NASA Technical Memorandum 100237

Auxiliary Propulsion Technology for Advanced Earth-to-Orbit Vehicles

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(NASA-TM-100237) AUXILIARY PROPULSION TECHNOLOGY FOR ADVANCED EARTH-TO-ORBIT VEHICLES (NASA) 16 p CSCL 21H

N88-14127

Unclas
G3/20 0116608

Prepared for the
1987 JANNAF Propulsion Conference
San Diego, California, December 15-17, 1987

NASA

AUXILIARY PROPULSION TECHNOLOGY FOR ADVANCED EARTH-TO-ORBIT VEHICLES

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E-3783

ABSTRACT

The payload which can be delivered to orbit by advanced Earth-to-Orbit vehicles is significantly increased by advanced subsystem technology. Any weight which can be saved by advanced subsystems design can be converted to payload at Main Engine Cut Off (MECO) given the same launch vehicle performance. This paper is concerned with the auxiliary propulsion subsystem and the impetus for the current hydrogen/oxygen technology program. A review of the auxiliary propulsion requirements of advanced Earth-to-Orbit (E-T-O) vehicles and their proposed missions is given first. Then the performance benefits of hydrogen/oxygen auxiliary propulsion are illustrated using current shuttle data.

The proposed auxiliary propulsion subsystem implementation includes liquid hydrogen/liquid oxygen (LH_2/LO_2), primary Reaction Control System (RCS) engines, and gaseous hydrogen/gaseous oxygen (GH_2/GO_2) vernier RCS engines. A distribution system for the liquid cryogens to the engines is outlined. The possibility of providing one "dual-phase" engine that can operate on either liquid or gaseous propellants is being explored, as well as the simultaneous firing of redundant primary RCS thrusters to provide Orbital Maneuvering System (OMS) level impulse. Scavenging of propellants from integral main engine tankage is proposed to utilize main engine tank residuals and to combine launch vehicle and subsystem reserves.

INTRODUCTION

Advanced auxiliary propulsion subsystems technology can significantly increase the payload of advanced Earth-to-Orbit (E-T-O) vehicles. This can be done by the development of higher performance, lower weight subsystems and/or by the development of subsystems which minimize the propellant reserves and residuals of the vehicle. This paper describes a proposed liquid hydrogen/liquid oxygen (LH_2/LO_2) primary Reaction Control System (RCS) and the gaseous hydrogen/gaseous oxygen (GH_2/GO_2) vernier RCS. This auxiliary propulsion subsystem has a high performance and propellant management which scavenges both liquid and gaseous propellants from the launch vehicle propulsion tankage after Main Engine Cut Off (MECO). This technology was proposed for the current shuttle two decades ago, but the embodiment proposed here seems to have been ruled out without detailed analyses.

The early technology development program placed requirements on the auxiliary propulsion subsystems that are still valid today. These requirements include: long life, high reliability, high performance, minimal system maintenance, minimum complexity, minimum weight, and the flexibility to operate over a wide range of environmental and operational conditions. Hydrogen and oxygen propellants were investigated to meet these requirements, taking advantage of using the same propellants as the launch propulsion system to simplify vehicle-propellant logistics. The nontoxic, noncorrosive properties of these propellants and their clean exhaust products were also felt to be advantages.

The sequence of events which took place in the early technology development program are outlined in Ref. 1. They include, first, abandoning LH_2/LO_2 because of uncertainties with pulse mode ignition and the distribution of liquid cryogens. Then GH_2/GO_2 was examined in detail.[2] Both high chamber pressure, (high-P_c) and low chamber pressure (low-P_c) engines were considered. Cryogenic liquid storage was needed to store the required propellants which were vaporized by an arrangement of pumps, heat exchangers, and gas generators. Development of the high-P_c engine was hampered when it was determined that 10 to 20 percent of the mass of tanked propellant was required for propellant conditioning. Thus, even though the thruster specific impulse was 430 sec, the system delivered specific impulse was estimated to be only 375 sec. The high-P_c engine was then abandoned when analytical studies indicated that a heavy high pressure accumulator was needed. This accumulator was to store enough preconditioned propellants to sustain adequate steady-state engine thrust for the duration of time required for the gas generators, turbopumps, and heat exchangers to be sequenced and brought to operational status. For example, a 140 kg (300 lbm) accumulator was required for a response time of 1.0 sec at a blowdown pressure ratio of 0.5.[2]

To circumvent the above mentioned difficulties, the low-P_c system was to receive its gaseous propellants by isentropic blowdown of the launch propulsion tankage supplemented by propellant drawn from cryogenic storage through a passive heat exchanger wrapped around the launch propulsion tanks. Because of the low chamber pressure of 104 kPa (15 psia), the nozzle expansion ratio was limited to 5:1 and a system delivered specific impulse of 375 sec was expected. The thrusters, however, were quite large with an estimated size of 59.2 cm length and 35.4 cm diameter.[2]

A comparison of these systems with earth storable bipropellant systems is given in Ref. 3. This paper concluded that a system weight savings could not be achieved through the use of GH_2/GO_2 unless the RCS total impulse exceeded 22 million N-sec (5 million lbf-sec). The relative state-of-the-art of the Earth storable bipropellant system and GH_2/GO_2 systems then favored the Earth storable bipropellant system. At this point, a vehicle studies effort concluded that a smaller shuttle orbiter with external, expendable main engine tankage would provide a more cost effective launch system. This resulted in a reduction in shuttle size along with a significant reduction in RCS impulse requirements from over 8.9 million N-sec to approximately 4.4 to 6.7 million N-sec. The abandonment of GH_2/GO_2 RCS did not occur, however, until analysis showed that sufficient volume was not available for a hydrogen-oxygen orbital maneuvering system (OMS). The development of GH_2/GO_2 technology was not considered justifiable without application to both the OMS and RCS. A program decision was then made to allow scheduled subsystem refurbishment paving the way for the present Earth storable bipropellant OMS and RCS.

This paper presents the results of a vehicle/mission analysis which projects the OMS and RCS requirements of typical advanced E-T-O vehicles. Then, the performance benefits of hydrogen-oxygen RCS are illustrated using current shuttle data. The potential benefits due to integration with the launch propulsion system are then illustrated. This integration, when carried to its fullest extent, also incorporates the propellant supplies of the auxiliary power units and the fuel cells. The objective of present technology program is to eliminate the uncertainties in the operation of both LH_2/LO_2 and GH_2/GO_2 engines over the range of propellant inlet conditions compatible with scavenging of both gaseous and liquid propellants from the launch propulsion tankage. The possibility is being explored of building one "dual-state" engine that can operate on both liquid and gaseous propellants. Finally, a propellant management system is outlined which scavenges the reserves and residuals from the launch propulsion system for use in an OMS/RCS combined system in which redundant RCS engines fired simultaneously provide OMS level impulses. This realization could eliminate one entire vehicle subsystem. The technologies proposed have a very time-critical development schedule to increase their probability of being incorporated into advanced vehicle designs.

VEHICLE/MISSION ANALYSIS

Many different vehicle designs have been proposed for advanced E-T-O transportation. Some examples of these vehicles are shown in Fig. 1. The vehicles differ primarily in launch mode; i.e., horizontal versus vertical; number of stages, ie., one versus two; and if it is a staged vehicle, the staging Mach Number, i.e., subsonic, supersonic or hypersonic. Different degrees of reusability of vehicles include the use of expendable launch vehicles and drop tanks. Results of a launch propulsion evaluation of many of these vehicles is given in Ref. 4. These vehicles will almost certainly have a LO_2/LH_2 orbiter stage because of the high performance of these propellants and the demands of the missions involved.

