

BEYOND LOW EARTH ORBIT  
AN OVERVIEW OF ORBIT-TO-ORBIT STAGES

Joseph F. Loftus, Jr.  
Warren L. Brasher

Lyndon B. Johnson Space Center  
Houston, Texas 77058

ABSTRACT

The transition of U.S. space launch activities from the expendable launch vehicles to the Space Transportation System (STS) (Shuttle) has led to extensive new developments in upper stages. Additional new developments can be expected as Space Station availability provides a different context for stage operations.

It is expected that upper stages will become reusable and spacebased. For some applications, both three burn strategies and aerobraking configurations are more attractive than all propulsive systems. The development and demonstration of storable propellant transfer systems has been accomplished, and experimental facilities for determining handling techniques for cryogenics are underway.

Current Status

Figure 1 illustrates the existing developed STS compatible stages. In this fig. and the next two, details of the stages are characterized in terms of the manufacturer, primary spatial dimensions, performance parameters, mass properties, delivery capability, profile view, planned schedules, and the nature of the enterprises which sponsor and execute the development. The objective in displaying this array of data is to characterize the number and diversity of the alternatives being developed and studied.

The payload assist module (PAM) D is delta compatible, as well as STS compatible, and was developed by the McDonnell-Douglas Company as a commercial venture under an agreement with NASA that the government would not develop an equivalent stage. The PAM-D has flown 16 times on the STS and 11 times on the delta launch vehicle. Currently, an additional 28 flights on the STS are scheduled.

As part of the agreement, McDonnell-Douglas also developed the PAM-A to provide an STS launch capability equivalent to the Atlas-Centaur. This unit has not been scheduled for flight.

The inertial upper stage (IUS) was developed to transition payloads developed to fly on the Titan 34D, to the STS. To date it has flown once on the Titan 34D and once on the STS. The next scheduled flight of the IUS, on the STS, is in early 1986. It may also be used on the complementary expendable launch vehicle (ELV) of the U.S. Air Force.

Figure 2 illustrates a number of stages currently under development to fly on the STS. These stages are in development and funded for certification but not all have identified a first use payload and flight schedule date.

The Centaur-G is a wide-body modification of the Centaur stage used on the Atlas and Titan expendable vehicles. It has been developed to provide geosynchronous orbit delivery of large satellites and dimensionally sized to make effective use of the STS Orbiter cargo bay width.

The Centaur-G (G<sup>1</sup>) Prime is a larger version of the stage developed to support NASA planetary missions. Its first flights are scheduled for the May 1986 Galileo and Ulysses missions. In addition to the larger hydrogen tank, the G<sup>1</sup> stage has a lower mixture ratio, a larger engine expansion ratio engine cone and 6 seconds increase in specific impulse.

The transfer orbit stage (TOS) is being developed by Martin-Marietta Corporation for Orbital Science Corporation, a new company formed to develop and market the system. The TOS uses the same first stage motor as the IUS and is targeted for payloads too heavy for the PAM and not large enough to warrant the use of a Centaur-G or G<sup>1</sup> system. It offers a different avionics and data processing system than the IUS.

Orbital Sciences is also developing an apogee and maneuver stage (AMS) which can be used independently or as an apogee stage in conjunction with the TOS. In this configuration the TOS/AMS has

### UPPER STAGES (EXISTING)




CHARACTERISTICS	PAM D	PAM A	IUS (TWO STAGE)
<b>STAGE:</b>			
MANUFACTURER	McDONNELL-DOUGLAS (MDAC)	MDAC	BAC
LENGTH (m)	2.4	2.3	1.1
DIAMETER (m)	1.3	1.2	2.1
<b>ENGINE:</b>			
MANUFACTURER	THIokol	THIokol	CGO
TYPE	(STAR 48)	(MINUTE MAN 3)	
NUMBER	1	1	SRM-1 SRM-2
FUEL	SOLID	SOLID	SOLID
COMPOSITION	TP-H-3340	AWP-2064	HTPB
TOTAL THRUST (N)	88,200	187,800	208,000 81,200
SPECIFIC IMPULSE SEC	285.8	274.3	282.9 300.8
BURN TIME SEC	30.9	68.8	183 104
<b>STAGE:</b>			
PAD WEIGHT (kg)	2,100	3,750	14,800
PROPELLANT WEIGHT (kg)	2,000	3,430	9,710 2,730
BURN-OUT WEIGHT (kg)	100	310	1,100 1,140
SUPPORT EQUIP. WEIGHT (kg)	1,740	2,000	3,350
<b>PAYLOAD<sup>1</sup>:</b>			
TO GEO - ONE WAY STAGE (kg)	826*	988*	2,310
TO GEO TRANSFER ORBIT (GTO) (kg)	1,200	1,800	~4,000
<b>ILLUSTRATION:</b>			
<b>NOTES:</b> 1. REF: 38.5° INCLINATION & 180 N.M. CIRCULAR REQUIRES RISE STAGE	 PAM D	 PAM A	 IUS (TWO STAGE)
<b>SCHEDULE:</b>			
START DATE	1976	1977	1978
OPERATIONAL DATE	1982	1982	1982
<b>TYPE OF DEVELOPMENT</b>	COMMERCIAL	COMMERCIAL	US GOVT
<b>SPONSOR</b>	MDAC	MDAC	USAF

