2. DEVELOPMENT COSTS (Vehicle Systems and Engines)

2.1 Development Cost Criteria

2.11 Launch Vehicle Programs' Cost Survey

Development Cost (DDT&E) estimation is one of the most difficult costing areas since a lot of subjective influence can be found in the definition of a development program. There are quite a number of technical criteria : the degree of cost engineering which has been applied in the technical project definition (or not), the technology status and the realism of dry mass estimates or margin strategy, as well as in the design verification area the scope of testing and the number of test units considered to be required (or not). In addition, the development costs are influenced by administrative, contractual and business aspects (see chapter 2.5).

The major criteria which impact the development cost of a launch vehicle are

- vehicle launch mass and size,
- number and type of stages,
- technology readiness/ scope of existing subsystems/components,
- type and number of engines,
- reliability and safety requirements,
- verification and test strategy,
- number of flight units and flight tests,
- company and team experience,
- program organization and management procedures,
- program budget planning and schedule/ ..delays,
- technical changes required or ordered by the customer,
- contract conditions, etc.

The same concept of launch vehicle with the same payload can come up with a wide range of development costs: by example between 40 000 and 70 000 MYr (or 8 to 14 Billion USD). The result simply depends on the better or worse realization of the above-mentioned criteria.

As a general survey the development program costs of a number of major launch vehicles and projects have been assembled in FIG. 2-01, indicating the cost trend vs. vehicle size in terms of launch mass (GLOW). The difference between expendable launch vehicles compared to reusable winged vehicles is evident: it is factor 4 for small vehicles and factor 2 for large vehicles. Ballistic reusable vehicles (BETA II as reference) are expected to range between the two groups.

The MYr-values of non-US projects have been adapted to the US productivity standard (cf. TABLE 1-II) and represent government-funded programs.

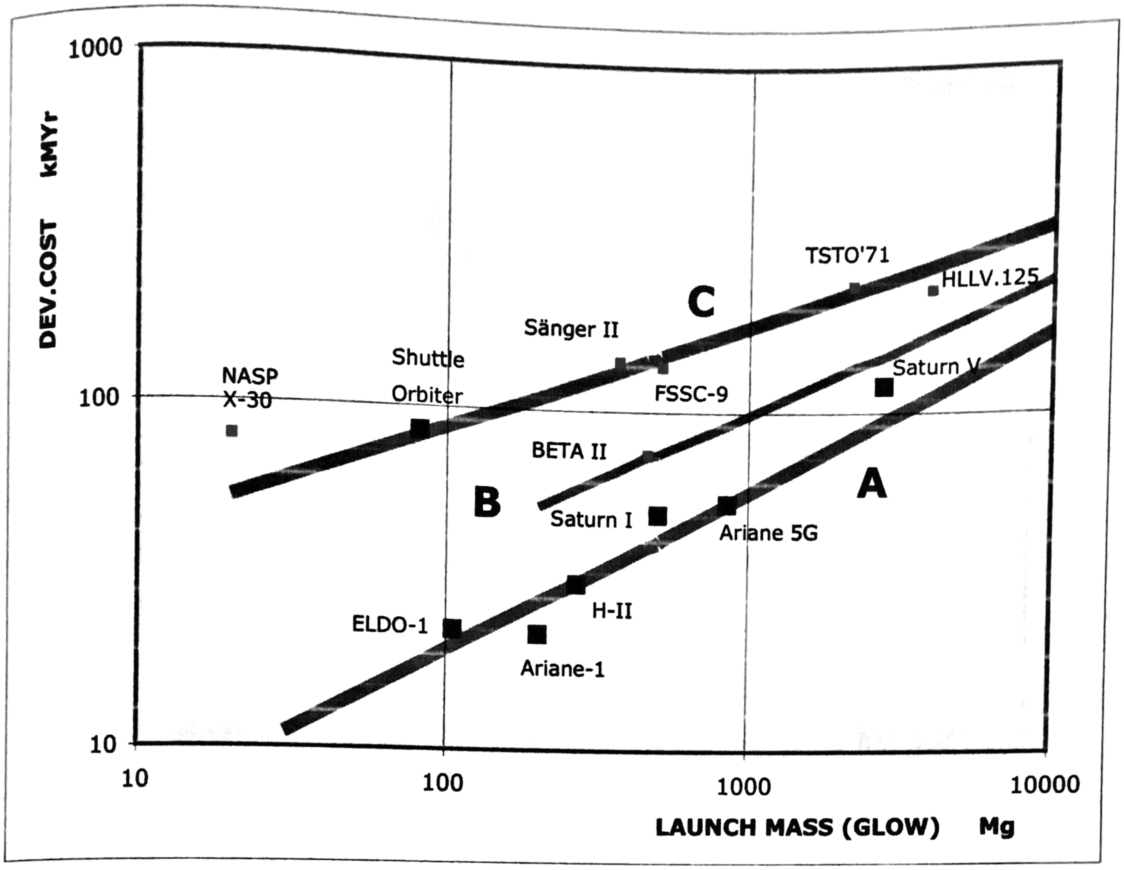


FIG. 2-01 : **Survey of Launch System Development Costs vs. GLOW**
A = Expendable Launch Vehicles, **B** = Ballistic RLVs, **C** = Winged RLVs

The range of development costs for complete launch systems as shown in FIG. 2-01 means that expendable vehicles (ELVs) of 100 to 1000 Mg GLOW did cost in the past between 5 and 217 Billion US Dollars - in 2006 value - depending on the launch mass - with the application of traditional development and contracting conditions.

However, it is shown in chapter 2.61 that even for government contracts the application of more modern experience-based conditions and cost engineering can reduce the development costs to 70 or even 50 % of the traditional costs. This fact may be very important for the development of future RLVs which are two to four times more expensive to develop than ELVs.

