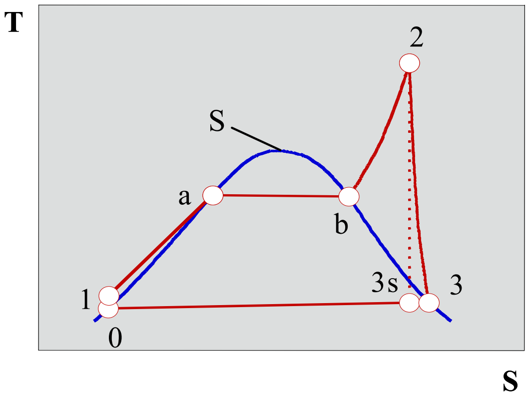


Figure 23. Rankine cycle T-S diagram (dry stream).

In order to reduce the size of the heat exchanger, steam or vapor can be partially condensed in the turbine. The more steam or vapor that is condensed in the turbine, the smaller the heat exchanger required. The literature on steam power generation turbines, for example Reference 14, recommends moisture content in the turbine be no more than 12%, since higher values causeturbine blade erosion. This recommendation is valid for steam power turbines with projected lifetimes of tens of thousands of hours. The expected lifetime for even reusable rocket engines is not likely to exceed tens of hours; therefore, appropriate amounts of moisture in the turbine exit can be expected to be significantly higher than 12%. A T-S diagram for the process with wet steam at the turbine exit is shown in Figure 24.

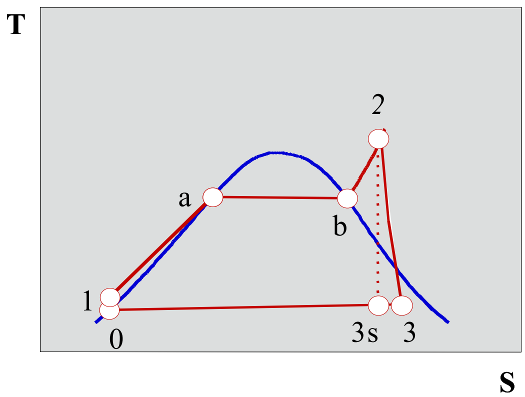


Figure 24. Rankine cycle T-S diagram (wet stream).

Major benefits of the TFC technology applied to the LOX/LH_2 engines are significantly higher (50-65%) engine thrust-to-weight ratio and feasibility of higher combustor pressures. A preliminary comparison with SSME shows that at the same combustor pressure, 34% of the structural weight saving can be expected. Due to significantly lower cycle pressures, TFC technology may be a key to rocket engine reusability.

Other significant advantages include the fact that compared to the staged combustion cycles of the SSME type, which utilize 3 combustion devices (2 preburners and a main combustor), the TFC eliminates 2 of these 3 combustion devices with an accompanying weight savings and no loss in performance. As a result, development cost and time savings can be expected. This is also true for the LOX/HC rocket engines.

The TFC configuration is a promising choice for the LOX/HC engine because it permits:

- The elimination of the preburner and associated systems
- Low turbine temperature
- Low maximum cycle pressure (on the level of the chamber pressure).

The TFC engine can be used for both booster rocket and upper stage rocket applications.

3.0 MASS INJECTION PRECOMPRESSOR COOLED TURBINE ENGINE

The MIPCC engine is listed among the cycles presented as an example of the efficient accelerator engine. Section 2.5 provides initial cycle information. This section is devoted entirely to the MIPCC cycle and shows the status of its development according to Reference 15.

3.1 Baseline Flight Trajectory

MIPCC engines can be utilized for launchers of different size and configuration. The basic operational concept developed for the small reusable launcher called RASCAL is depicted in Figure 25. This figure shows an airbreathing vehicle (either manned or unmanned) taking off from a conventional runway. The vehicle climbs to its loiter altitude, using a conventional airbreathing turbofan propulsion system. At this point, the vehicle begins a “zoom maneuver”, taking advantage of liquid injection ahead of the first-stage turbofan compressor. The liquid injection allows the vehicle to go to a much higher altitude and velocity than would be possible with an unmodified turbofan. At an appropriate point in the trajectory, the upper stages are released from the first stage. Once the upper stages of the vehicle have safely cleared the first stage, the first stage can return to the Earth for reuse.

