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Selection of an Optimized Integrated Propulsion System

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An optimized integrated propulsion system (IPS) for transfer of payloads from Shuttle parking orbit to various mission orbits was defined from a series of trade studies. The concept allows optimization of spacecraft and upper-stage design in that it balances the STS length and weight cost formulas, shares structural and housekeeping subsystems, and can be readily adapted to many missions by propellant tank resizing. Propulsion subsystem reviews have been made resulting in the selection of a low-thrust bipropellant approach which enables deployment of all satellite appendages and system testing while still attached to the STS Orbiter. The concept is compatible with extra vehicular activity (EVA) support for resolution of contingencies and for retrieval.

Introduction

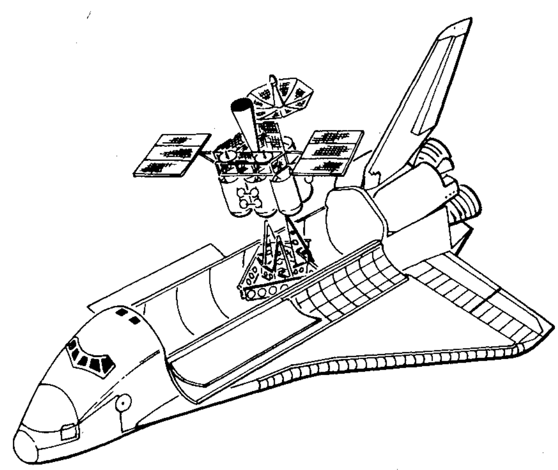
THE STS will be capable of placing payloads weighing up to 65,000 lb into low Earth orbit. The Shuttle Orbiter will be manned by a crew that is capable of assisting on-orbit assembly, payload deployment, on-orbit checkout, and satellite retrieval. These capabilities are new and need to be exploited through future satellite designs in order to reduce overall system cost and to improve reliability.

Work conducted in the past two years investigated an integrated propulsion system design which lends itself to low development cost, complements the features of the STS, and avoids separate duplicative subsystems for orbit transfer. The concept shown in Fig. 1 is one which extends the capabilities of the propulsion subsystems usually included in all satellites, for final injection and on-orbit maneuvers, to provide additional impulse for orbital transfer following egress from the Shuttle Orbiter. Figure 1 illustrates the concept with a representative payload, airborne support equipment (ASE), and a listing of key features. As in the case of normal on-orbit propulsive maneuvers, the satellite is dependent on its own housekeeping capabilities, attitude control, power, and telemetry. In order to use the satellite housekeeping subsystem for orbit transfer, the satellite must be self-sufficient prior to or immediately following Shuttle separation. Since the satellite with its total propellant load can be stored on-orbit, it is practical to launch early and commit to precise launch windows from orbit thereby avoiding launch range difficulties and/or range weather problems.

Mission Requirements

Mission requirement studies generated a set of desirable goals and key design requirements for a standard integrated propulsion system concept. The requirements include the capability of economically placing between 2500 and 10,000 lb in geosynchronous orbit. Corresponding performance for other orbits such as critically inclined and polar also had to be provided. The concept should have a low weight ratio, preferably less than 0.1 (weight of inerts and residuals divided by usable propellant weight). It is desirable to permit staging of empty tanks for high-performance missions.

The total airborne support equipment (ASE) weight used for satellite support during launch and return and for payload ejection should be minimized. Realistic ASE weights of less than 3000 lb and use of no more than 5 ft of the Shuttle bay,



- SHUTTLE OPTIMIZED
  - DESIGNED TO SHUTTLE'S WEIGHT AND LENGTH COST FORMULA
  - ALL APPENDAGES DEPLOYED WHILE ATTACHED TO THE SHUTTLE
  - ENABLES ON-ORBIT TESTING
  - PROVIDES PRECISE LAUNCH TIMING
  - COMPATIBLE WITH EVA SUPPORT
  - ENABLES RETRIEVAL
- LOW THRUST BIPROPELLANT PROPULSION
  - LOW ACCELERATIONS AND SOFT RIDE
  - MISSION FLEXIBILITY BY TANK SIZING
- PROPULSION INTEGRATED WITH SPACECRAFT
  - SHARES HOUSEKEEPING EQUIPMENT OF SPACECRAFT
  - SIMPLE GUIDANCE HARDWARE

Fig. 1 Integrated propulsion system.