Vehicle size (GLOW) is NOT a major cost driver: it is evident from FIG. 2-01 that increasing the vehicle size, resp. its payload mass (margin) by 20 %, results in only some 7 % increase of development cost in case of ELVs, and only of some 4 % in case of the more expensive RLVs. It is cost-effective, therefore, to oversize the vehicle somewhat in order to have sufficient payload margin. Reusability becomes more important and cost-effective with growing vehicle size (performance), as illustrated in FIG. 5-17.

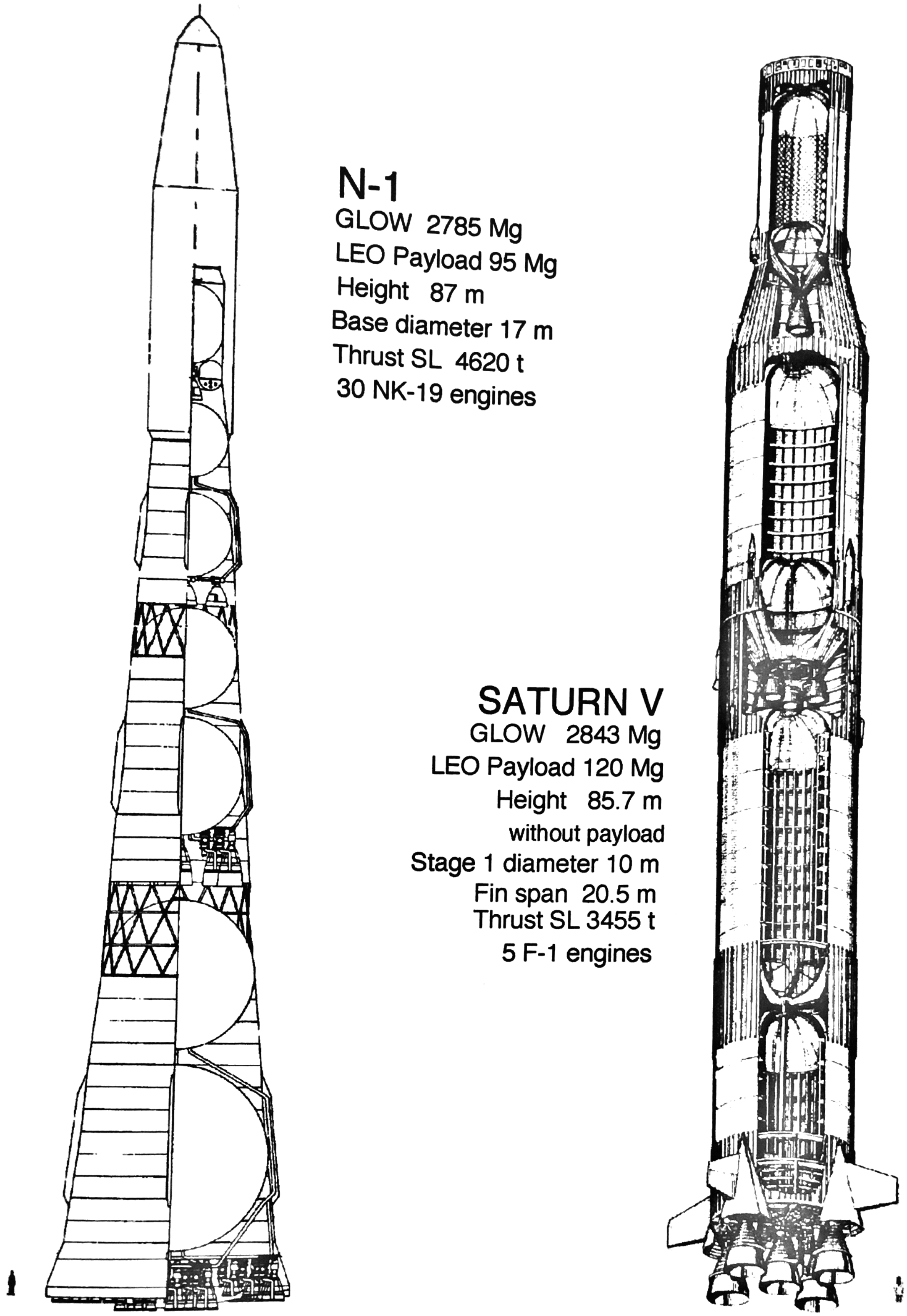


FIG. 2-02 : **The largest Launch Vehicles built: N-1 and SATURN V (2700 tons)**

The largest launch vehicles built up to now are the US SATURN V (1962-73) and the Russian N-1 Vehicle (1964 - 74), both developed for a manned lunar landing mission. FIG. 2-02 shows these vehicles and the key data. SATURN V employed two cryogenic stages and used 5 large engines for the first stage (680 t - thrust level). In Russia conventional propellants were used (hydrogen technology not yet developed), as well as an assembly of 30 engines in the 150 ton-class in the first stage. The lower performance required the use of 4 stages for reaching escape velocity while SATURN V only needed 3 stages. The launch mass (GLOW) of both vehicles was the same with 2800 tons, but the LEO payload of SATURN V with 120 Mg was higher than the N-1 payload of 95 Mg.

The number of stages has a major impact not only on the development cost but also on production, integration and operations costs. Solid-propellant launch vehicles require 3 - 4 stages, storable propellant vehicles require 2 - 3 stages and cryogenic propellant vehicles 1 - 2 stages. Cost engineering principles recommend to minimize the number of stages as well as of different engines. More stages do not only mean more systems to be developed and built but also the additional interface management (assembly and stage separation) reducing the vehicle reliability. Therefore, from the cost engineering viewpoint a single-stage vehicle to LEO is the most cost-efficient and reliable solution. This is feasible with present technology, for ELVs even with LOX/ Kerosene propellants (ATLAS V-I) and for RLVs with LOX/Hydrogen.