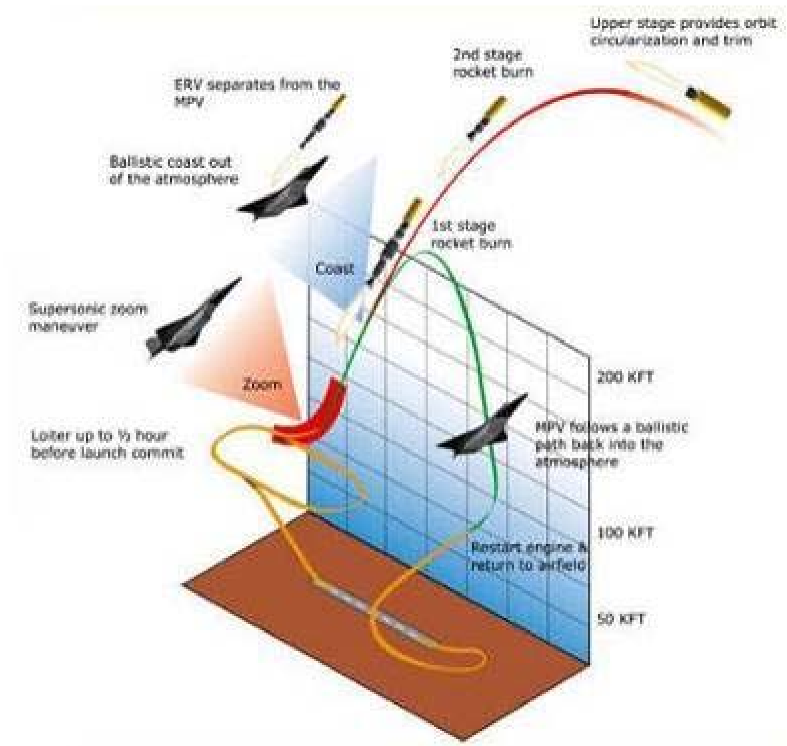


Figure 25. Baseline operational concept, Ref. 15.

3.2 Baseline MIPCC Concept

MIPCC is intended to offset performance losses incurred as an aircraft accelerates and the density of the air flowing into the inlet decreases. The engine subsystem adds water and liquid oxygen to the airflow in the inlet, reducing its temperature hundreds of degrees. The result is twofold: First, it places less stress on the engine and increases its durability because of the lowered temperatures. Second, and from a performance perspective the more important, mass flow is increased due to the increased density.

The baseline MIPCC configuration is shown in Figure 26. The following are the major principles applied by the MIPCC concept:

• The existing turbofan engine is utilized with water and LOX injection ahead of the fan. No modifications of the basic TF engine are planned, except for minor upgrades and adjustments of the control system;

• The inlet capture area is designed to allow the air flow rate required for turbomachinery to generate the maximum thrust. Only at the end of zoom at high altitude inlet limits the thrust restricting the air flow to the amount less than maximum allowed by turbomachinery;

• Injection is initiated in the transonic region;

• During the acceleration mode, the MIPCC engine develops maximum possible thrust. It reaches approximately 200% of the SLS thrust of the basic TF just before the zoom climb;

• Water is injected in an amount not to exceed saturation content;

• LOX is injected in the amount required to maintain normal molar oxygen concentration of 20.9% in the air/water/oxygen mixture. This concentration may be exceeded when additional thrust is required and small amounts of water are replaced by larger amounts of LOX (OXYBOOST mode);

• All applied pressure, temperature, geometry, and RPM limiters of the basic TF engine have been respected.

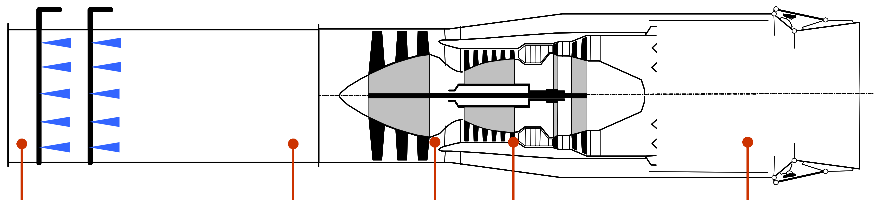


Figure 26. Baseline MIPCC engine configuration.

Water provides the better coolant, but the MIPCC system also is designed to introduce liquid oxygen to maintain combustion stability for the afterburner and for mass addition. Water and LOX injection results in significant thrust increase due to significantly higher mass flow rate and available expansion ratio over the nozzle.