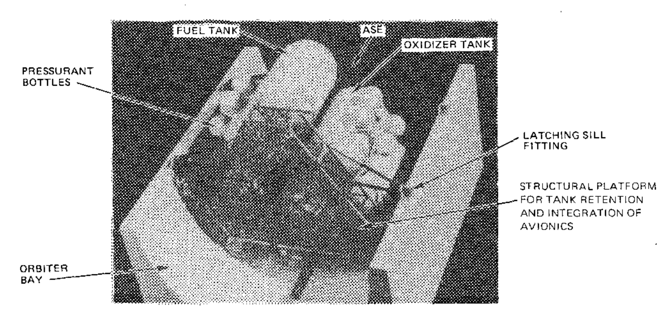


Fig. 2 Concept model.

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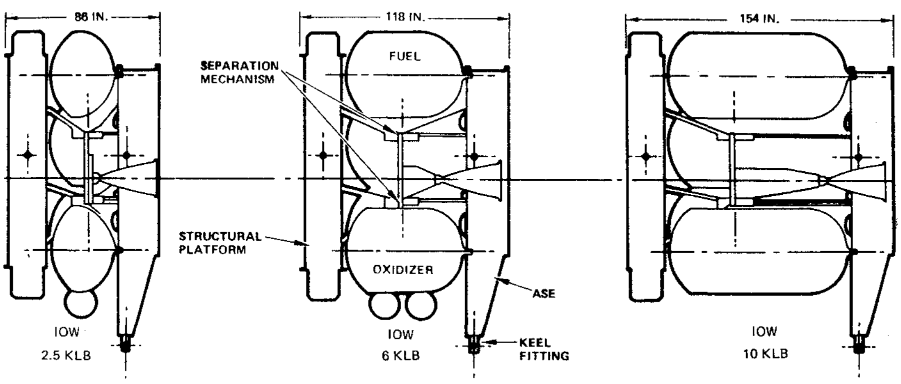

Fig. 3 Initial on-orbit weight (IOW) (lb) (synchronous equatorial).

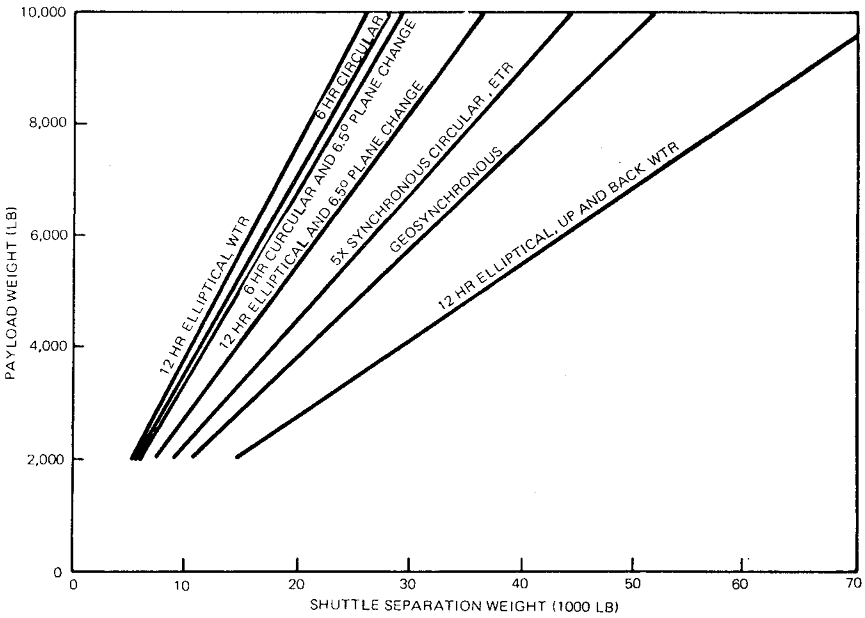

Fig. 4 Typical performance in high-altitude orbits.

Table 1 Configuration comparison

	Integral propulsion system:			IUS
Nominal beginning of life spacecraft weight, klb	2.5	6.0	10.0	5.0
Propellant weight, klb	9.5	22.7	37.8	27.7
Separation weight klb	12.6	30.2	50.3	37.3
Length, in.	86	118	154	201

including tilting for satellite ejection were calculated. All appendages (solar arrays and antennae) should be deployed while still attached to the Orbiter which enables visual verification from the Orbiter. Following deployment the appendages must be protected from exposure to high accelerations. The thrust range should be a minimum of 600 lb and produce a maximum acceleration of less than 0.2 *g*, to prevent structural weight penalties for the deployed equipments.