The impact of the number of rocket engines and their technology level on development cost is discussed in chapter 6.1.

2.12 The Different Comprehensions of „Development Cost“

The same term „Development Cost“ (DDT&E) has been used in the past with very different comprehensions : At least five different types of „development cost“ can be identified, resulting in very different total costs :

(1) *Effective Cost-to-Completion (CTC)*
Total cost after completion of the program, including inflation (respectively increase of annual hour-rate or ManYear cost)

(2) *Most Probable or Realistic Development Cost*
including margin for unforseeable technical problems and delays

(3) *Ideal or Theoretical Development Cost*
assuming that everything goes as planned, without technical or schedule problems (standard industrial proposal basis)

(4) *Minimum Credible Development Cost*
Unrealistic cost estimate under competitive situation in order to win the contract (some cost items neglected)

(5) *Unrealistic Development Cost*
Cost figures based on „believing“ , without cost studies and lack of experience, in order to sell a concept.

The development cost derived with *TRANSCOST*-algorithms are of TYPE (2) since the CERs are based on actual system development cost. They can even be upgraded to approximate TYPE (1) cost by using the predicted Man-Year cost values over the anticipated development period, instead of just using the MYr-value of the year of estimate.

TYPE (2) Costs are about 15 to 20 % higher (in average)[1] than the „Ideal Costs“ TYPE (3), which are normally derived by a „bottom-up“ cost estimate through addition of subsystem cost or by use of a subsystem-based cost model. Also normally the cost level of the year of estimate or proposal preparation (i.e. excluding the automatic cost growth during development by hour rate increases/ inflationary effects). TYPE 4, the minimum credible development costs can be as low as 75 to 85 % compared to Type 3 costs because some cost items have been excluded (and often carefully hidden in the proposal contract conditions), while TYPE 5 costs sometimes are just 30 to 20 % of a realistic cost estimate.

All these costs are *industrial or contractor's cost*, excluding program management, control and support costs (i.e .astronauts staff) at the customer agency. The so-called „wrap-cost factor“ can be between 22 and 27 % in case of conventional or „business-as-usual“- programs (as it has been for the Shuttle and Space Station programs), but could als be lower in case of intentional limitation of customer staff.

2.2 The TRANSCOST Development Cost Submodel

2.21 Model Definitions

The development effort as referred to in this submodel is defined as the traditional way of governmental agencies' (NASA, ESA, etc.) methods and procedures in the past. The effect of other (commercial / industrial) development strategies are discussed in chapter 2.5.

The scope of „development“ or non-recurring costs includes all activities from detailed design (Phase C) to hardware implementation and verification (Phase D) including all test models built, including protype or protoflight units. Also ground installations,, required tooling and test facilities are included in the total development cost, assuming that only supplementary equipment ist required. In case of completely new

[1] Statement of ESA's Director General, A. Rodota (Space News, 31.1.2000): In the past, programs were routinely 15% over their budgets“

facilities additional cost have to be taken into account. Also the desired or requested number of flight test vehicles and flight operations must be added with assistance of the Recurring Cost (VRC) Submodel and the Ground and Flight Operations Cost (GOC) Submodel.

The backbone or basis of each cost model are the CERs, the Cost Estimation Relationsships. Their definition has already been mentioned in chapter 1.24 and it is remembered that each type of vehicle system or engine required its own specific CER and its technically derived Quality Factor f_2. Although this is a system model the engines have to be considered separately since there can be different number of engines on a vehicle, existing ones or newly to be developed.

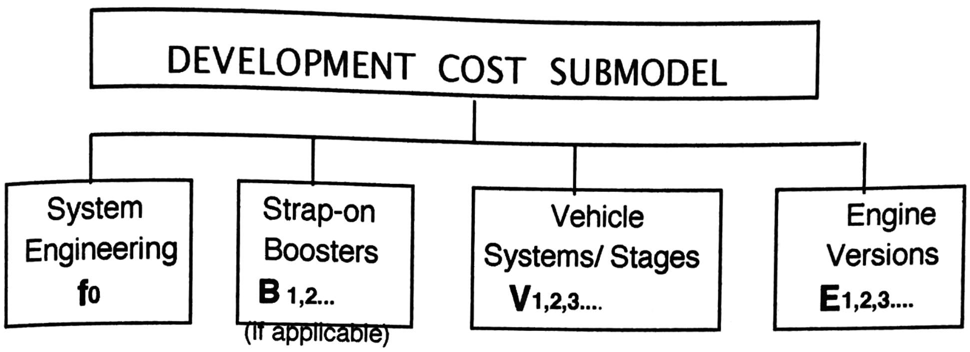


FIG. 2-03: **Development Submodel Cost Elements**

The accuracy of the elements' development cost estimates is relatively high since the CERs are dedicated system-level algorithms. In addition, specific and optimized technical quality factors are applied for the most elements. The CERs are derived from the actual costs of realized projects which are shown in each case. The data spread of the reference projects is in the range of 15 to 25 % which is considered the best possible and achievable only by the specific Technical Quality Factor.

The NRC Submodel consists of three elements only : the vehicle CERs, the engine CERs and booster CERs (if applicable). A system engineering / integration factor f_0 takes care of the number of vehicle stages involved; it has a value of 1.04^N, with N = number of stages.

The total development cost of a complete launch vehicle are the total added elements' costs H, multiplied with the system engineering / integration factor f_0 and the programmatic cost impact factors f_6, f_7, f_8:

$$\boxed{C_D = f_0\,(\Sigma H_B + \Sigma H_V + \Sigma H_E)\, f_6 \cdot f_7 \cdot f_8}$$

This looks relatively simple - but in principle only . In reality the problem of cost estimation is much more complex since it is not only a technical or mathematicall subject. Human factors also play a major role. Those are taken into account first in the elements development cost (chapter 2.13) as well in the total program cost estimate.