Fig. 1. Stages that are currently available and compatible with either Shuttle or expendable launch vehicles.

slightly better performance than the IUS due to the higher specific impulse of the storable propellants relative to the solid motor of the IUS upper stage.

The Italian research interim stage (IRIS) is being developed by Aeritalia under the sponsorship of the Italian Government. Its capability is a little less than the PAM-D. Its first use is scheduled for the LAGEOS mission in late 1987.

The Pam-D II is a larger version of the PAM-D system. The solid rocket motor is larger, as is the airborne support equipment (ASE) structure and spin table. The avionics system is not changed in any significant detail. Its first flight is in late 1985 in support of Satcom KU-1.

RCA Astronautics has been developing the Shuttle compatible orbit transfer stage (SCOTS) as a perigee or transfer orbit stage for future geosynchronous satellites. Its capability is slightly greater than the PAM-A, but not as large as the TOS. It will first fly with the RCA Direct Broadcast Satellite in 1987.

In fig. 3, some but not all, of the stages being studied for future development by commercial firms are illustrated. Orbital Systems Corporation is examining a larger version of the AMS and TOS/AMS combinations to provide performance comparable to the Centaur-G.

The satellite transfer vehicle (STV) is a storable bipropellant stage that has been studied by Scott Science and Technology and British Aerospace. It is designed to accommodate payloads too large for the PAM-A, but not so large as to need a TOS.

The liquid propulsion module (LPM) is a storable bipropellant stage using a pump fed engine to get higher specific impulse; therefore, mass fraction efficiency. The Aerojet Technical Systems Company has proposed it as a stage to accommodate payloads from the PAM-D close up to two and a half times that mass. The Transtar 1 engine for the stage is in development but no date for stage availability has been published.

**UPPER STAGES (UNDER DEVELOPMENT)**




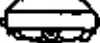




CHARACTERISTIC	CENTAUR G	CENTAUR G	TOE	AME	TOE/AME	HRB	PAM-D-H	SCOTS
<b>STAGE MANUFACTURER</b>	GENRAL DYNAMES (G.D.)	G.D.	ROCKET-MANUFACTURE (RMC)	ROCKET-MANUFACTURE (RMC)	ORC	ADRIALIA	ROPERCO-DOUGLAS (RPA-C)	ACA ASTRO
<b>LENGTH (M)</b>	1.8	2.4	3.2	1.7	2.0	2.2	2.4	2.8
<b>DIAMETER (M)</b>	1.2	2.1	1.4	1.2	1.2	1.1	1.6	1.8
<b>ENGINE MANUFACTURER</b>	PRATT-WHITNEY	PRATT-WHITNEY	CRJ	ROCKET-THREVE (RTH)	CRJ-42	SPR-SPERA (SPAS)	TROROL	TROROL
<b>TYPE</b>	R. RA-3-35	R. RA-3-35	SP-1	RS-41	SP-1-35-47	TR	TRP	RS-1
<b>NUMBER</b>	1	2	1	1	1	1	1	1
<b>FUEL</b>	$LO_2/LH_2$	$LO_2/LH_2$	SOLID	$H_2O_2/LO_2$	SOLID	$H_2O_2/LO_2$	SOLID	SOLID
<b>COMPOSITION</b>	5:1	5:1	HTPB	1:6:1	HTPB 1:2:1	HTPB	-	TPS
<b>TOTAL THRUST (KN)</b>	12,000,000	14,000,000	20,000	11,000	20,000-1,000	700	7,000	10,000
<b>SPECIFIC IMPULSE (SEC)</b>	460	460	30	310	300-310	300	-	30
<b>BURN TIME (SEC)</b>	200	200	100	1,000	100-150	700	10	100
<b>STAGE</b>		20						
<b>NOZ. HEIGHT (MM)</b>	10,000	10,000	10,000	4,200	10,000	1,000	5,000	4,200
<b>PROPELLANT HEIGHT (MM)</b>	12,000	10,000	5,700	3,200	3,700-5,200	1,200	2,300	1,000
<b>BURN-OUT HEIGHT (MM)</b>	1,200	1,000	1,000	900	900-1,200	500	100	500
<b>SUPPORT EQUIP. HEIGHT (MM)</b>	1,100	1,100	1,100	1,100	1,000	500	1,000	1,100
<b>PAYLOAD</b>								
<b>TO GEO. ORB (KG)</b>	4,000	4,000 <sup>1</sup> 4,000 <sup>2</sup>	2,000 <sup>3</sup>	600 <sup>4</sup>	2,000	500	500 <sup>5</sup>	1,100 (REUSABLE)
<b>TO GEO. TRANSFER ORBIT (KG)</b>	NOT APPLICABLE	NOT APPLICABLE	2,000	1,000	1,000	500	1,000	1,000
<b>ILLUSTRATION</b>								
<b>NOTES</b>	1. ASP 25% INCREASE IN MASS IS CIRCULAR 2. REQUIRES SIDA STAGE 3. STAGE PROPELLANT FULL 4. STAGE PROPELLANT PAYLOAD AND SIDA LIMITED TO 50% IS 5. PAM-D-H							
<b>SCHEDULE</b>								
<b>START DATE</b>	1988	1988	1988	1988	1988	1988	1988	1988
<b>OPERATIONAL DATE</b>	1988	1988	1988	1988	1988	1988	1988	1988
<b>TYPE OF DEVELOPMENT</b>	US GOVT	US GOVT	COMMERCIAL	COMMERCIAL	COMMERCIAL	GOVERNMENT	COMMERCIAL	COMMERCIAL
<b>SPONSOR</b>	NASA	NASA	DIGITAL SCIENCE CORP (DSC)	ORC	ORC	ITALY	RPA-C	ACA