The types of missions proposed for E-T-O vehicles differ primarily in the payload mass and length of stay in orbit. Payloads range from 2270 kg for a research type vehicle to 29 500 kg for an operational type vehicle. The vertically launched Shuttle II type vehicle has payloads ranging from 9000 kg for a single stage vehicle, 14 500 kg for a small two-stage vehicle to 18 100 kg for the advanced vehicle. Unmanned heavy lift vehicles are being proposed for payloads of 38 600 to 68 200 kg. The length of stay in orbit proposed for these vehicles varies from one low Earth orbit lasting 90 min to a Space Station rendezvous mission lasting 5 days.

Estimates of the auxiliary propulsion requirements for the NASP and Shuttle II type vehicles are given in Table I. These requirements vary with the type of vehicle and, for a given vehicle, vary from mission to mission. The NASP research type vehicle with a one orbit mission has the lowest delta-V requirements for both the OMS with 91 m/sec and the RCS with 23 m/sec. These requirements represent the minimum to insert a vehicle into low Earth orbit, maintain that orbit for one revolution, and then deorbit. The mass of the vehicle and the required maneuvering accelerations set the thrust level of the OMS between 1110 and 11 100 N. Likewise, the estimated thrust level of the RCS engines is less than 111 N.

The NASP operational vehicle will be a multimission vehicle. The lowest delta-V requirements for the OMS and RCS systems for this vehicle occur on a one orbit reconnaissance mission with the OMS at 151 m/sec and the RCS at 25 m/sec. These delta-V's are greater than for the research vehicle because of the on-orbit maneuvering required to accomplish the mission. The multi-orbit satellite maintenance mission has even greater requirements with the OMS at 506 m/sec and the RCS at 216 m/sec. This results primarily from the length of stay in orbit and the rendezvous with the satellite. Notice also that the length of stay in orbit for a multi-orbit mission of a NASP vehicle requires the OMS system to deliver a significant delta-V for plane change. The thrust level projected for the OMS and RCS rockets reflect the expected increase in mass of the operational vehicle as compared with the research vehicle. These thrust levels are 2230 to 22 300 N for the OMS and less than 223 for the RCS.

The Shuttle II vehicle will also be a multimission vehicle, but its primary mission is servicing the Space Station. This vehicle has higher delta-V's for orbit insertion and deorbit because of the 485 km orbital altitude of the Space Station setting the OMS delta-V at 412 m/sec. Once docked, the orbit maintenance requirements of the shuttle will be delivered by Space Station systems and the RCS requirements are accordingly reduced to 46 m/sec. Thrust levels are expected to be in the same range as that of the NASP.

INTEGRATED PROPULSION SYSTEMS

An integrated propulsion system is defined in this paper to be one that has the necessary plumbing and transfer apparatus to manage vehicle propellant reserves and residuals in a manner such that the total is minimized. Such a system allows the greatest mission flexibility through the transfer of propellants between subsystems and even allows a mission tradeoff between propellant and payload. Launch subsystem propellant is not jettisoned in such a system but is scavenged for use in the OMS, RCS, or power conversion subsystems. The integration of propellant supplies represents a certain risk from an interface point of view, but it can also offer the required redundancy for fail-return operation of the vehicle.

PRESENT SHUTTLE

The OMS and RCS on the present shuttle are pressure-fed propulsion systems utilizing storable hypergolic propellants (monomethlylhydrazine and nitrogen tetroxide). The RCS is comprised of three subsystems: one in the forward module and one in each of two aft propulsion modules as shown in Fig. 2. Also contained in each of the two aft propulsion modules is an OMS subsystem. The OMS and RCS subsystems are presently integrated to the extent that interconnection of propellant supplies within each of the propulsion modules and cross-connection between aft modules is provided. This allows the OMS and aft RCS to operate from either propellant supply and to operate on propellants from the opposite side of the vehicle for vehicle trimming.

Each aft module contains one 26 700 N thrust OMS engine with a specific impulse of 313 sec. Also, each aft module contains 12 primary RCS engines, each with a thrust level of 3870 N and a specific impulse of 280 sec. In addition, each aft module contains 2 vernier RCS engines, each with a thrust level of 107 N and a specific impulse of 260 sec. The forward module contains 14 primary RCS engines and 2 vernier RCS engines. The total auxiliary propulsion engine count on the present Shuttle is 2 OMS engines, 38 primary RCS engines and 6 vernier RCS engines.

FUTURE E-T-O VEHICLES

An integrated propulsion system on advanced E-T-O vehicles would use hydrogen and oxygen as propellants for the auxiliary propulsion subsystem (OMS and RCS) to take advantage of scavenging propellants from the launch propulsion subsystem. These propellants have a higher specific impulse than storable hypergolic propellants. The integrated hydrogen/oxygen (H/O) auxiliary propulsion subsystem could have a higher subsystem dry mass fraction but, scavenging of propellants could lead to a lower on-orbit mass fraction required by the total auxiliary propulsion subsystem. A simple analysis shows these tradeoffs. The rocket performance equation can be written

$$\frac{W_p}{W_{orbit}} = 1 - \exp\left(\frac{\Delta V}{I_s g_c}\right) \tag{1}$$

where

W_p total mass of auxiliary propulsion propellant required for the mission

W_{orbit} total mass of vehicle in orbit

ΔV total vehicle change in velocity to be supplied by the auxiliary propulsion subsystem

I_s average specific impulse of auxiliary propulsion system

g_c gravitational acceleration at Earth's surface

The dry mass fraction of the auxiliary propulsion system is defined

$$f_d = \frac{W_d}{W_d + W_p} \tag{2}$$

where

W_d propulsion system dry mass

The total mass of auxiliary propellant (W_p) is the sum of the mass of auxiliary propellant which is loaded (W_{pl}) into the auxiliary propulsion tankage on the Earth and the mass of auxiliary propellant which is scavenged (W_{ps}) from the launch propulsion system after MECO.

$$W_p = W_{p\ell} + W_{ps} \tag{3}$$

The fraction of the total auxiliary propellant which is scavenged is defined

$$f_s = \frac{W_{ps}}{W_p} \tag{4}$$

Combining these equations, the on-orbit mass fraction required by the auxiliary propulsion system can be written

$$\frac{W_{p\ell} + W_d}{W_{orbit}} = \left(\frac{1 - f_s + f_s f_d}{1 - f_d}\right)\left(1 - \exp\left(\frac{\Delta V}{I_s g_c}\right)\right) \tag{5}$$

This quantity is plotted as a function of auxiliary propulsion system ΔV in Fig. 3. A comparison is made of the unscavenged (f_s = o) auxiliary propulsion system on-orbit vehicle mass fraction typical of the present shuttle (average specific impulse, I_s = 310 sec) with that of an unscavenged H/O auxiliary propulsion system (average specific impulse of 427 sec). This high specific impulse assumes that the H/O propellants can be used in the as delivered state by the auxiliary propulsion system, i.e., no propellant conditioning is required. Typical shuttle mission data shown by the symbols are obtained from Ref. 5. The missions include: 29 500 kg payload launched due east, 11 300 kg payload launched on a sortie mission, 14 500 kg payload launched on a deploy mission, and 1130 kg launched on a retrieve mission. These data show an auxiliary propulsion system dry mass fraction of f_d = 0.3 for the storable hypergolic propellants. They also show that the increased specific impulse of hydrogen/oxygen propellants will allow this dry mass fraction to increase to f_d = 0.5 and still maintain the same on orbit vehicle mass fraction required by the auxiliary propulsion system. For example, with a ΔV = 457 m/sec (1500 ft/sec) and with the on-orbit vehicle mass W_{orbit} = 109 000 kg typical of the current Shuttle, this allows the mass of a H/O system to be 5620 kg heavier than the equivalent storable hypergolic propellant system. This mass represents a margin which can be used to store the less dense H/O propellants and can be used for propellant subsystem integration.