2.22 From CER's to Complete Program Cost Estimates

For the development cost estimation of a future project several criteria have to be taken into account in addition to the basic CER data. These criteria are

- the development standard of the new project, compared to previous ones of the same type, and
- the experience of the team to be entrusted with the new development related to its previous activities.

In addition, there are a number of programmatic cost impacts such as

- the quality and scope of project preparation such as detailed definition studies and technology pre-developments,
- the type of program organization planned for the project implementation, and
- the planned or defined time schedule and potential budget restrictions.

All of these have to be considered in order to get a realistic range of expected maximum and minimum development cost which easily can differ by a factor 2 or more.

Otherwise, past experience also has shown that not only high development cost and / or cost overruns can be avoided by careful cost-concious project planning but also essential *c o s t r e d u c t i o n s* are feasible compared to the traditional development methods sometimes called BAU = „Business As Usual". These different development strategies are discussed in chapter 2.6.

In order to illustrate the procedure and to show the different steps from the basic CER element cost data to realistic element cost, to vehicle cost, and finally to the development program cost, the following definitions are used:

Step 1: From

BASIC Dev. **CER**	$C = a \cdot M^x$

Step 2: to

ELEMENTS' Dev. COST (Vehicles, Engines, Boosters)	$H = C \cdot f_1\ f_2\ f_3\ f_8$

Step 3: to

LAUNCH VEHICLE Dev. COST
C_D

$C_D = f_0 \sum H$

Step 4: to

TOTAL SYSTEM DEVELOPMENT COST
C_{tot}

$C_{tot} = f_0 \sum H \; f_6 \, f_7 \, f_8$

The launch vehicle system development cost C_{tot} are the total industrial or technical development cost.

The complete PROGRAM COST would include additionally the

- -- governmental agency's program management and supervisory staff's cost,
- -- reserve funding for potential changes or additional development tasks,
- -- technical support cost, and
- -- ground support and launch site investments.

The additional cost compared to C_D - also called the "wrap factor" - has been historically between 22 to 27 %. The trend for future programs is towards the lower value of 22 %.

Beside the industrial cost and the program support cost there are the cost of inflation, resp. the cost of the increasing hour rates during the development period, which have to be taken into account for the annual budget planning.

2.23 Development Cost Factors f_1, f_2, f_3

In order to establish the CERs with satisfactory accuracy, as well as for development cost estimates it is required to introduce a number of factors which can be major cost drivers. In the *TRANSCOST*- Model there are three technical factors which will be defined in this chapter (the three programmatic factors are dealt with in chapter 2.5).

The DEVELOPMENT STANDARD FACTOR f_1 :

The development effort is influenced by the relative status of the project in comparison to the state of the art, or the relation to other similar projects.

This means it could be a completely new, first-generation system involving new techniques and new technologies (examples: the Manned Lunar Lander of the Apollo-Program, or the Space Shuttle Orbiter). Or it can be a new development, but based on existing state-of-the-art (i.e. the ARIANE H.155 Stage, or new Solid-

Propellant Boosters). On the other side, a project also can be composed of existing subsystems, or is just a modification of an already existing system (example: DELTA First Stage, derived from the THOR missile).

It cannot be avoided in this case that some subjective judgement is required ; as a guideline the following numerical values could apply:

f_1	First generation system, new concept approach, involving new techniques and new technologies	f_1 = 1.3 to 1.4
	New design with some new technical and/ or operational features	f_1 = 1.1 to 1.2
	Standard projects, state-of -the-art (similar systems are already in operation)	f_1 = 0.9 to 1.1
	Design modification of existing systems,	f_1 = 0.7 to 0.9
	Minor variation of existing projects	f_1 = 0.4 to 0.6

The TECHNICAL QUALITY FACTOR f_2

In contrast to factors f_1 and f_3 the Technical Quality Factor f_2 is a factor derived from the technical characteristics of the project. This factor is different and characteristic for each technical system and it may be the most important cost driver. The factor is based either on the relative net mass fraction, the performance or another important cost impact factor, such as the number of qualification firings in case of liquid rocket engines.

> f_2 = Specific for each system (or element type), defined by an inherent technical criterion

The different specific Quality Factors are discussed and defined in the following chapter together with the CERs for each technical element.

The TEAM EXPERIENCE FACTOR f_3

The relevant experience of the team entrusted (or to be entrusted) with the development of a new project is another major cost driver. Clearly an unexperienced new team will need a higher development effort than a team which has dealt, by example, with a very similar task before.

Some experience-based values for the f_3-factor are as follows:

f_3	New team, no relevant direct company experience	f_3 =	1.3 to 1.4
	Partially new project activities for the team	f_3 =	1.1 to 1.2
	Company / industry team with some related experience	f_3 =	1.0
	Team has performed development of similar projects	f_3 =	0.8 to 0.9
	Team has superiour experience with this type of project	f_3 =	0.7 to 0.8

The „team" can be from one company only, or can consist of a group of different companies, depending on the development task. For a cost estimation the most probable case should be assumed in case the potential contractor(s) are still unknown.

Examples for the application of these factors are given in chapter 2.8 which compares TRANSCOST-derived NRC with actual project development cost.

It is emphasized here that the development cost derived by the TRANSCOST-CERs are 15 to 20 % higher than the „Ideal Cost" which result from a bottom-up subsystems estimate since the TRANSCOST CERs are based on actual cost, including costs of unforeseen technical problems and delays. Therefore, the TRANSCOST development cost represent the „most probable realistic cost"

2.3 Propulsion / Engine Development CERs

All CERs in the following chapters have been established with the procedure described in chapter 1.24. The reference projects are indicated in the graphical display of the basic CER which represents the best degression fit of historical cost data.