Fig. 2. These are stages that are being developed to be compatible with the Shuttle. Some are also compatible with expendable launch vehicles.

The high performance propulsion module (HPPM) is a joint effort by Aerojet Technical Systems Company and Ford Aerospace to examine a somewhat smaller system also based on the Transtar engine and sized to roughly twice the PAM-D performance.

It is noteworthy that so many of the developments and studies identified here are commercial undertakings not sponsored by government agencies. It also is worth noting that there is significant interest in the storable fluid systems rather than continued reliance upon solid motors.

**Future Options**

The availability of both low and high pressure pump fed engines in the late 1980's is stimulating a significant number of storable propellant stage studies at a number of U.S. aerospace companies. The high specific impulse, (328-343 seconds), the volumetric density, and consequently STS payload bay use factor efficiency, long-term storability, multiple starts, thrust throttling and potential reusability make such systems both economically and operationally attractive. While some of the stages being studied are expendable as currently conceived, increasing attention is being given to reusable configurations.

There are several influences which make reusable stages attractive, particularly when a Space Station is available. A space-based, reusable stage saves a large fraction of the cost of an expended stage, reduces the requirement to transport the stage itself to orbit, both a mass and length factor, and allows the stage structure and tankage weight to be reduced to a minimum since the stage need not carry propellant loads through the launch phase. The Space Station's availability makes both the staging of propellant and the recovery phasing operations to recover the stage more attractive and efficient, both operationally and economically.

A Space Station at 371.5 km (200.6 n. mi.) and 28.5° inclination, would be accessible every third day from the Kennedy Space Center. The Shuttle can deliver 20,154 kg (45,145 lb) to that orbital altitude. Table 1 illustrates several alternative altitudes and in-plane launch opportunities. For the Space Station to be an effective transportation mode and depot, it needs to be readily accessible and low enough to allow substantial mass to be delivered to the station. To meet other station functions, particularly microgravity operations and to minimize drag, higher altitudes for the station are attractive. Final orbit parameters will be selected as a part of the Phase 2 Trade Studies.