Comparison of the key parameters of the basic F100 engine and the same engine equipped with the MIPCC system at Mach 2.25 are shown in Table 2. Stations are given in Figure 26. Note the difference in flow rates (MIPCC station) and afterburner pressure. The resulting MIPCC engine thrust is more than doubled at this particular point. This difference increases at higher Mach numbers until the basic turbofan becomes non-operable primarily due to temperature limitations.

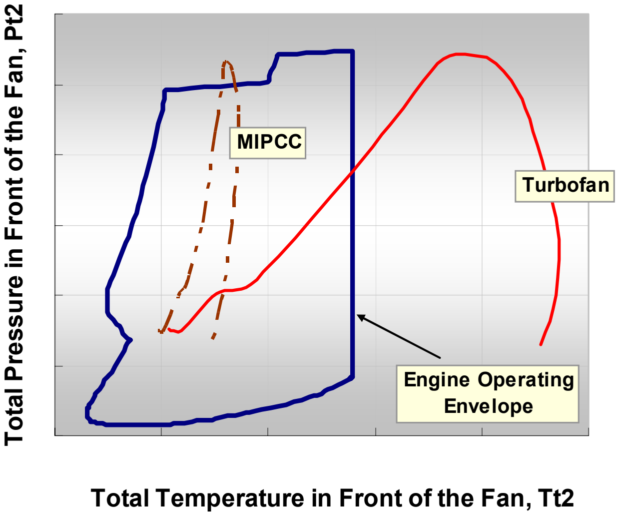
Table 2. Engine parameters comparison at Mach 2.25, Alt 46000 ft.

Station	Parameter	MIPCC F100 over Basic F100
Inlet	Flow rate	1.66
	Temperature	1.00
	Pressure,	1.00
MIPCC	Flow rate	**1.73**
	Temperature	0.74
	Pressure	0.98
HP compressor	Temperature	0.92
	Pressure	2.15
Afterburner	Pressure	**2.05**
Exit	Net Uninstalled Thrust	**2.01**

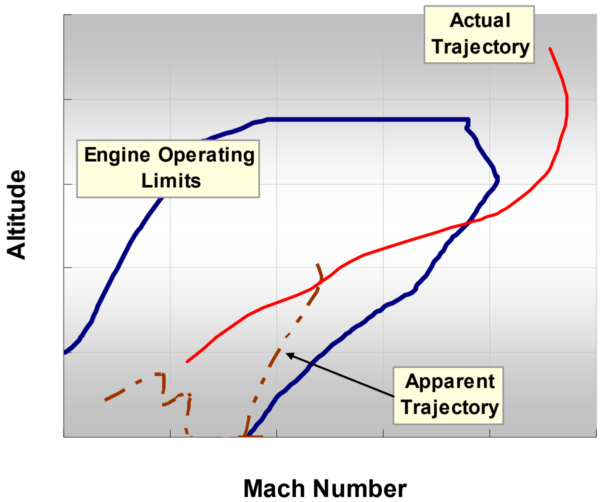
3.3 MIPCC Operation

The key to the MIPCC concept is that the engine does not recognize it is flying at extreme Mach numbers and altitudes. Two of the major engine control inputs are total pressure and total temperature upstream of the fan – Pt2 and Tt2. These parameters indicate the speed and altitude at which the engine is flying. The MIPCC engine fan sees significantly lower temperature compared to the basic TF flying the same Mach number and nearly the same pressure. The solid line in Figure 27 shows the Pt2-Tt2 relationship for the basic TF engine flying a launch trajectory. In the case of a MIPCC engine, this line shows the pressure-temperature relationship upstream of the MIPCC system. The dashed line shows the same relationship upstream of the fan of the MIPCC engine. The latter is entirely within the operating envelope of the basic TF, also shown in Figure 27a.

Figure 27b shows actual flight trajectory and apparent trajectory according to Pt2 and Tt2 sensor readings upstream of the MIPCC engine fan. The maximum velocity seen by the engine (apparent trajectory) is two times lower than the maximum velocity for the actual trajectory, due to the temperature reduction. The maximum altitude for the apparent trajectory is also twice as low as the maximum altitude for the actual trajectory. Therefore, despite the extreme actual flight conditions, the engine with mass addition upstream of the fan always stays within the Mach/altitude and Pt2/Tt2 envelopes prescribed for the basic turbine engine.



a)



b)

Figure 27. MIPCC engine operating conditions: a) Tt2-Pt2 operating envelope; b) Mach number-Altitude profiles.

The trajectory shown is a result of the interactive MIPCC engine/vehicle analysis. The goal of this effort was to select the baseline MIPCC engine, size the MIPCC injection system, and identify the required modifications to the basic engine control system. The engine thrust, fuel efficiency characteristics and schedules for all required internal engine parameters were obtained in this analysis.