2.31 Solid-Propellant Rocket Motors

This group of propulsion systems comprises motors with solid propellants which have segmented or single motor cases. Motor cases can be built of steel, Kevlar-fibre Composite or of Carbon-fibre Composites. Applications are for satellite-integrated apogee propulsion, propulsion modules (attached to satellites) or as simple strap-on -boosters for thrust augmentation of launch vehicles. In this case only small boost motors are considered with fixed nozzles.

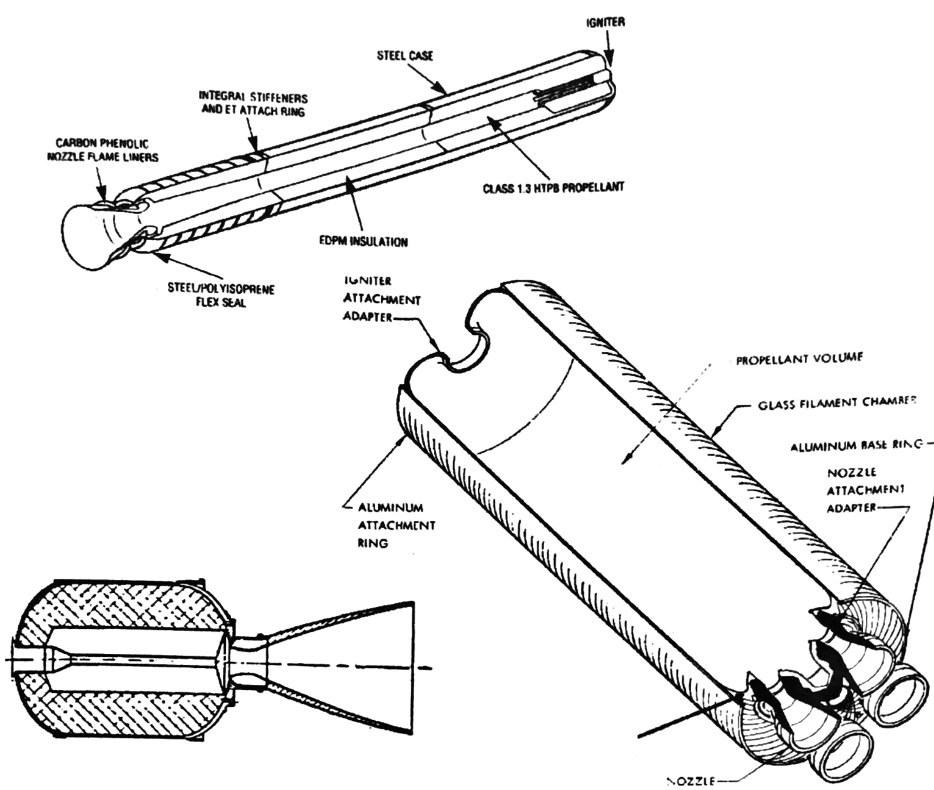


FIG. 2-04: **Design Examples of Single-case and Segmented Motors**

Larger Boosters with thrust vectoring system (either a movable nozzle or a fluid injection system) and Reusable Boosters with solid-propellant motors are covered in chapter 2.41.

FIG. 2-04 shows examples of typical solid propellant rocket motors; in FIG. 2-05 the relationship between propellant mass and motor net mass is depicted. Above some 100 Mg propellant mass segmented units are used. The difference between the mass of the motor proper and an equipped recoverable booster unit is also indicated.

There are ten reference projects with sufficient technical and cost data available, from small satellite apogee motors to large multi-segment launch vehicle boost motors

The resulting basic CER for solid-propellant rocket motors is depicted in FIG. 2-06. Due to the extraordinary fact that a good correlation exists already for the 10 reference projects without application of a Technical Quality Factor, such a factor has not been introduced here. The CASTOR-120 project has lower development cost as a commercial, company-funded project (with f_1 and f_3 taken into account with 0.7

each).