### UPPER STAGES (PRE-DEVELOPMENT)






CHARACTERISTICS	AMS HIGH PERFORMANCE	TOS/AMS HIGH PERFORMANCE	STV	LPM	HPPM
STAGE MANUFACTURER	BAFTER-BAUMPTA (MBC)	MBC	BRITISH AEROSPACE	AERONEY TECH LTD (ATC)	ATC
LENGTH (m)	2.4	2.1	2.3	1.8	1.8
DIAMETER (m)	2.2	2.2	2.4	2.1	2.2
ENGINE MANUFACTURER	ATC	CSO-ATC	TBO	ATC	ATC
TYPE	TRAMSTAR 1	TRAMSTAR 1	TBO	TRAMSTAR 1	TRAMSTAR 1
NUMBER	1	1/2	1	1	1
FUEL	$N_2O_4$ / MMH	SOLID $N_2O_4$ / MMH	$H_2O_2$ / MMH	$N_2O_4$ / MMH	$N_2O_4$ / MMH
COMPOSITION	1:2	HTPB-12:1	TBO	1.2:1	1.2:1
TOTAL THRUST (kN)	16,700	10,000-16,700	TBO	16,700	16,700
SPECIFIC IMPULSE (SEC)	284	282-292	210	300	280
BURN TIME (SEC)	1000	100-1,000	TBO	1,000	—
STAGE					
PAG WEIGHT (kg)	15,100	26,700	11,700	6,600	6,500
PROPELLANT WEIGHT (kg)	6,600	6,700-6,800	6,600	6,600	6,200
BURN-OUT WEIGHT (kg)	1,400	282-1,200	1,400	900	750
SUPPORT EQUIP WEIGHT (kg)	1,900	2,700	TBO	—	—
PAYLOAD					
TO GEO - ONE WAY STAGE (kg)	1,500	6,000	ESTIMATED 1,100 - 1,200	1,300	1,300
TO GEO TRANSFER ORBIT (HTO) (kg)	5,700	10,000 <sup>a</sup> 6,710 <sup>b</sup>	1,000 - 1,000	4,000	2,000
ILLUSTRATION					
NOTES: 1. REF. 25° RECLINATION & 100 KIL. CIRCULAR 2. REQUIRES INCH STAGE 3. STAGE PROPELLANT FULL 4. STAGE PROPELLANT PAYLOAD AND A.S.E. LIMITED TO 2000 kg					
SCHEDULE					
START DATE	1984	1984	1984	—	—
OPERATIONAL DATE	1986	1988	1987	—	—
TYPE OF DEVELOPMENT	COMMERCIAL	COMMERCIAL	COMMERCIAL	COMMERCIAL	COMMERCIAL
SPONSOR	OSG	OSG	SCOTT SCIENCE & TECHNOLOGY	ATC	FORG AEROSPACE AND ATC

Fig. 3. These stages that are being studied are intended for use on the STS Shuttle.

The 371 km orbit is attractive because of the regular availability of large mass delivery. This is a slightly higher orbit altitude than the Shuttle now uses for geosynchronous stage and satellite deliveries. Characteristically, such flights have a load factor of 70 to 80 percent, using Shuttle engine settings of 104 percent and standard solid rocket boosters (SRB). If the Shuttle were to deliver current stage and spacecraft combinations (including in the spacecraft mass apogee kick motors (AKM)), to a station for subsequent deployment, it could also deliver 3300 to 7100 kg of additional payload which could be station supplies or storable propellants. If the station had reusable stages available, the Shuttle could deliver the mission objective spacecraft and propellants and attain a load factor of 100 percent because fluids can be jettisoned readily and this factor relieves many of the constraints which currently limit load factors.

Historically, more than half the mass to orbit is propellants for moving spacecraft to higher energy orbits. Propellants are, therefore, an ideal commodity to bring the load factor on every Shuttle flight to 100 percent and to accumulate at the station. A dozen flights delivering current solid propellant transfer and apogee kick stages could deliver as much as 35 000 kg of pro-

TABLE 1 STS DELIVERY CAPABILITY TO THE SPACE STATION AS A FUNCTION OF ORBITAL ALTITUDE

1 = 28.5

ALTITUDE	DELIVERY PERFORMANCE	IN-PLANE OPPORTUNITIES
500 km (270 n. mi.)	17073 kg (38 245 lb)	EVERY TWELFTH DAY
475.6 km (256.8 n. mi.)	17799 kg (39 870 lb)	DAILY
371.5 km (200.6 n. mi.)	20154 kg (45 145 lb)	EVERY THIRD DAY
320.8 km (173.2 n. mi.)	22487 kg (50 370 lb)	EVERY OTHER DAY

pellants at no increase in cost. A reusable stage with storable propellants and an Isp of 342 could deliver four 1786 kg (4000 lb) spacecraft to geosynchronous orbit with that much propellant. If the Shuttle carried only spacecraft and propellant to the station, there would be further savings because of some reduction in airborne support equipment currently carried to deploy those stages.

There are two significant sources of savings, the higher impulse of pump fed engines and stages that are not mass fraction determined by launch loads, and the increase in propellant available at the kinetic energy equivalent of the low earth orbit facility for no net increase in earth-to-orbit delivery costs.

There is another significant savings in that the primary cost factor, the earth-to-orbit delivery system operation, can become more highly standardized and regularly scheduled.

Figure 4 is a logic tree indicating a number of the major design trades to be considered in future upper stage studies. The highlighted boxes reflect areas in which the NASA has initiated technology efforts to provide the basis for future design. As the chart infers, all of our studies indicate that reusable, space-based stages are more economic than expendable stages. Aerobraking enhances the economics of reuse and; additionally, offers other potential capabilities. The tradeoff between storable and cryogenic stages is still an open question, highly dependent upon specific mission model characteristics. The general pattern of studies indicates that cryogenic stages are attractive when payload masses are very large and the level of activity quite high. At moderate levels of activity, storable propellant stages appear more attractive.