Concept Description

The design developed to meet these requirements is shown in Fig. 2 as generated from a 1/20 scale model. The model represents a configuration capable of placing 6000 lb in geosynchronous orbit. This concept exhibits mission flexibility in that payload capabilities can be expanded or reduced by changes in propellant tank length. Figure 3 shows this scaling relationship, while Table 1 shows a comparison of selected weights and overall length. Figure 4 shows the payload capability of the system for various separation weights out of the Orbiter payload bay.

Propulsion Trades

The initial propulsion trades were performed to determine the most effective propulsion system alternative for transferring 5000-10,000-lb payloads from Shuttle orbit to geosynchronous orbit. The use of flight qualified propulsion hardware was a primary requirement of the study. Further goals of the study were to determine an estimate of development and recurring costs, and technical risk associated with each concept. The systems selected for this detailed trade study were:

1) Chemical liquid including monopropellant, Earth storable bipropellant, high performance space storable cryogenics, resistojets, and water electrolysis;
2) solids in various forms;
3) hybrid liquid/solid;
4) electric-ion;
5) hybrid chemical/ion.

These concepts wre divided into low-thrust and high-thrust categories. Several of them could accommodate either high- or low-thrust configurations and were included in both categories.

Propulsion Trade Study Results

Figure 5 and Table 2 summarize the results of the initial propulsion concept trades. These trades were conducted in parallel with the system concept trades and do not necessarily reflect optimum configurations. The data are most effective

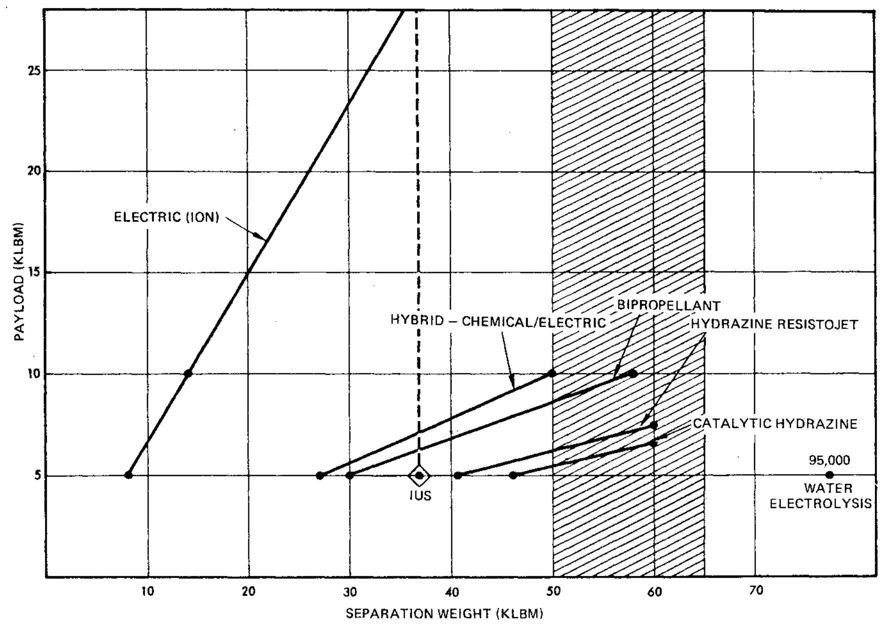

Fig. 5 Low thrust systems—separation weight vs payload capability.