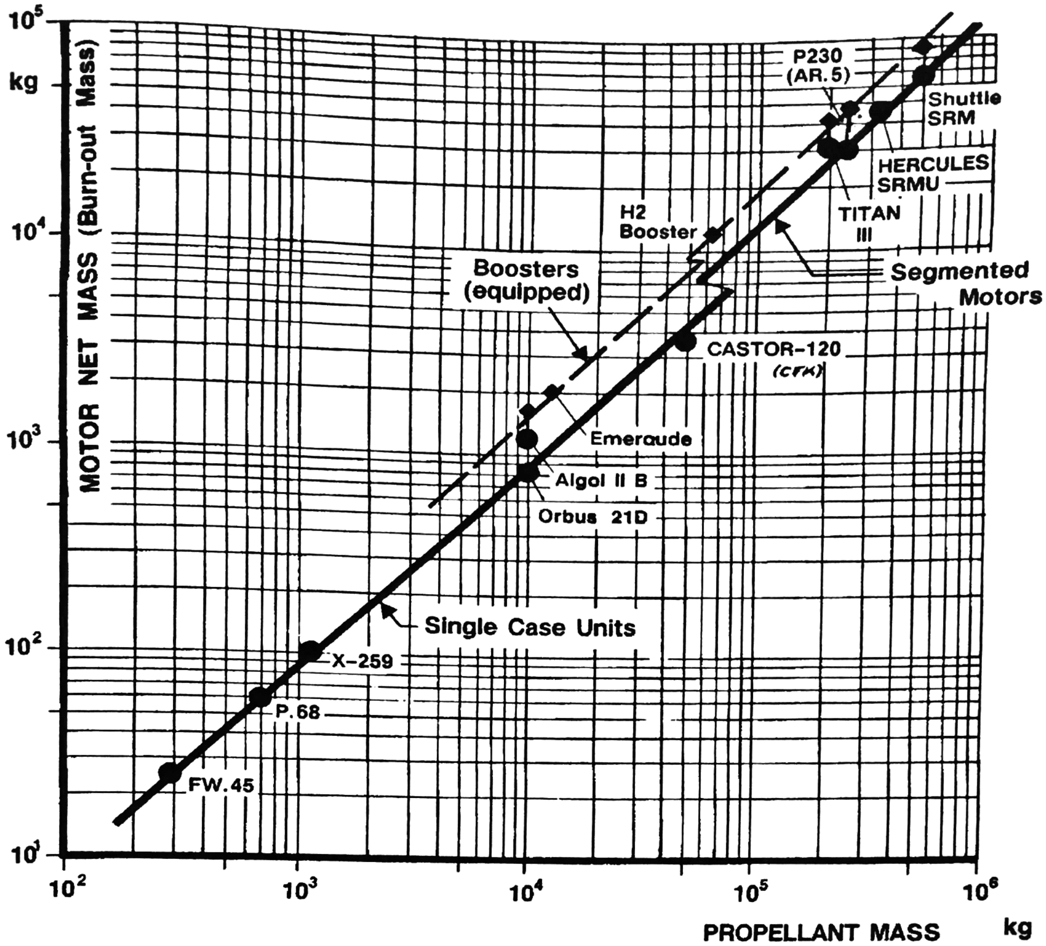


FIG. 2-05: **Net Mass of Solid-Propellant Motors vs.Propellant Mass**

The resulting basic CER for Solid-Propellant Rocket Motors is accordingly :

$$\boxed{\mathbf{H}_{ES} = 16{,}3 \ M^{0.54} \ f_1 \ f_3} \qquad \text{MYr}$$

The number of motor test firings for qualification is only high for small motors. It decreases, however, with motor size to only 3 to 6 tests for large motors - in contrast to liquid propellant engines. The nominal reliability is determined in this case by probabilistic methods, not by the number of tests. In addition, the cost ratio between development effort and unit fabrication cost is about 30, therefore, the number of test units which are built is not a cost driver for the development effort.

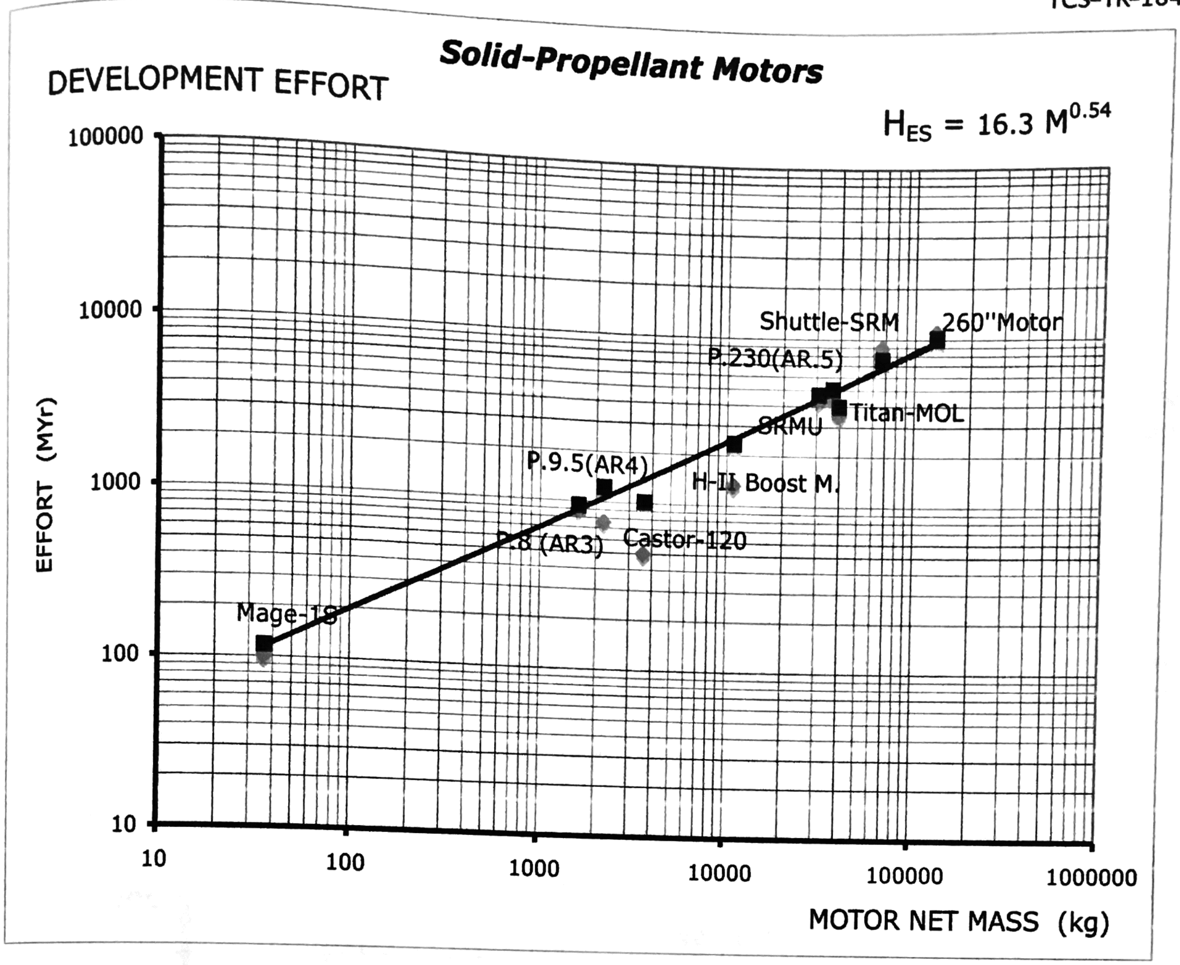


FIG .2-06 : **Solid-Propellant Rocket Motors' CER with Reference Projects**
(Red rhombs are original data, black squares are the reference points after regression process)