When the subsystems are integrated, propellant is scavenged from the launch propulsion subsystem after MECO and the propellant mass which is loaded into the auxiliary propulsion subsystem on Earth ($W_{p\ell}$) decreases. The on-orbit vehicle mass fraction required by the auxiliary propulsion system (ordinate) then decreases as shown in Fig. 3. The minimum is achieved when 100 percent of the required propellant is scavenged. This is shown by the lower dotted line in Fig. 3 for (f_s = 1, f_d = 0.5). For the example given previously, (ΔV = 457 m/sec, W_{orbit} = 109 000 kg), this amounts to 11 300 kg of scavenged propellant for which payload could be substituted. This is the maximum possible mass savings due to scavenging for this example. The reserves and residuals in the shuttle external tank are only 7200 kg according to Ref. 5 for comparison. Therefore, the propellant estimated to be available is not enough to achieve 100 percent scavenging. The above numbers suggest that 64 percent is possible for Shuttle II.

INTEGRATED PROPULSION ROCKET ENGINES

When H/O auxiliary propulsion was baselined for the current Shuttle Orbiter, an extensive thruster technology program was set in motion. An excellent review of this technology program is given in Ref. 6. This work was continued even after the baselining of Earth storable bipropellants on the current shuttle in order to provide an alternate or backup approach and for other applications such as the space tug. Three types of H/O auxiliary propulsion systems were studied. They were differentiated by the state of the propellants delivered to the thrusters, i.e., LH_2/LO_2, GH_2/LO_2 and GH_2/GO_2.

LH_2/LO_2 ROCKET

The auxiliary propulsion system using LH_2/LO_2 thrusters was considered to be the simplest, lightest weight, highest performance system. However, major technical concerns existed, such as ignition, thrust chamber cooling, pulse mode operation, combustion stability, delivered performance, and thruster heat soak back to the propellant feed lines. These concerns were addressed and resulted in the successful hot fire demonstration of a 5560 N LH_2/LO_2 thruster in pulse mode operation. The results are reported in Ref. 7. A spark/oxygen torch igniter was employed, supported by ignition experiments with liquid, two phase and gaseous propellants. Rapid ignition in less than 20 msec was achieved at the very low spark energy levels of 10 mJ. Thermal management in the system was achieved via a very low velocity propellant recirculation loop integrated into the valves, injector manifolding and igniter to maintain these components at the propellant supply temperature. The LH_2/LO_2 injector incorporated a 24 element like-on-like doublet pattern with dual wall low volume manifolds and close coupled thermally isolated valves. The injector face was actively cooled during firing by diverting a portion of the LH_2 through dump circuits located in the injector face. The low volume manifolds were required for good pulsing performance and low response time. The combustion chamber was a barrier cooled columbium chamber cooled by a low temperature fuel rich gas barrier at the chamber wall. The thruster was run at an overall mixture ratio of 4.5, a chamber pressure of 3450 kPa (500 psia) and a nozzle expansion ratio of 40. The delivered vacuum specific impulse was 426 to 427 sec and was achieved with the response of 75 msec from electrical signal to 90 percent thrust. Of the 75 msec, 50 msec was associated with electrical energization of the valve solenoids. Pulse mode performance was 90 percent of the steady state values for impulse bits down to 889 N-sec and at 222 N-sec, performance was measured to be 75 percent of steady state. The design was found to have stable combustion with external excitation at a high frequency of 18 000 Hz and was found to recover from 100 percent overpressure in 1.0 msec when bombed with a 2 grain RDX (Cyclotrimetylene-trinitramine) charge. Only ten tests were conducted with the columbium chamber when the protective coating was eroded and burn through occurred. The erosion was the result of high temperature injector induced heat streaks.

GH_2/LO_2 ROCKET

The auxiliary propulsion system using GH_2/LO_2 thrusters was more complex since a gas generator, heat exchanger and accumulator were needed to condition the hydrogen which was stored as a liquid. The thruster technology concerns, however, were the same as for LH_2/LO_2 thrusters, but the availability of GH_2 allowed a regeneratively cooled combustor to be employed to address them. This technology development resulted in the hot fire demonstration of a 5560 N GH_2/LO_2 thruster in pulse mode operation. These results are reported in Ref. 7. A spark/oxygen torch igniter was employed which was identical to that used on the LH_2/LO_2 thruster. Thermal management in the LO_2 system was identical to that used on the LH_2/LO_2 thruster. However, the lower density of GH_2 allowed regenerative cooling of a portion of the combustion chamber. The GH_2/LO_2 injector incorporated a 36 element coaxial tube pattern with dual wall, low volume manifolding and close coupled, thermally isolated valves on the LO_2 circuit only. The fuel manifold was fed from the discharge of the regeneratively cooled chamber which employed a thermally isolated valve. The combustion chamber consisted of the regeneratively cooled section discussed above and a dump cooled section which formed the convergent portion of the cooled chamber. The throat-nozzle skirt section was dump cooled and fabricated from Haynes alloy. The thruster was run at an overall mixture ratio of 4.5, a chamber pressure of 3450 kPa (500 psia) and a nozzle expansion ratio of 40. The delivered vacuum specific impulse was 435 sec using 18 percent of the fuel to dump cool the throat-nozzle skirt. A response time of 75 msec from electrical signal to 90 percent thrust was achieved with 50 msec of this time associated with electrical energization of the valve solenoids. Pulse mode performance was 93.4 percent of the steady state value at an impulse bit of 245 N-sec. The design was found to have stable combustion when externally excited at a high frequency of 18 000 Hz. A series of 44 tests were conducted on this GH_2/LO_2 thruster for a total test time of 282 sec and no heat transfer problems were noted.