Table 2 Low-thrust systems summary

System	Payload (lb)	Separation Weight (lb)	Total Propellant Weight (lb)	ISP (sec)	F (lbf)	Apogee Burn (sec)	Perigee Burn (sec)	Time To GSO (days)	Power (W)	Weight, Power (L)	Comments and Issues
w Thrust Storable ɔropellant (2 stage)	5000	30000	22000	296	1200/600	2400	4200	6	--	--	State of the art system,Co figured with developed co ponents - low technical ri approach.
	10000	58000	44000			2700	8400	12	--	--	
ɛctric-Ion ɔiral Orbit)	5000	8000	1250	2900	0.174	5173 (hr)		309	31×10^3	1165	Total system needs to be qualified. Large solar arra not state of the art.Van A belt degradation significa (40%). Electronic black b hardening not evaluated.
	10000	14000	2140			9921 (hr)		537	31×10^3	1165	
/brid-Chemical ectric	5000	27000	19200/237	296/2900	600/0.174	N/A	N/A	55	18×10^3	700	System feasible but high with only modest payloa advantage. Could reduce t weight by reducing numb of thrusters and power. F quires both systems to be qualified.
	10000	50000	34200/423					100			
lonopropellant ydrazine Catalytic)	5000	46000	37300	235	600	2350	10800	6	--	--	System not practical beca of low I_{sp} and concomit large throughput requirem for thrusters.
	6520	60000	48650					8			
lonopropellant lydrazine-HiPEHT 3 stage - spiral orbit)	5000	40600	32200	290	0.6	--	--	180	2400	180	System not practical bec of low thrust and large throughput requirement.
	7400	60000	47200					264			
/ater Electrolysis	5000	95000	73700	345	5			60	130×10^3	4250	System not practical bec of excessive weight of el trolisis cells and large pc requirement
olids-Modular Low Thrust)											Concept not feasible if (required. Minimum migl 0.5 G. Low burn rate pr lants would require maj development. Developm of new motors also requ

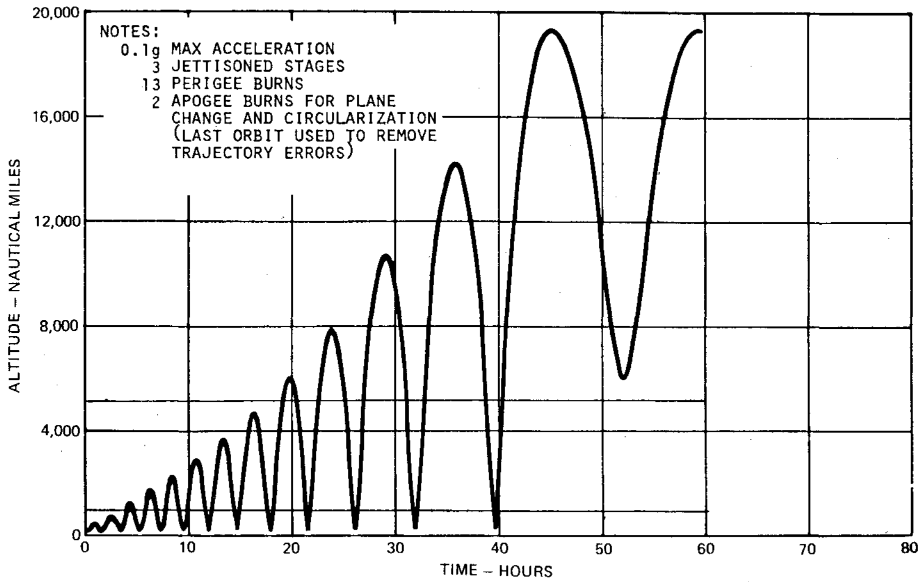

Fig. 6 Typical time history—transfer to synchronous altitude.

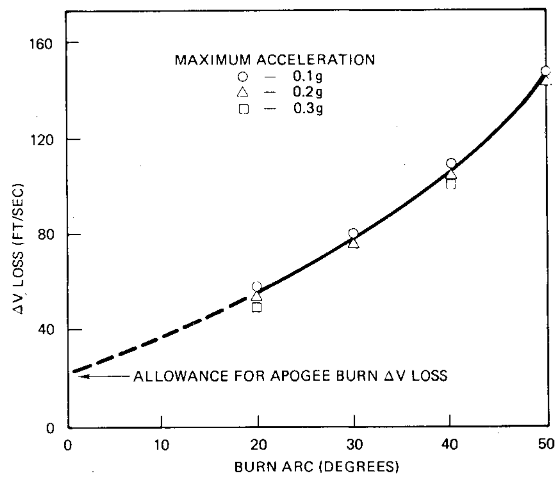

Fig. 7 Finite burn arc velocity inefficiency, geosynchronous mission.

in sorting concepts on a relative basis for subsequent refinement.

At this point in the program, it was decided to use an all low-thrust liquid bipropellant system due to other mission requirements.