2.32 Liquid-propellant Rocket Engines with Turbopumps

This specific group of engines comprises all types of liquid-propellant rocket engines with turbopumps, both with cryogenic and storable propellants. FIG. 2-07 illustrates typical engine configurations (Pressure-fed engines see chapter 2.33). The relation between engine thrust in vacuum and the engine mass is depicted in FIG. 2-08. This chart helps in cases where only the engine thrust level is known but not the engine mass. There is no evidence for a mass difference between cryogenic propellant and storable propellant engines.

The mass of two new engines is above the median: the Rocketdyne RS-68 engine which has been built intentionally somewhat heavier for reasons of cost reduction and reliability improvement. The higher mass of the new VINCI space engine for the

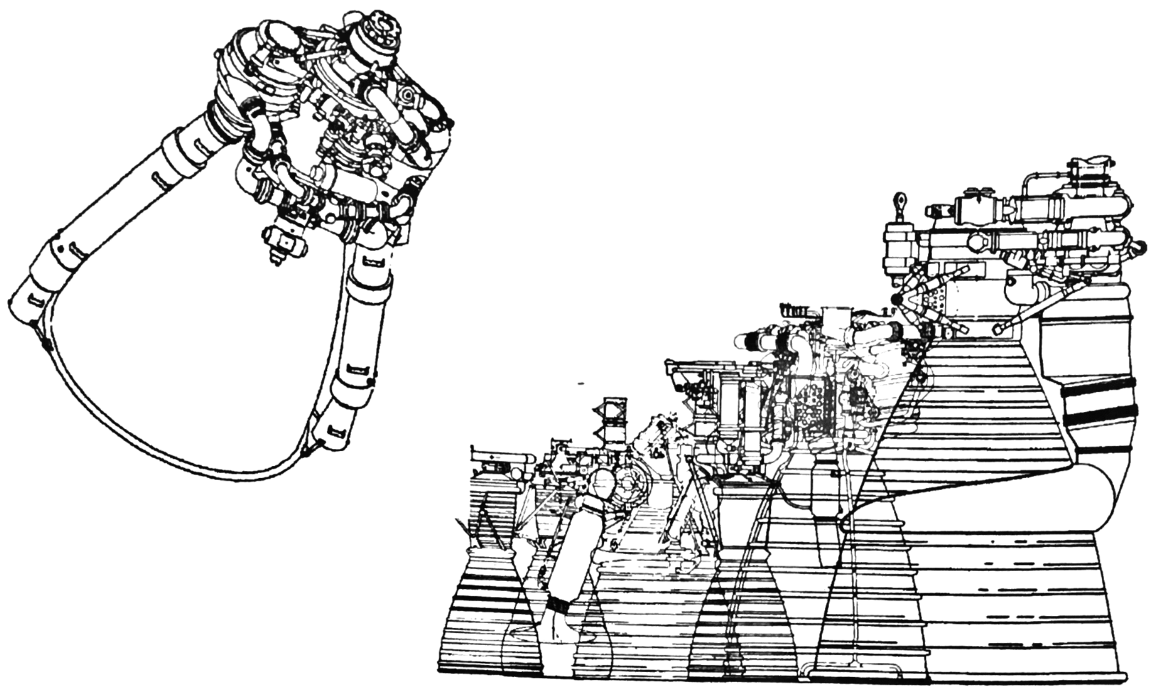


FIG. 2-07: **Typical Pump-fed Rocket Engines**

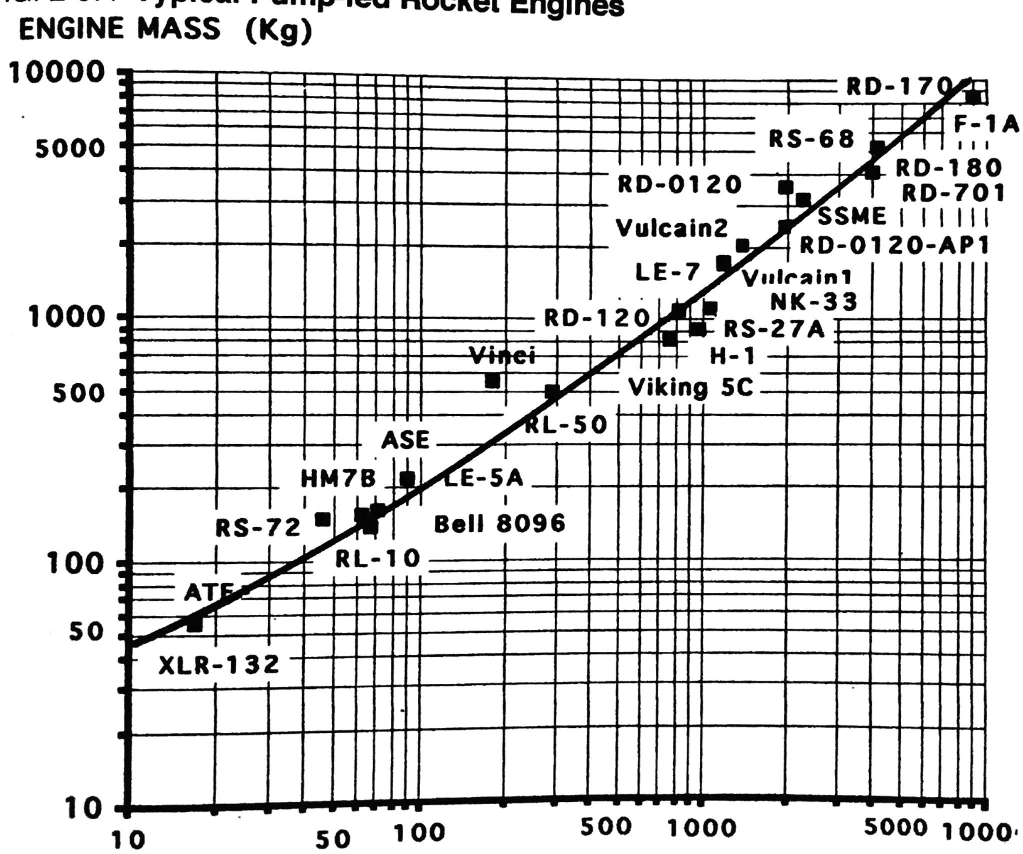