3.4 MIPCC Configuration

Figure 28 shows the MIPCC system integrated with an engine of F100 size. The configuration shown was sized after accomplishing the first series of MIPCC engine tests and was coordinated with the airframe integrator. The MIPCC system includes one LOX injection plane and two water injection planes. The distance between the last water injection plane and the engine face was defined as the minimum required for evaporation of the substantial amounts of water.

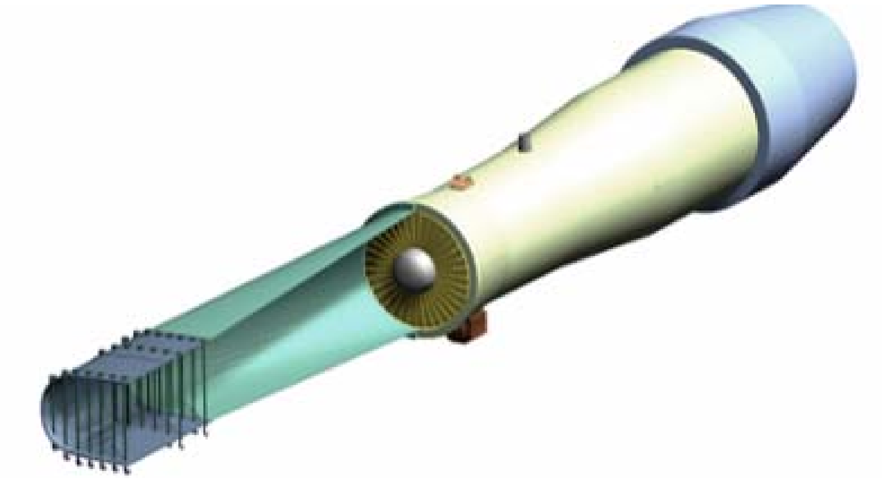


Figure 28. MIPCC engine configuration.

For the MIPCC system demonstration and verification of the major principles involved, a MIPCC duct has been designed, built and tested at the MTB. Configuration of the demonstration MIPCC duct is shown in Figure 29. As in the flight version of the duct, there is one plane of LOX injector bars and two planes of water injector bars. Each injector bar could be individually switched on or off.

Figure 29. Demonstration MIPCC duct.

3.5 MIPCC Test Bench (MTB)

Ground testing of the MIPCC system was conducted as an early validation of this innovative propulsion cycle. A group of companies under DARPA sponsorship built the MIPCC Test Bench (MTB) at the Mojave Airport, Mojave, CA. The facility is intended to simulate flight environments up to Mach 4 on the ground. A flow diagram of the test setup is shown in Figure 30.

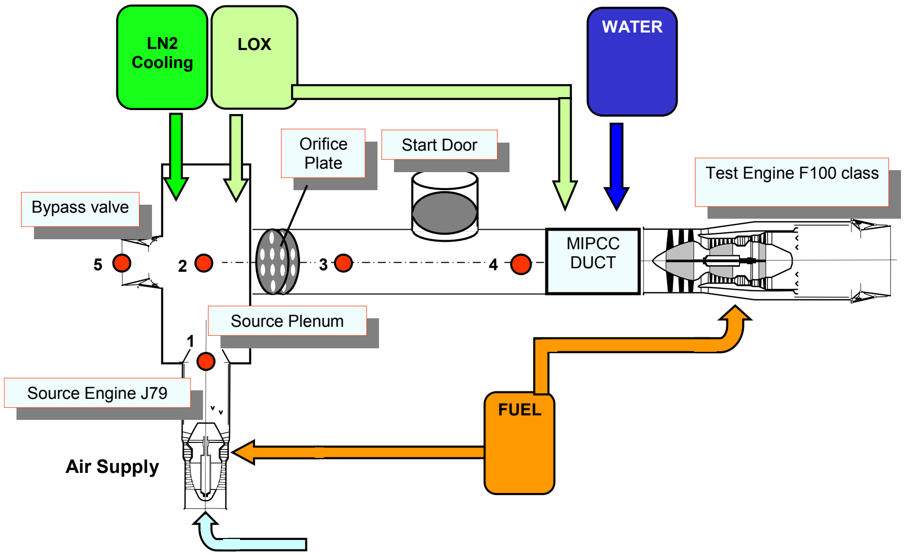


Figure 30. MTB flow diagram.