Figure 5 presents the comparative payload capabilities for the low-thrust system. The abscissa of the figure shows the separation weight of the propulsion system as loaded into the space Shuttle Orbiter cargo bay. The maximum orbiter capability is 65,000 lb but the shaded area shows an approximate allowance for the balance of the stage weight (and ASE) over and above the calculated weight of the propulsion system itself. A review of existing stages indicates that this value is approximately 15% of the propulsion system weight. The ordinate of the figure shows the payload capability of the systems for a given separation weight.

Figure 5 can be used in two ways. The payload limit for any of the system concepts can be found by following that system's curve until the Shuttle orbiter maximum capability is reached (minus the shaded area allowance for stage weight and ASE). Thus, the liquid bipropellant system can deliver 10,000 lb of payload to geosynchronous orbit with a separation weight of 58,000 lb. If the stage and ASE weight are estimated to be higher than the 7000-lb allowance available at this point, the payload capability is found by backing down the system curve.

The other data available from the figure are a direct comparison with the IUS on a propulsion system basis. By vertically following the dotted line from the IUS notation, a comparison is attained. As shown, the electric propulsion system capability exceeds that of the IUS by about 24,000 lb. The chemical system's capabilities are not quite so remarkable due to their lower specific impulse performance.

Table 2 presents the key parameters and issues for the low-thrust systems. As can be seen from the last column, only the first three propulsion systems listed are practical.

High-vs-Low Thrust, Ascent Trajectory Trades

An initial tradeoff was made to determine thrust level vs velocity losses vs vehicle weight impacts. The ascent propulsion system must be sized to provide the total velocity needed to achieve the final orbit. The velocity requirement is a function of the thrust level and the arc over which propulsive burns are made. In the typical high-thrust, single perigee-burn case, the burn time is very short and the arc is short. The velocity required to inject the satellite into a Hohmann transfer is very close to the ideal velocity. If a low-thrust system is used, many burns are required at successive perigee passes to raise apogee to geosynchronous altitude. Figure 6 shows a typical low-thrust ascent trajectory to geosynchronous orbit.

The difference between the velocity requirement and the ideal velocity can be considered to be a velocity inefficiency. Figure 7 shows the low-thrust velocity inefficiency as a function of perigee-burn arc and maximum acceleration. It is seen that the latter has only a small influence. A typical value for the velocity inefficiency at the apogee burns has been included in the figure. It should be noted that the use of the high-thrust perigee-burn system entails velocity penalties caused by pointing errors. The low-thrust system with its many burns can reduce this source of error by correcting pointing errors between burns. Although the velocity inefficiency can be made as small as one pleases by restricting burn arc, the number of burns and the total ascent time are increased.

Pump Fed vs Pressure Fed

A review of the options available for propellant transfer were influenced by the engine selection. However, the review of pumps available for the developed engine candidates resulted in the selection of a pressurized system in order to avoid pump development and qualification costs.

Engine Selection

The choice of engines was limited to three candidates because of the constraints imposed to consider only highly developed engines and by our decision to use Earth storable bipropellants. The coast time to high-altitude orbits and the possibility of satellite retrieval following mission completion eliminate cryogenic propellants due to expected evaporation losses. The engines considered were the R-40 STS-RCS-primary thruster, the MMBPS, and the Viking orbiter main engine. The Viking engine was not a serious contender because of the low-thrust (300 lbf) resulting in higher gravitational losses during orbit transfer, and the high inlet pressure requirement (225 psia) causing increased tank weight. The MMBPS engine was also eliminated because of low thrust. A summary comparison of the three engines is shown in Table 3.

The other orbiter bipropellant engine, the OMS (STS orbit maneuvering system engine), was eliminated previously in the trade because of high thrust and accelerations that would be too great to enable appendage deployment in low Earth orbit without an added structural weight penalty. The R-40 was chosen for the bus concept and its characteristics are shown in Fig. 8.

Staging Concepts

Figure 9 shows a number of the concepts investigated as part of the analysis of staging concepts. The objective of the study was to obtain configurations which were easily staged while maintaining a high propellant loading factor (PLF). The measure of propellant loading factor is propellant tank area divided by available platform area. The three-stage configuration without core tanks was selected and has a propellant loading factor of 0.49, but has a simple staging. Tank diameter has been increased to improve the loading factor up to 0.54.