GH_2/GO_2 HIGH-P_c ROCKET

The auxiliary propulsion system using GH_2/GO_2 thrusters was the most complex since gas generators, heat exchangers and accumulators were required to condition both propellants which were stored as liquids. The GH_2/GO_2 thrusters received the greatest amount of development effort because of perceived difficulty in delivering cryogenic liquids over long distances to the liquid engines and because GH_2/GO_2 thrusters offered the best pulse mode performance. The thruster technology concerns were the same as for both the LH_2/LO_2 and GH_2/LO_2 thrusters, but the availability of gaseous propellants allowed a wide variety of component technologies to be employed to address them. The technology development program, reviewed in detail in Ref. 6, resulted in at least four different hot fire demonstrations of 6680 N (1500 lbf) thrusters in pulse mode operation. A number of different ignition candidates were evaluated including the spark/oxygen torch similar to that used on the LH_2/LO_2 and GH_2/LO_2 thrusters. This candidate was evaluated as the most successful at all test conditions and provided rapid ignition in 10 msec with the very low spark energy level of 5 mJ. Other ignition systems were also successfully employed. They included the combustion chamber wall-mounted spark plug, the catalytic torch, and the compression-resonance tube torch. Five different GH_2/GO_2 injectors were evaluated and found to provide acceptable combustion efficiency and stable combustion. They included: coaxial tubes, premix, triplet, trislot, and reverse flow/vortex cup injectors. A variety of combustion chamber designs were tested. These designs used different combinations of three basic cooling methods using the gaseous fuel as coolant: regenerative, dump, and film cooling. All of the designs were found capable of meeting the performance requirements. Film cooled designs provided a lighter weight thruster but also delivered a somewhat lower specific impulse than the regenerative designs. The thrusters were run at overall mixture ratios of 4.0, chamber pressures of 2070 kPa (300 psia) and a nozzle expansion ratio of 40. The delivered vacuum specific impulse was 432 to 447 sec and was achieved with a response time of 30 to 50 msec from electrical signal to 90 percent thrust. This time includes 10 to 15 msec for valve opening. Pulse mode performances ranged from 79 to 92 percent of the steady state values at impulse bits ranging from 147 to 289 N-sec. The total time put on any thruster ranged from 250 to 325 sec. The total number of pulses on any thruster ranged from 2672 to 3625 pulses. No critical GH_2/GO_2 thruster technology problems remain. Further work should be directed toward optimization of thruster performance and variable mixture ratio operation.

GH_2/GO_2 LOW-P_c ROCKET

Another basic design approach for GH_2/GO_2 auxiliary propulsion system eliminated the pumps and compressors from the system and utilized propellants directly from the launch propulsion tankage. This tankage had a maximum operating pressure of 276 to 310 kPa (40 to 45 psia). Therefore, a low chamber pressure GH_2/GO_2 thruster development program was conducted (refs. 2, 8, 9). The nominal design point for this thruster was 6680 N at a chamber pressure of 104 kPa (15 psia), mixture ratio of 2.5 and a nozzle expansion ratio of 5. The technology program was conducted with at least two heat sink type thrust chambers. No cooled thruster configurations were tested. The preferred injector concepts consisted of a 200 element coaxial design and a 50 element concentric tube with swirler. The chamber cooling concepts which were proposed included combinations of film, dump, and regenerative cooling. Based on the injector testing and cooling loss predictions, thruster performance was predicted at 375 to 389 sec. Pulse mode operation, although not demonstrated, was shown to be feasible by the less than 50 msec response time from electrical signal to 90 percent thrust in the heat sink chamber tests.

WATER ELECTROLYSIS ROCKET

GH_2/GO_2 propulsion systems have also been proposed for satellite propulsion. The propellants for these thrusters were derived from water electrolysis and a development program for these thrusters was conducted.[10,11]. Two design points were included in this program. The first was 22 N at a chamber pressure of 345 kPa (50 psia), mixture ratio of 8.0 and a nozzle expansion ratio of 40. The second was 0.4 N at a chamber pressure of 552 kPa (80 psia), mixture ratio of 8.0 and a nozzle expansion ratio of 100. A six element premix injector with integral H_2 film cooling was used for the 22 N thruster. One of these elements was used for the 0.4 N thruster. A spark plug mounted on the injector face was used for ignition with a small bleed of H_2 gas between the electrode and injector face. This ignition system had an energy of 35 to 85 mJ per spark. Valves were thermally isolated from the combustion chamber by the use of thin tube and structural member. The combustion chamber and exit nozzle were constructed of $MoSiO_2$ coated molybdenum. The method used for chamber cooling was a combination of a refractory metal to maximize the temperature limits, radiation cooling, film cooling and conduction cooling from the throat. Flightweight designs of these thrusters were constructed and tested to demonstrate life capability and performance. The delivered vacuum specific impulse of the 22 N thruster was 355 sec and was achieved with a response time of 12 msec. Pulse mode performance was 91 percent of steady state at impulse bits of 1.1 N-sec. Total firing time on this thruster was 4.16 hr and 152 015 pulses. The engine was in excellent condition after completion of the life test program. The delivered vacuum specific impulse of the 0.4 N thruster was 331 sec and was achieved with a response time of 12 msec. Pulse mode performance was 91 percent of steady state at impulse bits of 0.02 N-sec. Total firing time on the thruster was 10.07 hr and 301 726 pulses. This engine consisted of a coated columbium chamber. A similar molybdenum chamber failed in 3.52 hr of hot firings.

SPACE STATION ROCKET

The space station currently has GH_2/GO_2 propulsion systems baselined for attitude control and drag make-up. A technology program to develop 111 to 222 N thrusters has been recently conducted and is reported in Ref. 12. A photograph of the resulting hardware is shown in Fig. 4. These thrusters are to operate over a wide mixture ratio range from 3 to 8 and have chamber pressures in the range 518 to 690 kPa (75 to 100 psia). Nozzle expansion area ratios varied from 30 to 100. Three different GH_2/GO_2 injectors were evaluated and found to provide acceptable efficiency and stable combustion. They included: a 12 element coaxial, platelet, and reverse flow/vortex cup. The combustion chamber designs are basically regenerative but also use different combinations of dump and film cooling. All three designs were found capable of meeting the performance requirements. Spark torch and wall-mounted spark plugs are used for ignition systems. The delivered vacuum specific impulses were greater than 400 sec at a mixture ratio of 4 and greater than 346 sec at a mixture ratio of 8. All thrusters had response times less than 80 msec from electrical signal to 90 percent thrust, indicating that pulse mode operation with 9 N-sec impulse bits is possible. The greatest hot firing time on any of these thrusters was 23.9 hr over the mixture ratio range of 3 to 8. The physical state of the thrusters upon completion of the testing was excellent and testing to end of life is a major goal of the space station thruster program.

Table II is provided to summarize the thruster performance data for these technology development programs.

DUAL-STATE ROCKET

The preferred integrated auxiliary propulsion system for advanced E-T-O vehicles appears to be LH_2/LO_2 for the primary RCS engines and GH_2/GO_2 for the vernier RCS engines. This configuration is chosen, first of all, because both propellant states are available in any scheme to scavenge propellants from the launch propulsion tankage. Second, the technology developed to date shows that smaller impulse bits and faster response times are available for GH_2/GO_2 thrusters to perform the vernier RCS function. However, because of the system performance penalty paid to gasify propellants, this system is recommended to be kept small and the majority of the RCS propulsion is to be accomplished with LH_2/LO_2 primary RCS propulsion.