Heated air is provided by a surplus General Electric J79 turbojet, which exhausts into a source plenum. Air leaving the plenum is conditioned to the proper enthalpy and oxygen level by the addition of liquid nitrogen and liquid oxygen. Air flow through the test engine is controlled by throttling of the J79 source engine, mass addition through LOX injection upstream of the source engine, and source engine exhaust bypass. The air temperature ahead of the test engine is controlled through the J79 engine setting and LN2 injection into the source plenum. The air pressure upstream of the test engine is controlled through LAir injection ahead of the source engine, as well as source engine throttling. The oxygen concentration in the test engine is controlled through makeup LOX injection. A photograph of the facility conveys the scale, Figure 31.

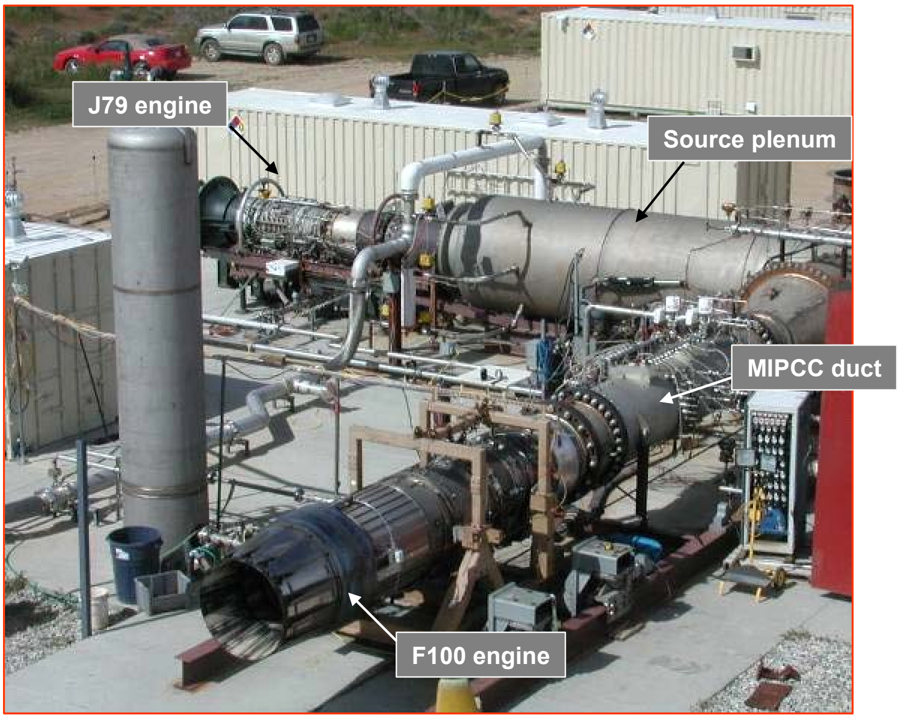


Figure 31. MTB facility.

Following facility construction and checkout, testing proceeded in incremental steps. First, a sequence was run with a substitute small J85 turbojet engine in place of the eventual full-scale F100. The testing conducted at a simulated 60,000 ft altitude, served to provide valuable experience in starting and running the test engine. Following completion of this test series, the J85 was removed and the full-sized F100 MIPCC duct was substituted. There followed a series of so-called "hot duct" tests, meaning with just the MIPCC duct installed, prior to installation of the F100 engine. Finally, the entire system was tested with the F100 installed.

3.6 Full Scale MIPCC Engine Demonstration

In the 2004-05 timeframe, a series of F100-200 MIPCC tests was conducted at MTB. MIPCC engine operation in the all the important points of the baseline flight trajectory was demonstrated.

The test series was interrupted by a mishap involving the F100 engine. As of this writing (March 2005), shakedown hot-duct testing of the upgraded MTB has been accomplished and a new F100 has been installed and prepared for another series of full-scale demonstration tests.

Engine operation at two test conditions corresponding to Mach 1.6 is discussed below. Thrust and flow rate data from test 040910 are plotted in Figure 32. These test results clearly show the significant impact that MIPCC has on engine performance. The NEPP model accurately matches both MIPCC and non-MIPCC operation.