Figure 5 illustrates the incremental velocity required to make a plane change using four different strategies assuming a Space Station at 371.1 km (200.6 n. mi.). A two-impulse maneuver strategy is unacceptably costly. A three-impulse maneuver, with the plane change introduced at an apogee of 278 000 km (150 000 n. mi.), substantially reduces the velocity requirements in exchange for an increase in mission duration of several days. Aerobraking (L/D 0.3) and aeromaneuver (L/D 0.3) are superior to all propulsive strategies because the heat shield (assuming that the shield can be kept to 15 percent of total inert mass) has an effective, specific impulse several times that of an engine. Aeromaneuver has no significant advantage over aerobraking because the plane change is effected at such low inertial velocities that lateral lifting maneuvers are used only to control flight path precisely enough to dissipate the required kinetic energy and aeromaneuver shapes have a higher mass fraction cost than aerobraking configurations.

Figure 6 illustrates the velocity required to deliver payloads from a Space Station to a number of the more commonly used orbits. It is interesting to note that if the maneuver is made at the optimum time, that is when the orbit plane of the Space Station, and the desired destination orbit intersect, all the orbits of interest can be reached for a velocity increment of 4267 mps (14 000 fps) or less which is the velocity budget for geosynchronous delivery from a 28.5° inclination with an expendable system. The velocity budget for a reusable aerobraking geosynchronous delivery stage is 6400 mps (21 000 fps) which also provides access to sunsynchronous orbits if the stage is expended.

Control of the space environment is an additional attraction of reusable stages. There are over 5000 objects tracked and catalogued in low earth orbit and 200 at geosynchronous orbit. Objects in low earth orbit have high relative velocities and hypervelocity impacts can generate large

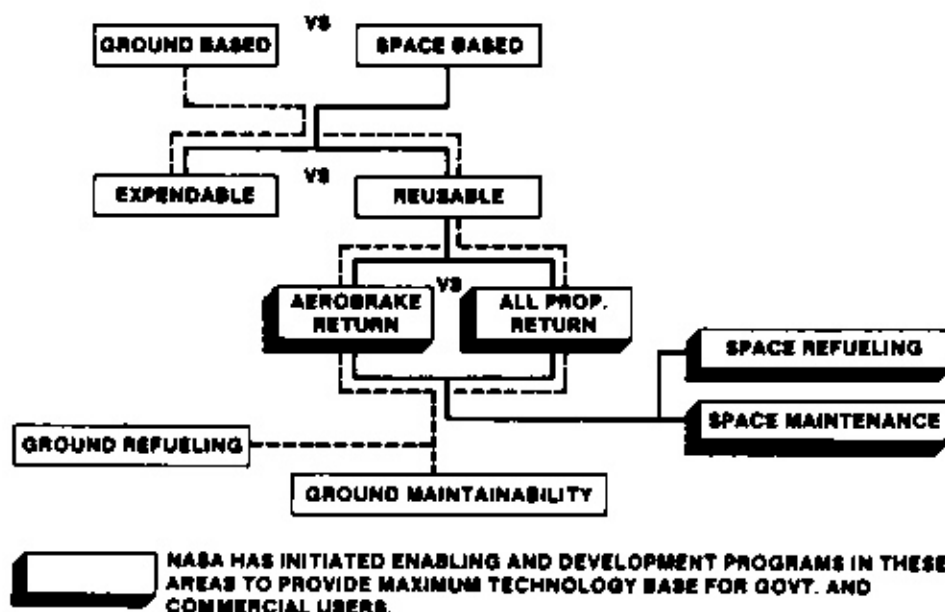


Fig. 4. This logic tree reflects the major parametric trade issues for upper stages. The highlighted cells indicate areas in which NASA has technology and study efforts in progress.