One multi-purpose rocket is currently proposed for development to satisfy both the primary RCS and vernier RCS requirements of advanced E-T-O vehicles. This multi-purpose rocket will be a "dual state" hydrogen/oxygen engine that has the capability to operate on either liquid or gaseous propellants. Specifically, the dual state rocket would operate at a primary RCS thrust level by using high pressure liquid propellants and at a vernier RCS thrust level by using low pressure gaseous propellants. As currently envisioned, the "dual state" rocket engine is to be designed for the LH_2/LO_2 primary RCS function with thrust levels in the range 2230 to 6680 N. The thruster is to be pressure fed at a fixed pressure in the range 3450 to 6900 kPa (500 to 1000 psia) with liquid propellants delivered by a cryogenic distribution system with bellows contraction as discussed in the next section of this paper. The temperature of the delivered propellants will vary from cryogenic storage temperature (93 K for LO_2, 21 K for LH_2) to 294 K depending on the rate of usage and the thermal soak into the distribution system. The thruster is to be designed for high performance in this operation, but because of the uncontrolled variable density of the propellant in the manifold, the mixture ratio and thrust level is not anticipated to be held fixed. A secondary design requirement on this "dual state" rocket is that operation in the pulse mode on low pressure gaseous propellants be possible in order to provide the vernier RCS function with the same rocket. These gases will be derived by moderate compression of cryogenic vent gases from the launch propulsion tankage. These gases result from thermal soak into the tankage and are anticipated to be compressed to pressures in the range of 690 to 1380 kPa (100 to 200 psia) as discussed in the next section. The temperatures of these compressed gases varies from 156 to 183 K for GO_2 and 41 to 57 K for GH_2 when compressed to the pressure range of 690 to 1380 kPa (100 to 200 psia). In addition, thermal soak into this gaseous distribution system can raise these temperatures to 294 K depending on the rate of usage. The details of this "dual state" engine design and its performance are to be determined.

PROPELLANT MANAGEMENT SYSTEM

Propellant management systems proposed for advanced Earth-to-orbit vehicles will determine the component technologies that need to be developed. The propellant management system envisioned here has propellants stored as subcritical liquids in well-insulated tanks to protect them from environmental heat soak and to minimize the storage volume. An integrated propulsion system on advanced E-T-O vehicles allows the transfer of propellants between this tankage and the launch propulsion tankage. A cryogenic distribution system to the thrusters was proposed in 1972[13] for the current shuttle and is again proposed here. The cryogenic liquids are pumped from low pressure storage into a pressurized distribution system as shown in Fig. 5. If the pumps are located near the tanks, heat leaks and pressure drops which cause cavitation can be avoided. A contracted bellows accumulator is attached to this system and as heat soaks into the system, the fluid will expand into the accumulator. This accumulator is sized at 23 times the piping volume for liquid oxygen and 26 times the piping volume for liquid hydrogen. This allows a propellant temperature rise from cryogenic temperature to Earth ambient temperature without loss of propellant. Other system components include a small circulation pump to maintain the fluid at a uniform temperature within the system and to actively cool the valves, injector and igniter on the thruster itself. A relief valve to accommodate overheating or overfilling of the system is also required.

The need for vacuum jacketed lines has not been established, but, if they are required, they will afford dual containment of the propellant for a reliability improvement which can eliminate the need for some isolation valves by monitoring the vacuum in the jacket during checkout and flight. The system will operate as a pressure fed system until the bellows are contracted, making the system response insensitive to pump start up time. If the systems are pressurized to 3450 kPa (500 psia), single phase hydrogen is ensured because the critical pressure is 1290 kPa (187 psia). Two-phase oxygen could exist, however, since its critical pressure is 5044 kPa (731 psia). If this should happen, the oxidizer to fuel ratio is reduced and the thruster is protected from an oxidizing environment.

A schematic for scavenging propellants from the residuals of the launch propulsion subsystem is shown in Fig. 6. These residuals are in the main engine tanks after MECO to assure that the main engine turbopumps do not overspeed due to loss of propellant. For example, Ref. 5 shows that the current shuttle external tank has 2750 kg of LO_2 and 1920 kg of LH_2 in residuals. On advanced E-T-O vehicles, it is proposed that the bulk of these propellants be transferred in the liquid state to well-insulated tanks to protect them from environmental heating. These propellants are scavenged either by a capillary acquisition screen in the tank or during a special propellant settling maneuver.

The propellant which is heated by environmental soak is drawn off the tanks in the gaseous state through a liquid/vapor separator which returns the liquid to the tanks. The gas is compressed to moderate pressures so that the physical size of the vernier RCS engines is small. This moderate pressure is here assumed to be 690 kPa (100 psia) for compatibility with power conversion apparatus such as fuel cells and auxiliary power units (APU). This boiloff from the main engine tankage will almost certainly be acquired at a steady rate determined by the tank insulation until only vapor remains in the main tankage. In order to avoid storing large quantities of gaseous propellants, integration of this system with the fuel cell and APU subsystems is proposed. These subsystems use propellants at a steady rate which can be matched to the boiloff rate by tank insulation. The vernier RCS system is only an intermittent user of these propellants. An alternative supply of gas is provided to this gaseous accumulator from the well insulated cryogenic liquid storage tank through a liquid pump and a phase change heat exchanger. The heat exchanger can use very low temperature heat rejected by other subsystems such as the fuel cells or even environmental heating. The accumulator needs only to be sized for the gaseous reserves needed by the power conversion subsystem and the low thrust vernier RCS in order to assure fail-return operation of the vehicle. The weight and power requirements of all system components are kept low by operating the system in the 690 to 1380 kPa (100 to 200 psia) pressure range.

The primary RCS thrusters are shown being fed by the pump, manifold-bellows accumulator system discussed previously. Only one thruster is shown to symbolize the "dual state" thruster design discussed in the previous section. The pumps are sized for the primary RCS thrust levels such that they can be driven by electric motors. Another possibility, which is also proposed for advanced E-T-O vehicles, is to provide the OMS impulse requirements by simultaneously firing redundant LH_2/LO_2 primary RCS engines along with their associated manifold feed pumps. These engines can be configured in pods in the aft end of the vehicle as in the current shuttle design or, if adequate thermal protection can be developed, they can be mounted in the wings. Reference 14 shows that the current 38 primary RCS thrusters can be reduced to 28 wing tip and nose mounted thrusters with more roll and yaw but less pitch authority. Simultaneously firing these thrusters for $\pm x$ translation is made even more feasible today by the developments in control system technology.

The advantages of this propellant management system lies in its ability to manage the vehicle's residuals and reserves to some minimum total value depending on mission reliability desired. This value will almost certainly be less than the sum of residuals and reserves required to achieve the same reliability with separate systems. Propellant which is scavenged from the launch propulsion tankage is jettisoned in non-integrated systems. With integrated systems payload substitute for this jettisoned mass is possible.

SUMMARY

The hydrogen/oxygen auxiliary propulsion subsystem is targeted in a technology development program to increase its probability of being incorporated into advanced E-T-O vehicle designs because of its inherent high performance, nontoxic and noncorrosive propellant properties, clean exhaust products and simplified vehicle-propellant logistics. Any mass saved in advanced auxiliary propulsion subsystem design can be added directly to the vehicle payload. The auxiliary propulsion requirements of these vehicles vary significantly depending on such vehicle mission parameters as mass of payload and length of stay in orbit. The performance benefit of hydrogen/oxygen auxiliary propulsion is illustrated using current Shuttle data. A projected increase in average specific impulse from 310 to 427 sec is possible. The mass of a hydrogen/oxygen auxiliary propulsion subsystem is allowed to be 5620 kg heavier than the equivalent storable hypergolic propellant system for the same mission $\Delta V = 457$ m/sec (1500 ft/sec) and an orbiter mass of 109 000 kg. In addition, a mass savings of 11 300 kg is possible with 100 percent scavenging for auxiliary propellant. Shuttle data, however, suggest that only 7200 kg is available for scavenging.