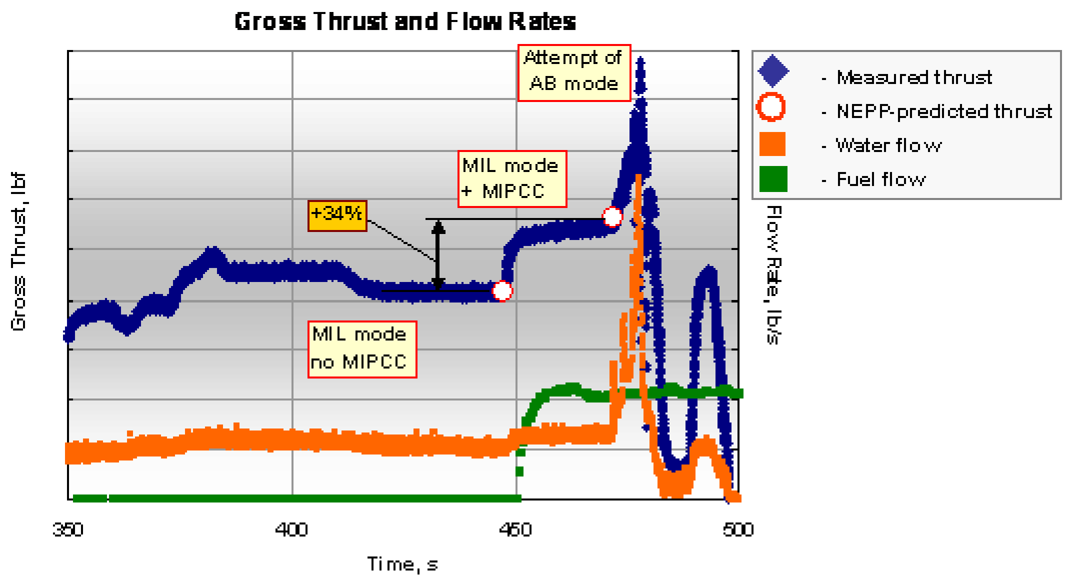


Figure 32. F100-200 MIPCC engine performance in test 040910.

Figure 32 shows gross thrust and two flow rates – water and fuel- versus testing time. MIPCC operation corresponds to the time when water flow is non-zero. The fuel flow spike corresponds to initiation of the F100 engine afterburner (AB). This attempt to light AB was not successful. Experience gained from this and subsequent tests shows that AB should be initiated before water injection.

Stable thrust between approximately 420s and 450s corresponds to non-MIPCC engine operation in military mode. Thrust at 447s has been exactly matched with the NEPP code (white circle in Figure 32 at approximately 450 s). Lower than nominal efficiencies have been applied in the model due to the age and usage of the F100 engine at MTB. The NEPP model adjusted to the military non-MIPCC operation point was applied to the MIPCC mode. Estimated thrust matches the thrust measured in the test very closely (white circle in Figure 32 at approximately 470s). Water flow rate at this point was 2.2% of the air flow rate. Injection of this relatively small amount of water led to a 34% increase of gross thrust of the engine operating without the AB.

Results from test 040914 are shown in Figure 33 as thrust and water and fuel flow rate profiles versus test time. Three different engine modes can be found in this plot, namely: 1) AB mode without MIPCC, 2) AB mode with MIPCC, and 3) MIL mode without MIPCC. Several unsuccessful attempts of AB relight can also be seen in Figure 33.