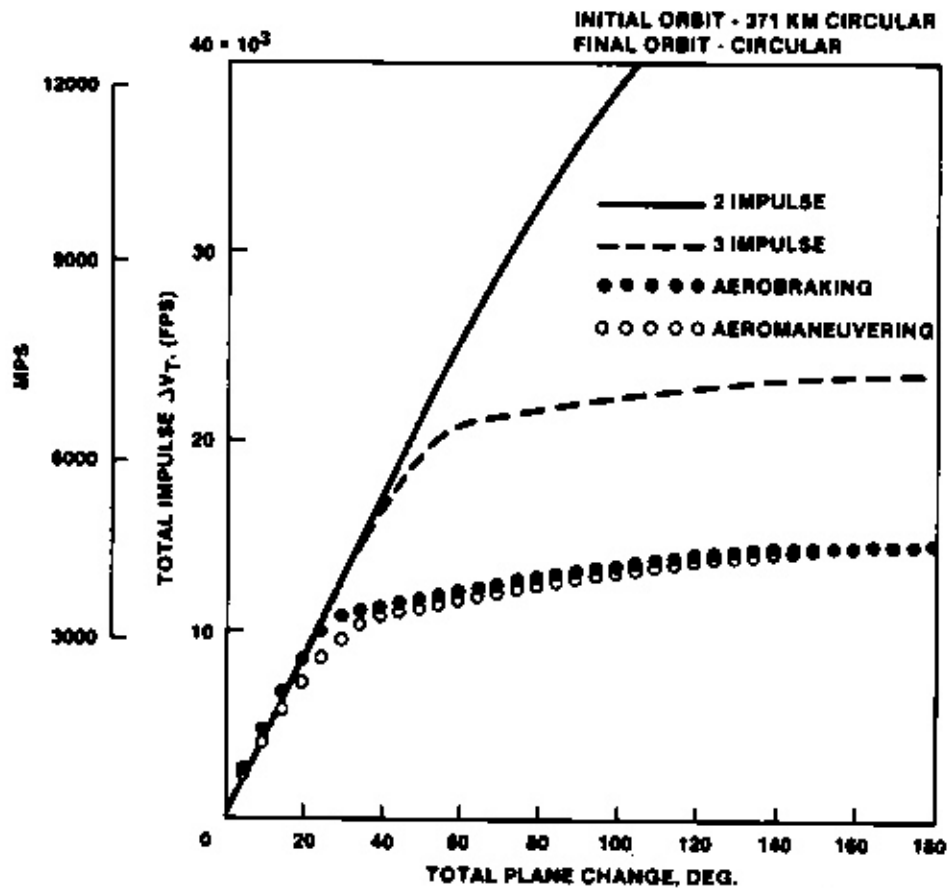


Fig. 5 The total velocity increment to change plane is a function of the maneuver strategy and system configuration. Three impulse maneuvers require longer mission duration but reduce the velocity increment required. Aerobrake and aeromaneuver systems reduce propulsive velocity requirements at the cost of an increase in inert mass.

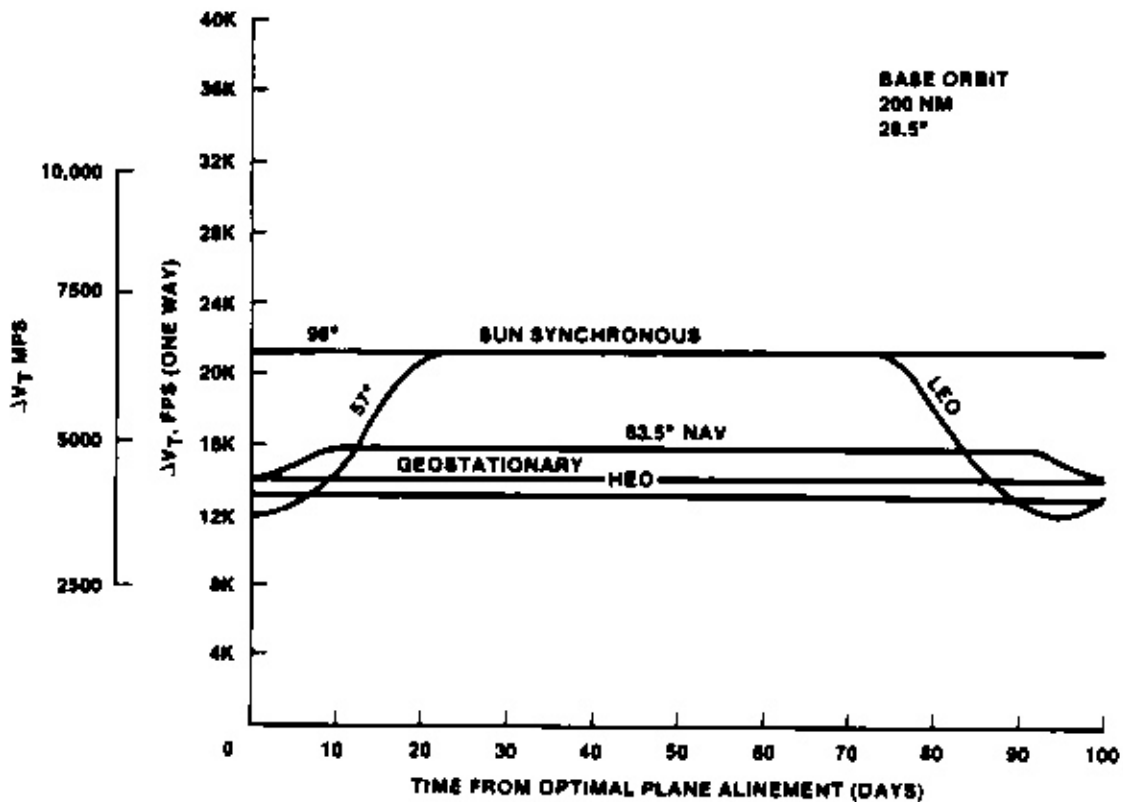


Fig. 6. The velocity increment required to change plane and altitude is a function of the alignment of the original and destination orbits and the maneuver strategy used to effect the transfer.

numbers of fragments which increase the probability of future collision events. Objects in geosynchronous orbit have low relative velocities and while collisions may damage particular spacecraft, they do not modify the environment as do low earth orbit collision events.