The preferred auxiliary propulsion system consists of LH_2/LO_2 for the primary RCS engines and GH_2/GO_2 for the vernier RCS engines. The possibility of designing one "dual-state" engine which can operate on either liquid or gaseous propellants is being explored. Much of the basic technology for these engines was developed for the current Shuttle but was dropped when the program shifted to the smaller Shuttle orbiter with the expendable main engine tankage. GH_2/GO_2 engines are currently under development for the Space Station. LH_2/LO_2 engine ignition, combustion stability and performance in pulse operation were addressed in past technology programs which indicated the feasibility of LH_2/LO_2 engines for this E-T-O vehicle application. A pressurized cryogenic distribution system with a bellows expansion accumulator would deliver cryogens to the engines. Interconnection of the auxiliary propulsion system with the launch propulsion tankage is proposed as a method to reduce reserves and residuals on the vehicle. A scheme for scavenging propellants is proposed. One set of propellants decreases vehicle logistics requirements and the interconnection of systems provides the greatest possible mission flexibility.

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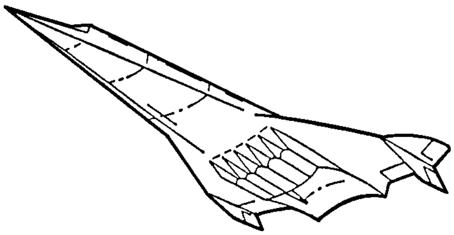
TABLE I. - AUXILIARY PROPULSION SYSTEM REQUIREMENTS

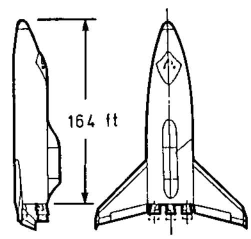
Event	# Burns	ΔV/Burn, m/sec (ft/sec)	OMS ΔV total, m/sec (ft/sec)	RCS ΔV total, m/sec (ft/sec)	Thrust, N (lbf)
NASP Research (one-orbit mission)					
Orbit insert.	1	61 (200)	61 (210)	--------	1110-11 100 (250-2500)
Attit. contr.	100	0.076 (0.25)	--------	7.6 (25)	<111 (<25)
Orbit maintenance	20	0.76 (2.5)	--------	15 (50)	<111 (<25)
Deorbit	1	30 (100)	30 (100)	--------	1110-11 100 (250-2500)
			91 (300)	23 (75)	
NASP Operational (one-orbit recon. mission)					
Orbit Insert.	1	91 (300)	91 (300)	-------	2230-22 300 (500-5000)
Positioning	10	1.5 (5.0)	--------	15 (50)	<223 (<50)
Attit. Contr.	100	0.1 (0.3)	--------	10 (30)	<223 (<50)
Orbit Maint.	20	1.5 (5.0)	30 (100)	-------	2230-22 300 (500-5000)
Deorbit	1	30 (100)	30 (100)	-------	2230-22 300 (500-5000)
			151 (500)	25 (80)	
NASP Operational (multi-orbit sat. maintenance mission)					
Orbin Insert.	3	37 (120)	100 (360)	---------	2230-22 300 (500-5000)
Positioning	20	1. (3.0)	---------	18 (60)	<223 (<50)
Maneuvering	1000	0.06 (0.2)	---------	61 (200)	<223 (<50)
Orbit Maintenance	150	1. (3.0)	---------	137 (450)	<223 (<50)
Orbit Change	2	30 (100)	61 (200)	---------	2230-22 300 (500-5000)
Plane Change	2	137 (450)	274 (900)	---------	2230-22 300 (500-5000)
Deorbit	2	30 (100)	61 (200)	---------	2230-22 300 (500-5000)
			506 (1660)	216 (710)	
Shuttle II (space station rendezvous mission)					
Orbit Insert.	3	96 (315)	290 (950)	---------	2230-22 300 (500-5000)
Positioning	20	0.5 (1.7)	---------	11 (35)	<223 (<50)
Attit. Contr.	1000	0.04 (0.1)	---------	35 (115)	<223 (<50)
Deorbit	1	122 (400)	122 (400)	---------	2230-22 300 (500-5000)
			412 (1350)	46 (150)	

TABLE II. - HYDROGEN/OXYGEN RCS ENGINE PERFORMANCE SUMMARY

	LH_2/LO_2	GH_2/LO_2	GH_2/GO_2 high-Pc	GH_2/GO_2 low-Pc	Water electrolysis		Space station
Injector	Doublet	Coaxial	Various	Coaxial	Premix	Premix	Various
Igniter	Spark/Oxygen	Spark/oxygen	Various	Spark/Oxygen	Spark/H2	Spark/H2	Spark/Oxygen
Cooling	Film	Regenerative	Various	Various	Film	Film	Various
Thrust, N (lbf)	5560 (1250)	5560 (1250)	6680 (1500	6680 (1500)	22 (5)	0.4 (0.1)	111-222 (25-50)
Mixture ratio	4.5	4.5	4.0	2.5	8	8	3-8
Pc kPa, psia	3450 (500)	3450 (500)	2070 (300)	104 (15)	345 (50)	552 (80)	518-690 (75-100)
Exp. ratio	40	40	40	5	40	100	30-100
I (vac.), sec	426-427	435	432-447	375-389	355	331	>400 @ Mr = 4
Response time, msec	75	75	30-50	50	12	12	80
Pulse performance, percent	90	93.4	79-92	------------	91	91	----------------
Min. impulse bit, N-sec (lbf-sec)	889 (200)	245 (55)	147-289 (33-65)	------------	1.1 (0.25)	0.2 (0.005)	9 (2)
Test time, sec	9.3	282	250-235	------------	14 980	36 250	86 040
Condition	Burn through	Excellent	Excellent	------------	Excellent	Excellent	Excellent