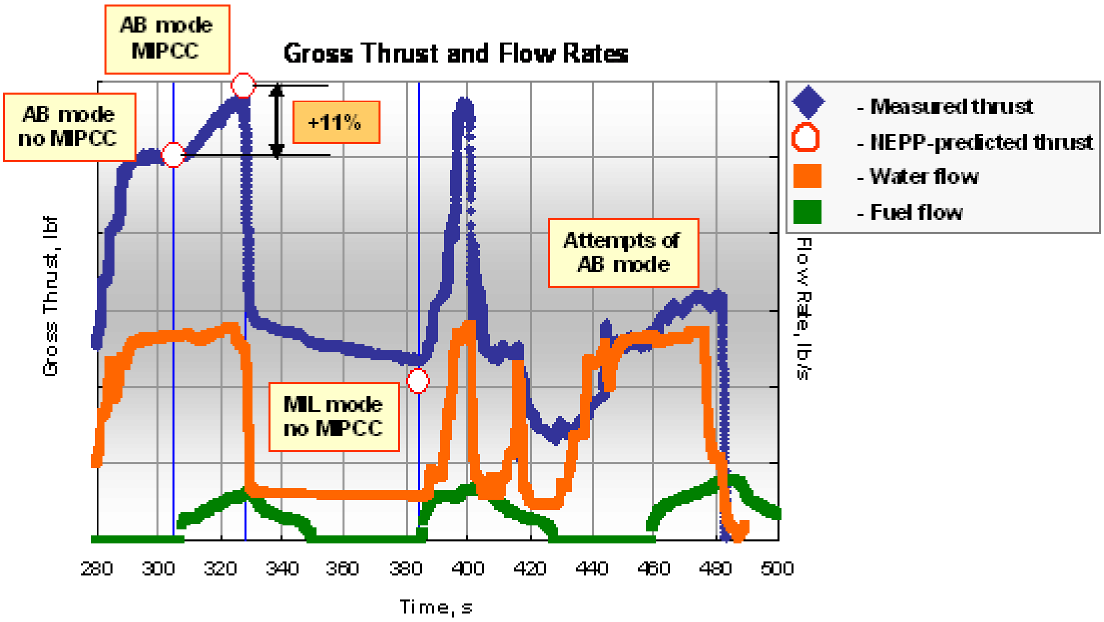


Figure 33. F100-200 MIPCC engine performance in test 040914.

An 11% thrust increase in AB/MIPCC mode compared to AB/no-MIPCC mode was observed. The NEPP model adjusted to exactly match engine thrust on MIL/no-MIPCC mode in the 040910 test, has been applied to MIL/MIPCC mode of the 040910 test and three points of the 040914 test (three white circles in Figure 33). The model closely matches engine thrust. Figure 34 is a comparison of the predicted and observed gross thrust. A maximum disagreement of 5.2% is observed for the “simplest” MIL/no MIPCC mode in the 040914 test. In this case, thrust is underpredicted, meaning that lower thrust is predicted than demonstrated in the test.

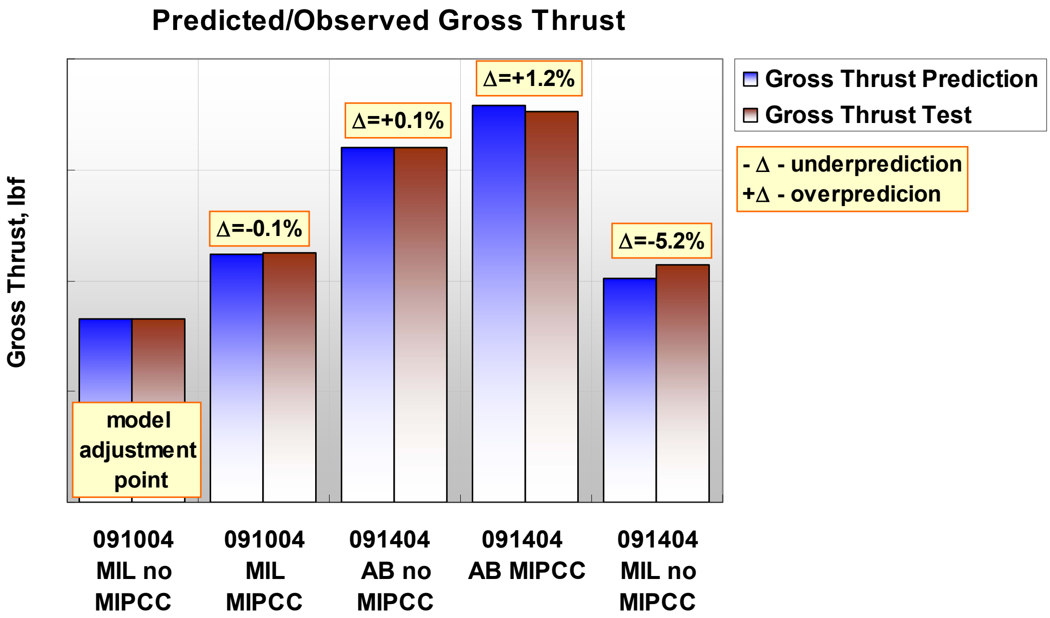


Figure 34. Comparison of the predicted and observed MIPCC engine thrust.

3.7 Concluding Remarks on MIPCC

The feasibility of a launch system using MIPCC as the primary propulsion system of the reusable first stage has been proven in numerous analytical, design, and experimental efforts. It is commonly recognized, that MIPCC is a key enabling technology for such launch system types. Flying at high Mach number and altitude, the engine works within conventional operating envelopes due to the judicious use of coolant injection. This permits limited technology demonstration prior to a flight test program. For this demonstration, MTB has been developed and proven to provide high Mach number/high altitude conditions at the engine entrance. The highest Mach number of 3.5 at an altitude 81,000 ft has been simulated to date with full- scale air flow rate.