A representative staging mechanism concept was configured to demonstrate the feasibility of separating the lines and jettisoning the tanks. This concept is illustrated in Fig. 10. The line separation and tank jettisoning are critical to the success of the propulsion system. Tanks are mounted on their forward end domes during low-thrust operation. A pin at the aft dome end provides additional support during STS launch. A quick disconnect mechanism similar to the disconnect built and qualified on the Apollo and other space programs is used. The packaging of the disconnects into a unit having high- and low-pressure gas, liquid disconnects and separation springs in a single package represents a new development. The separation hinge device providing an unlocking action is controlled by the release of the quick disconnect followed by the rotation of the tanks (tank pairs).

Initiation of tank pair staging is based on engine loss of thrust. The thrust decay is implied from monitoring a thrust chamber pressure transducer. Use of the pressure transducer output will enable the start of a staging sequence only during predetermined burn periods. Following a suitable sampling time, a fire signal is sent to the quick disconnect ordnance releasing the clamp band. Springs in the disconnect housing provide energy to rotate and eject the tanks.

Table 3 Engine options

Application	MMBPS	Viking-75	R-40 Space Shuttle RCS
Thrust, lbf	100	300	870 (600-1300)
Specific impluse, s	295 at 50:1	296 at 100:1	298 at 100:1
Mixture ratio, WO/WF	1.64	1.51	1.51
Inlet pressure, psia	185	225	150 at 600 lbf
Size			
Length, in.	15	29	35
Diameter, in.	7	14	20
Weight, lb	12	18	18

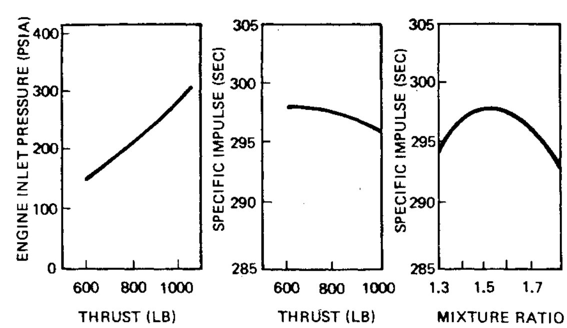

Fig. 8 R-40 performance characteristics.

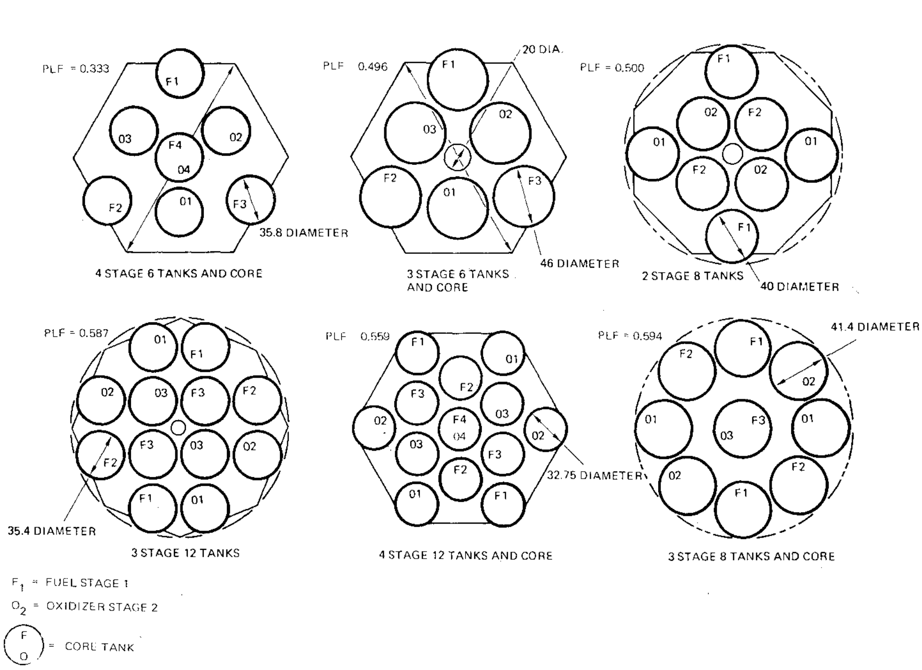

Fig. 9 Stage propellant tankage arrangements.