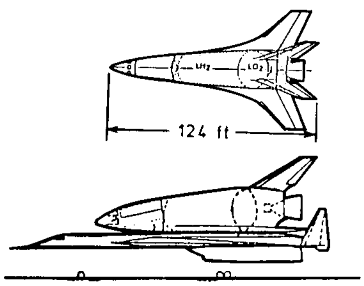


NATIONAL AEROSPACE PLANE
29 500 KG (65 000 LB) PAYLOAD

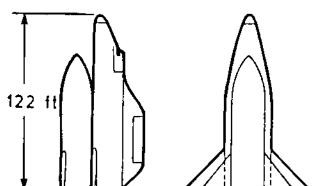


SHUTTLE II CONCEPT
9 000 KG (20 000 LB) PAYLOAD

SINGLE-STAGE TO ORBIT



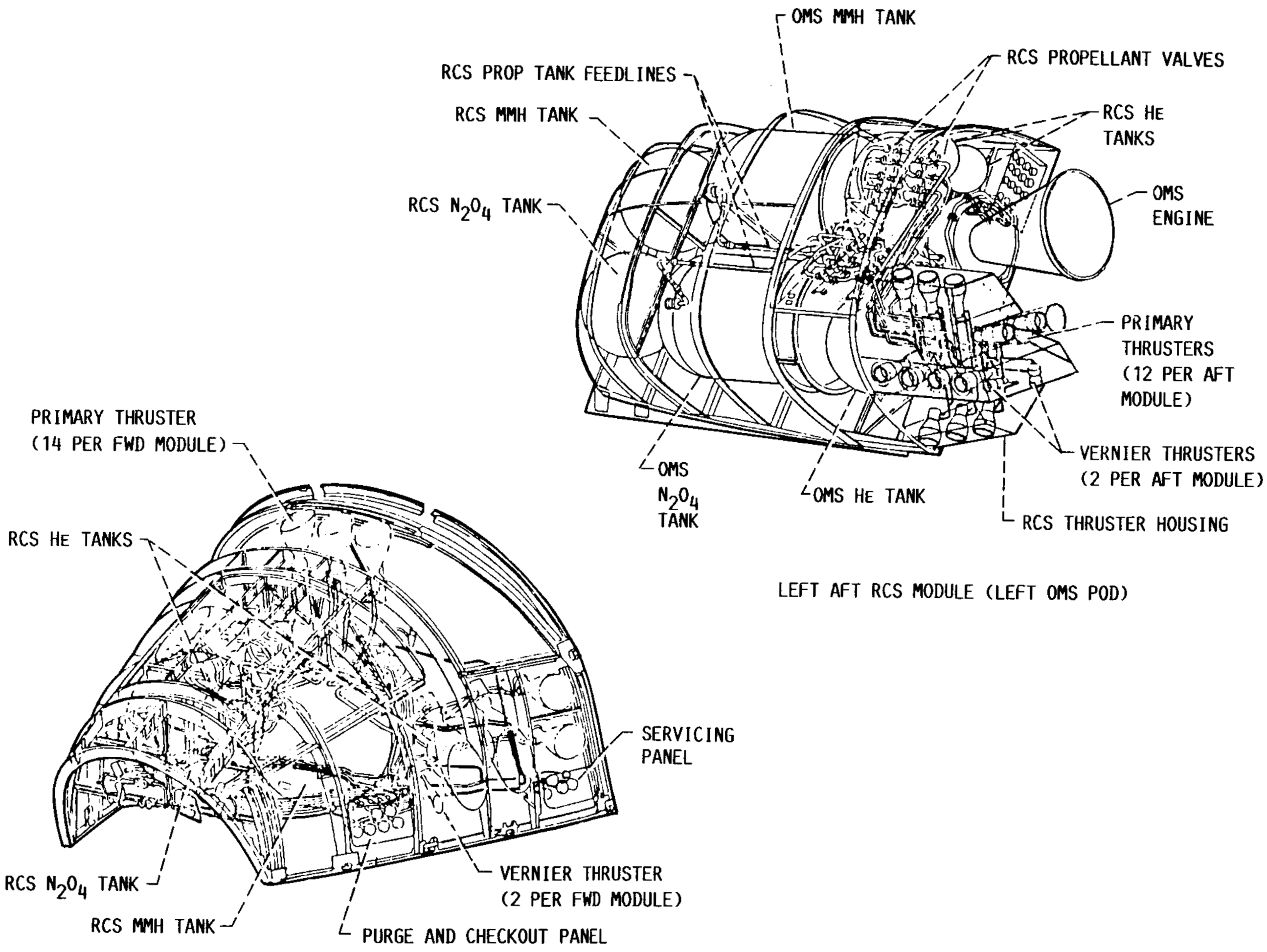
TRANSATOMOSPHERIC VEHICLE
9 000 KG (20 000 LB) PAYLOAD



SHUTTLE II CONCEPT
14 500 KG (32 000 LB) PAYLOAD

TWO-STAGE TO ORBIT

FIGURE 1. - EXAMPLES OF PROPOSED EARTH-TO-ORBIT VEHICLES.



LEFT AFT RCS MODULE (LEFT OMS POD)

FIGURE 2. - CURRENT SHUTTLE OMS AND RCS SUBSYSTEMS FOR REFERENCE AND COMPARISON.

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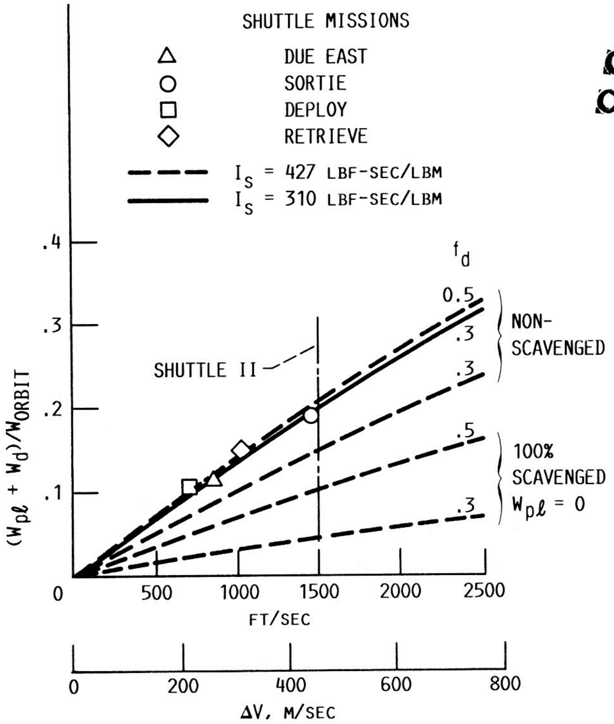

FIGURE 3. - ON-ORBIT VEHICLE MASS FRACTION REQUIRED BY THE AUXILIARY PROPULSION SYSTEM VERSUS AUXILIARY PROPULSION SYSTEM ΔV AS A FUNCTION OF DRY MASS FRACTION OF THE AUXILIARY PROPULSION SYSTEM.

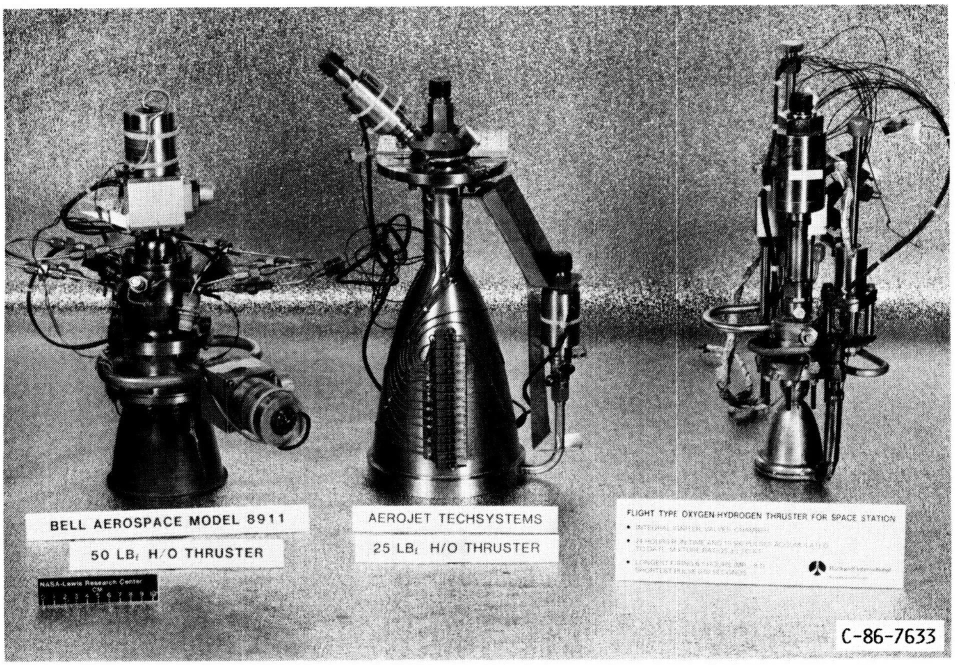

FIGURE 4. - SPACE STATION GH_2/GO_2 TECHNOLOGY PROGRAM THRUSTERS.