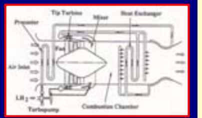
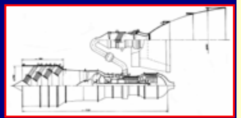
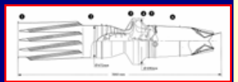
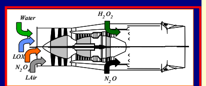
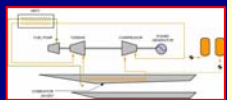
An F100 engine with MIPCC installed generates thrust as predicted at Mach 1.63 and an altitude of 35,000 ft trajectory point. 34% gross thrust improvement at MIL mode has been observed at MTB for Mach 1.6 engine inlet conditions (water/air ratio 2.2%).

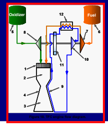
The NEPP model adequately describes MIPCC engine performance at the demonstrated MTB conditions. It has been verified in five different operational modes.

4.0 CONCLUSION

Eight cycles were considered in this lecture. The common feature is judicious use of thermodynamic properties of the working fluids. In most cases (ATREX, ATRDC, KLIN, AspiRE, MIPCC) this permits performance enhancement and/or flight envelope expansion compared to the basic cycles. In other cases (SFC, TFC) it permits enhancement of the internal cycle thermal management and either operational range extension (SFC) or enhancement of the engine reliability and weight characteristics (TFC). A summary of the cycles considered is presented in Table 3.

Table 3. Summary of the synergistic cycles.

Cycle		Application	Pros/Advantages	Cons/Challenges	Synergistic Features
ATREX – expander air turbo ramjet		• First stage of the TSTO • Mach 0-6	• High I_{SP} • Low development and production cost • High TRL	• Low T/W ratio • Low specific thrust • Only LH_2 fuel	LH_2 works as a : • Coolant • Turbine gas • Fuel
ATRDC – deeply cooled air turborocket		• SSTO • First stage of the TSTO • Mach 0-6	• High T/W ratio • High specific thrust • Can be integrated with ramjet	• Moderate I_{SP} • Complexity • Only LH_2 fuel • Low TRL	LH_2 works as a : • Coolant • Turbine gas • ATRDC fuel • (Ramjet fuel)
KLIN cycle – thermally integrated TJ and LRE		• SSTO • First stage of the TSTO • Mach 0-6	• Balance between I_{SP} and T/W • Promising launcher performance	• Complexity • Only LH_2 fuel • Low TRL	LH_2 works as a : • Coolant • Rocket turbine gas • Rocket fuel • Turbine fuel
AspiRE cycle – rocket engine utilizing liquefied air on the "low speed" mode		• SSTO • First stage of the TSTO • Mach 0-6	• Balance between I_{SP} and T/W • Promising launcher performance • Simplicity compared to KLIN cycle	• Only LH_2 fuel • Low TRL	LH_2 works as a : • Coolant • Rocket turbine gas • Fuel
MIPCC engine – conventional turbine with coolant injection prior to compression		• SSTO • First stage of TSTO • Other accelerators • Mach 0-4.5	• High thrust • Extended flight envelope • High TRL, applicable to existing turbines	• Reduced I_{SP} • Increased length for water evaporation	Water injection provides: • Lower air inlet temperature • Higher air compression • Higher air flow
Turbine engine with rocket augmentation		• SSTO • First stage of TSTO • Other accelerators • Mach 0-4.5 (with MIPCC)	• High thrust • May reduce airflow reqs. • Flexible operation • Applicable to existing turbines	• Low I_{SP} • Requires AB redesign	Oxidizer injection provides: • Higher engine mass flow • Higher AB temperature
SFC scramjet – scramjet with closed loop second fluid cooling		• Launch vehicle • High speed cruise vehicle • Mach 4-8+	• Extended flight Mach numbers for HC fuels • Higher I_{SP} at high Mach • Higher thermal margin	• Complexity • Low TRL	Second fluid is used to : • Cool engine • Drive turbopump • Generate power (optional)

Cycle	Application	Pros/Advantages	Cons/Challenges	Synergistic Features
TFC rocket engine – cycle with closed loop third fluid cooling 	• Booster rocket engine • Upper stage rocket engine	• High I_{SP} compared to GG cycle • 2-3 times lower max cycle pressure compared to SC cycle • Flexible fuel	• Additional unit – heat exchanger/condenser • Start up/shut down issues	Third fluid is doing job usually done by fuel or hot gas from preburner. This allows lower max cycle pressure and turbine loads

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