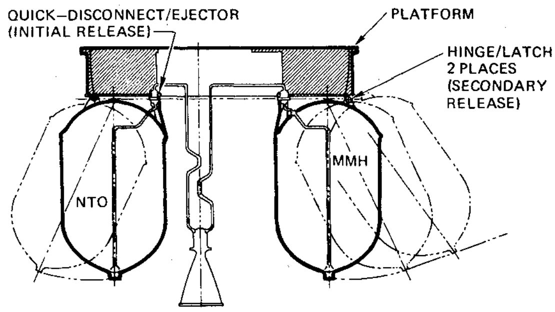

Fig. 10 Tank staging.

Propellant Tank Study

The propellant tank study compared various tank materials and the effect of design criteria to determine the comparative weight, cost, and availability of candidate design options. The materials investigated included heat-treated and annealed Ti-6Al-4V titanium alloy, cryogenically formed 301 stainless steel, and 2021-T62, 2219-T87, 6061-T6 aluminum. The study also examined the effect of using safety factors of 2.0 and 1.25. The current concept is based on using a safety factor of 1.25 over the maximum operating pressure for operation outside the Shuttle bay. Since some pressure in the tanks is required to stabilize the N_2O_4 and also help minimize structural weight, it was assumed that the tanks would be

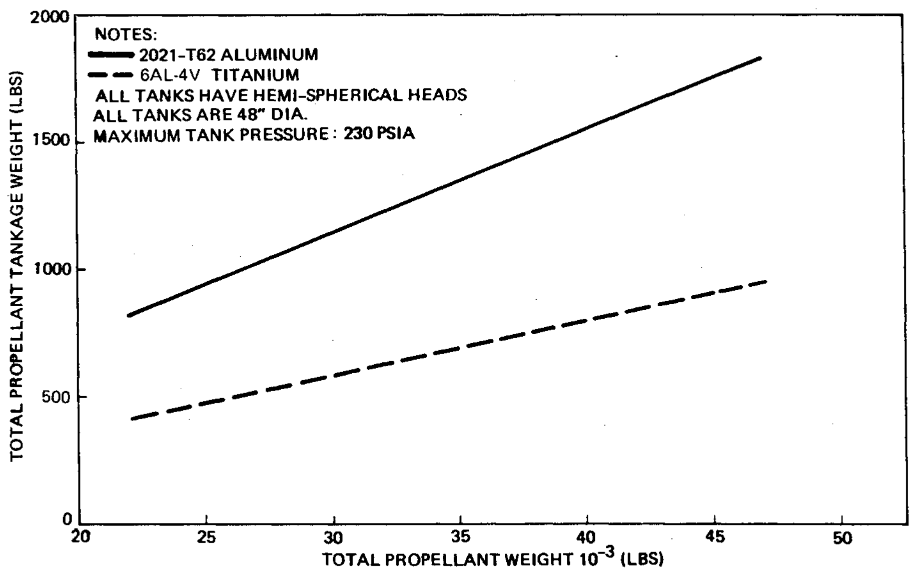

Fig. 11 Total propellant tank weight vs total propellant weight.