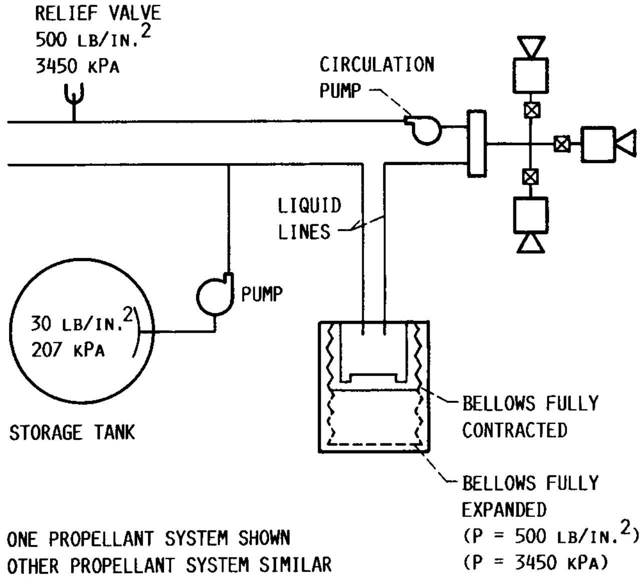

FIGURE 5. - CRYOGENIC LIQUID RCS DISTRIBUTION SYSTEM SCHEMATIC.

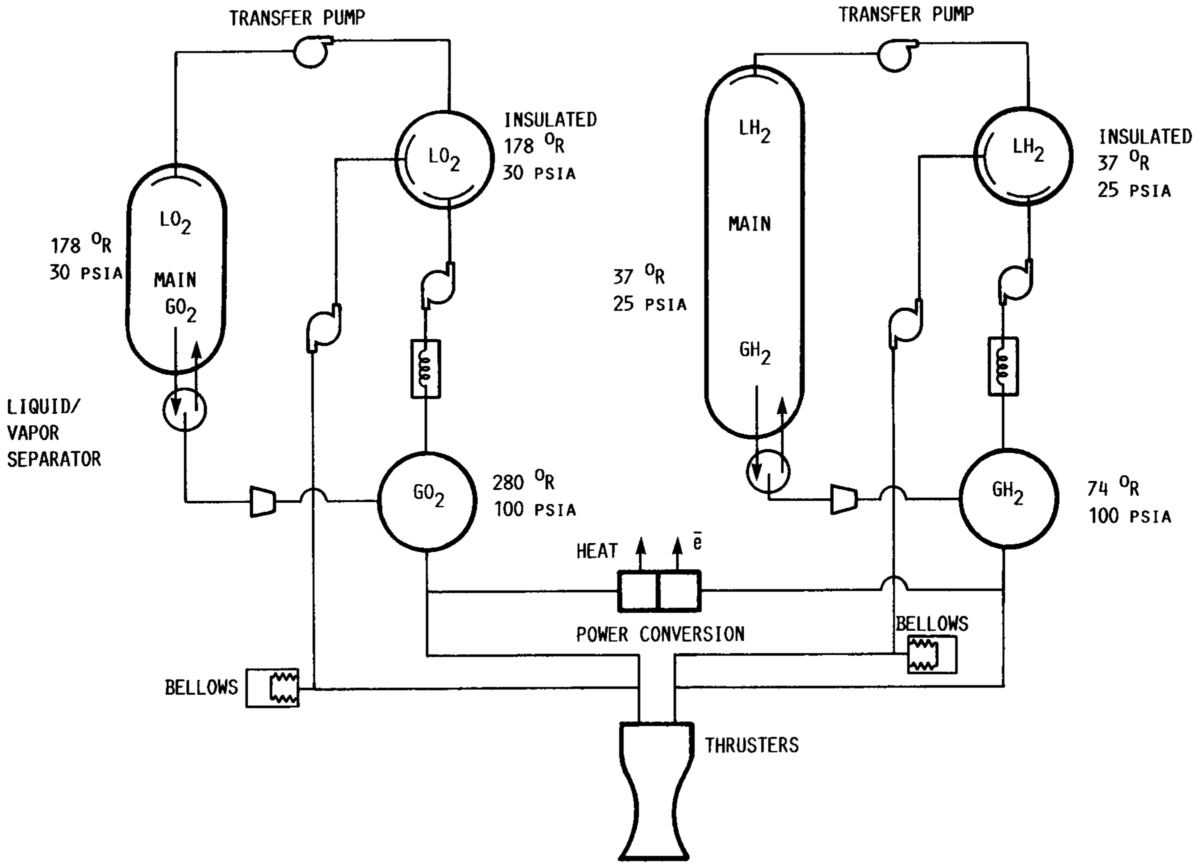

FIGURE 6. - SCHEMATIC FOR SCAVENGING PROPELLANTS

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Report Documentation Page

1. Report No.	2. Government Accession No.	3. Recipient's Catalog No.
NASA TM-100237		

4. Title and Subtitle	5. Report Date
Auxiliary Propulsion Technology for Advanced Earth-to-Orbit Vehicles	
	6. Performing Organization Code
7. Author(s)	8. Performing Organization Report No.
Steven J. Schneider	E-3783
	10. Work Unit No.
	506-42-81
9. Performing Organization Name and Address	11. Contract or Grant No.
National Aeronautics and Space Administration Lewis Research Center Cleveland, Ohio 44135-3191	
	13. Type of Report and Period Covered
12. Sponsoring Agency Name and Address	Technical Memorandum
National Aeronautics and Space Administration Washington, D.C. 20546-0001	14. Sponsoring Agency Code

15. Supplementary Notes

Prepared for the 1987 JANNAF Propulsion Conference, San Diego, California, December 15-17, 1987.

16. Abstract

The payload which can be delivered to orbit by advanced Earth-to-Orbit vehicles is significantly increased by advanced subsystem technology. Any weight which can be saved by advanced subsystems design can be converted to payload at Main Engine Cut Off (MECO) given the same launch vehicle performance. This paper is concerned with the auxiliary propulsion subsystem and the impetus for the current hydrogen/oxygen technology program. A review of the auxiliary propulsion requirements of advanced Earth-to-Orbit (E-T-O) vehicles and their proposed missions is given first. Then the performance benefits of hydrogen/oxygen auxiliary propulsion are illustrated using current shuttle data. The proposed auxiliary propulsion subsystem implementation includes liquid hydrogen/liquid oxygen (LH_2/LO_2) primary Reaction Control System (RCS) engines and gaseous hydrogen/gaseous oxygen (GH_2/GO_2) vernier RCS engines. A distribution system for the liquid cryogens to the engines is outlined. The possibility of providing one "dual-phase" engine that can operate on either liquid or gaseous propellants is being explored, as well as the simultaneous firing of redundant primary RCS thrusters to provide Orbital Maneuvering System (OMS) level impulse. Scavenging of propellants from integral main engine tankage is proposed to utilize main engine tank residuals and to combine launch vehicle and subsystem reserves.

17. Key Words (Suggested by Author(s))	18. Distribution Statement
Auxiliary propulsion; Earth-to-orbit vehicles; Integrated propulsion; Liquid rockets; Gaseous rockets; Propellant scavenging	Unclassified - Unlimited Subject Category 20

19. Security Classif. (of this report)	20. Security Classif. (of this page)	21. No of pages	22. Price*
Unclassified	Unclassified	15	A02

NASA FORM 1626 OCT 86

*For sale by the National Technical Information Service, Springfield, Virginia 22161