Some control of spent stages will eventually be necessary. At geosynchronous orbit the relative velocity of one body to another is quite low but the velocity of transfer stages is high, 1.8 km/sec (6000 fps). Fortunately, the time the stage spends in passing through the geosynchronous belt is very short.

Since all transfer orbit stages repeatedly cycle through both low earth orbit and geosynchronous orbit, it could well be in the interest of all parties to begin to control the future orbital lifetimes of perigee or transfer orbit stages. The objective is to preclude a stage being struck in low earth orbit and becoming a mass of fragments that then passes regularly through geosynchronous orbit. There are only three strategies for such control at present

- a. Control the stage such that at apogee a maneuver of 15.5 mps (50 fps) is executed to ensure a perigee below 100 km (54 n. mi.) and; therefore, entry to the earth's atmosphere within 100 days. The incremental propellant mass, approximately 4 kg (10 lb) is minor, the issue is one of system complexity and, in particular, timing and mission control, and the maneuvers required to assure that the stage will not recontact its payload.
- b. Establish a basic trajectory with a perigee at 100 km (54 n. mi.) or below by doing the perigee burn with thrust oriented below the optimum vector which would reduce the delivery capability of present stages by 18 percent or more.
- c. Constrain the time of launch of geosynchronous transfer systems to an interval such that natural forces, e.g., Sun, Moon, J2 Earth properties and atmospheric drag act to lower the transfer stage orbit perigee. The effect of such an operational constraint is to require that the time of deployment be constrained so that the apogee direction is oriented to the vernal equinox. Such a constraint can be translated into day of launch and time-of-day limitations on transfer orbit stage operations.

These are significant design and operations constraints for communications and weather observing systems, the dominant users of geosynchronous orbit, but such measures may be necessary to reduce the risk of damage to spacecraft in stationary orbit. How soon such precautions should be introduced has not been assessed, but it is obviously much easier to do with storable stages and is accomplished inherently with reusable stages.

#### Technology Initiatives

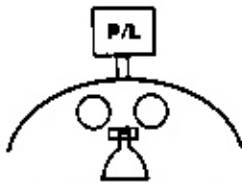
Propellant transfer on orbit is not difficult with storable propellants. A demonstration system was flown on STS mission 41-G in October 1984. Design work has been initiated on a valve for automatic mating and demating on orbit and to define a system for carrying large amount of propellant in the Orbiter cargo bay. Figure 8 illustrates the system and its potential uses. Spacecraft such as the Gamma Ray Observatory are being designed for on orbit fueling. Many other spacecraft have operational lifetime limited by maneuver propellant depletion and could have extended productive operation if refueled.

Cryogenics are more difficult to manage and the behavior of such fluids in the weightless environment is not well understood. A facility for testing the management of cryogenic fluids in space is being developed by the Lewis Research Center and will be flown on the Shuttle to develop and demonstrate suitable techniques. It is desirable to be able to transfer cryogenics on orbit both to make space-based cryogen fuel stages and to be able to replenish cryogen coolers for various instruments.

Because aerobraking offers so many attractions, it has been studied extensively for 20 years. Figure 7 illustrates many of the system configuration options studied. The variables in these studies have been the number of stages in the system and application of the aerobrake to recover only the stage or both the stage and the payload. A number of aerobrake configurations have also been studied, among them an inflatable system called the bollute, a number of deployable structure configurations, and recently a fixed structure raked cone illustrated in the lower right of the fig. This has proved an attractive configuration because it minimizes structural mass and positions the payload close to the shielding of the aerobrake. As noted, it requires extensive thrust vector control to accommodate the large center of mass excursions which occur between the initial fully fueled and end of mission conditions.

A flight experiment to test an aerobraking configuration is under study. The objectives of the experiment are to be qualification of the aerobrake design and materials, verification of the control laws and determination of the heating rates. The nonequilibrium gas dynamic processes that take place at the velocities of interest are not well understood, and there is disagreement by factors of 2 to 10 as to the amount of radiant heat load imposed on the vehicle.