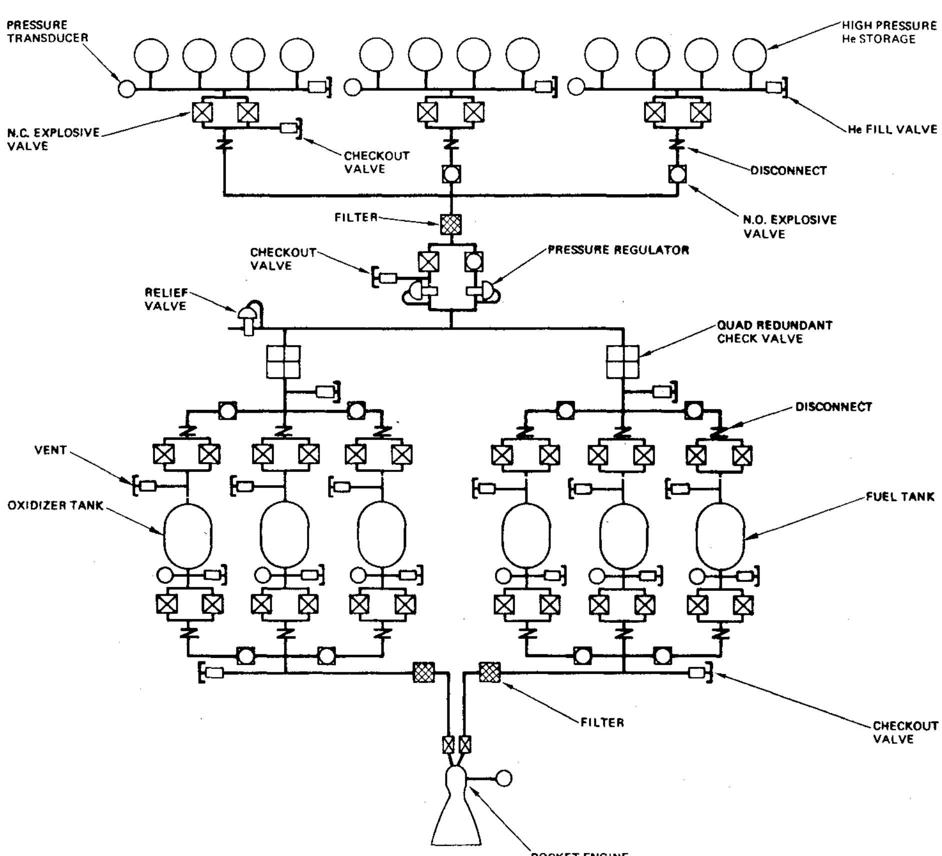

Fig. 12 IPS propulsion subsystem schematic diagram.

pressurized to 50 psia during STS operations. This results in a factor of safety of approximately 4, which exceeds the STS design criteria.

The fracture mechanics properties for N_2O_4 were used in the analyses. The assumption was that the tanks were designed by the pressure loads. Based on this assumption, the titanium tanks were determined to be the lightest, with the steel tanks about 10% heavier, and the highest strength aluminum tanks 2021-T62 approximately twice the weight of the titanium tanks as shown in Fig. 11.

The aluminum tanks were found to be superior on a cost basis as well as being more readily obtainable in terms of manufacturing lead time (15 months compared with 34 months for titanium). These later factors outweighed the added system weight for most of the missions anticipated.

Pressurant Tanks

The pressurant tanks were selected on the basis of a study which compared various composite material tank designs with titanium and cryogenically formed stainless tanks. The Kevlar tanks, which appear to be a good compromise between cost and weight, were ruled out along with S-glass/epoxy tanks because pressurization life data are not available to substantiate the possible long-term storage requirements. The difference in weight and cost between steel and titanium tanks was not a significant factor, and cryoformed 301 stainless tanks were selected for use in the performance calculations.

Summary of Trades

Based on the results of the above studies, the baseline IPS ascent propulsion system design was selected to be the following:

1) Gimballed R-40 engine providing 700 lbf thrust;
2) six 6061-T6 aluminum propellant tanks (tank cylindrical section is lengthened for added payload capability);
3) twelve cryoformed 301 stainless-steel helium pressurant tanks;
4) propellant tanks are staged in pairs along with their particular pressurant tanks;
5) propellant and pressurant line quick disconnects used for staging;
6) multistages used for heavy payloads;
7) multiburn Hohmann transfer orbit scheme used;
8) aluminum mounting structure uses isogrid design.

The design concept is shown schematically in Fig. 12.

Conclusions

Several of the propulsion concepts studied for future upper-stage application were determined to be practical and to offer certain advantages over current designs. The solid propulsion techniques offer the lowest nonrecurring costs when applied to individual systems. The liquid propellant techniques have a lower prorated development cost when applied to several systems because of their inherent total impulse flexibility. Recurring costs of both techniques are approximately equal, resulting in a selection based on nonpropulsion factors such as ease of satellite integration and low acceleration during powered flight.

The availability of the STS with its own housekeeping and payload services enables upper-stage designs which are more integral with satellite design. Expansion of the basic satellite propulsion capability can provide orbit transfer impulse following egress from the orbiter. Studies conducted have validated this concept for conditions when the satellite is self-sufficient prior to or immediately following STS ejection. Low-thrust propulsion with corresponding low acceleration is a complementary feature necessary for this early self-sufficiency. The velocity losses due to low acceleration are acceptable for the missions studied as long as time to final orbit is not critical. The present state of the art also provides qualified propulsion equipment necessary for this approach.

No fundamental limitations to the concept have been uncovered in the two years of intensive study. Continuing efforts are aimed at refinement of the concept and the acquisition of test data on engine performance and separation mechanisms.