● **SEPARATE AEROBRAKE**



- STAGE AEROBRAKE RETURN
- PAYLOAD DELIVERED AND LEFT @ GEO

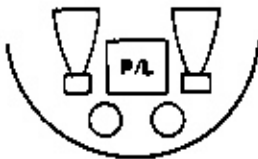


- 1ST STAGE EXPENDABLE
- 2ND STAGE AND PAYLOAD AEROBRAKE RETURN



- BOTH STAGES AEROBRAKE RETURN

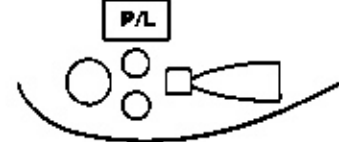
● **SINGLE INTEGRATED AEROBRAKE (P/L AND STAGE AEROBRAKE RETURN)**



- AFT FIRING ENGINES
  - POTENTIAL PAYLOAD SIZE LIMIT
  - PLUME INTERFERENCE



- FORWARD FIRING ENGINES
  - AEROBRAKING COMPLEXITY
  - ENGINE EXHAUST DOORS



- SIDE FIRING ENGINES
  - PROPULSION SYSTEM IMPACT
  - LARGE C.G. AND GIMBAL EXCURSIONS

Fig. 7. Aerocapture techniques are attractive to reusable systems if the mass of the aeroshield can be limited to a reasonable proportion of the stage inert weight. Configuration options also need to consider constraints on payload dimensions for both delivery and retrieval.

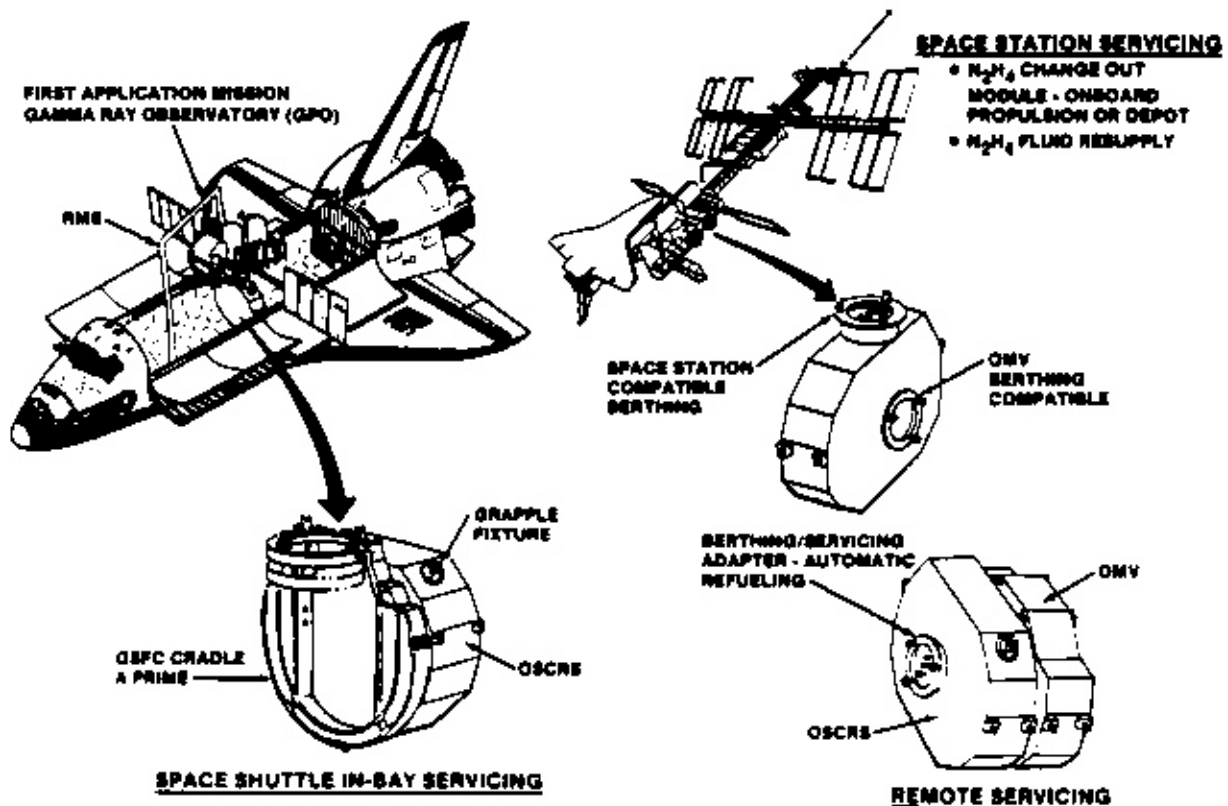


Fig. 8 The equipment needed to do an orbit propellant transfer to many classes of spacecraft exists or is in development.

## SUMMARY

As space operations mature and become more frequent, it becomes economically attractive to make many of the elements reusable. When it is necessary or desirable to retrieve things in space, reusability becomes an inherent attribute. It seems reasonable to expect that upper stages will become reusable in the context of the Space Station and its reusable facilities.

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