FIG. 2-08: **Rocket Engine Dry Mass vs. Vacuum Thrust Level**

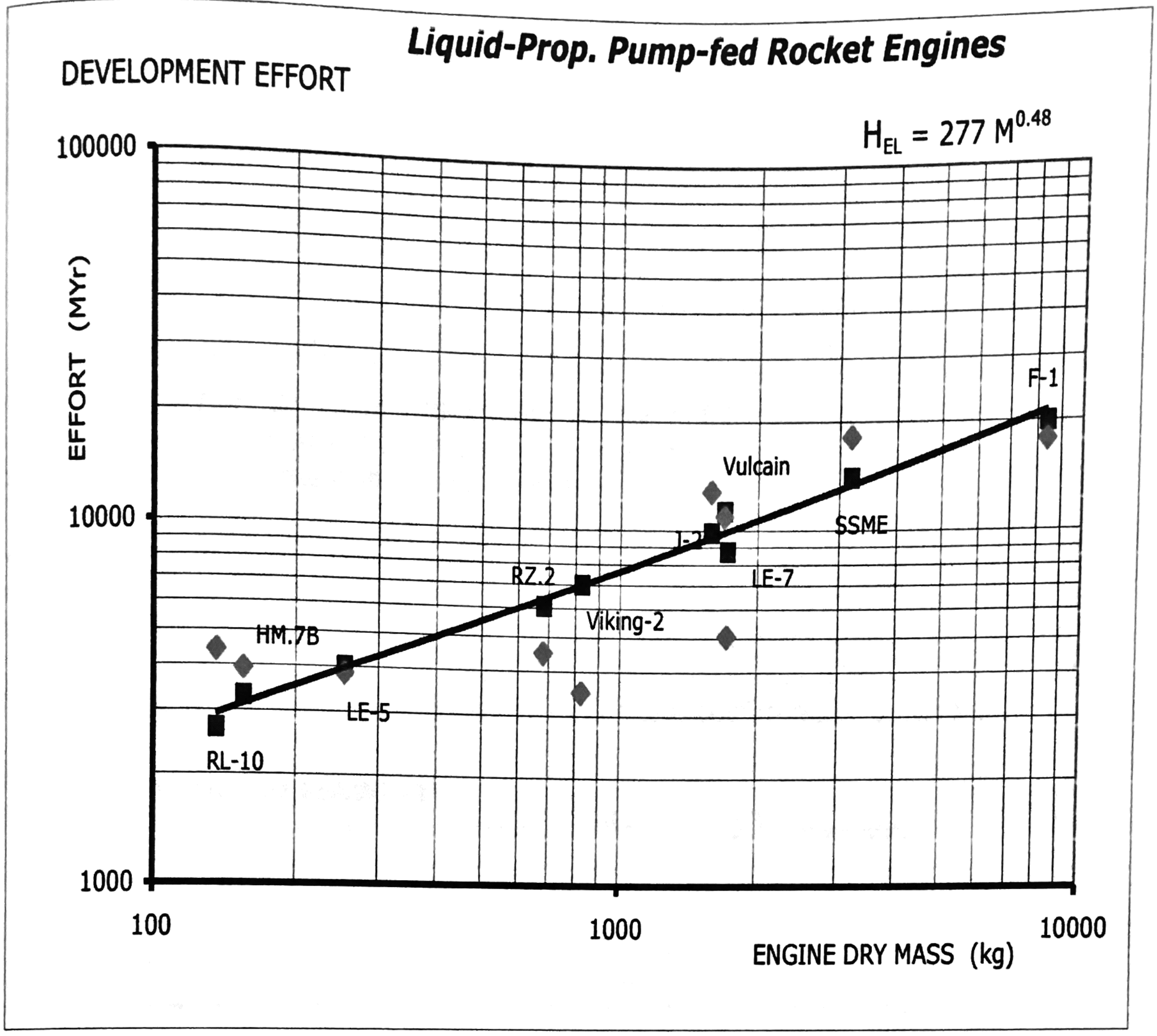


FIG. 2-09 : **Basic CER for Rocket Engines with Reference Projects**
(Red rhombs original data, black squares: reference data with regression factors applied)

cryogenic third stage of ARIANE 5 is due to the large extendible nozzle (area ratio 240 !). The CER-base is presented in FIG. 2-09 with 10 reference projects from the USA, Europe and Japan (compared to only 6 projects in the last *TRANSCOST* - Version). Only a small change in the basic CER resulted. The application of the productivity correction factor (cf. TABLE 1-II in chapter 1.8) for engines developed in Europe and Japan proved to be very useful for the correlation of the data points.

The original MY development effort for each engine is indicated in FIG. 2-09, as well as the CER reference data after the regression process. The resulting CER for the development effort of pump-fed rocket engines both with cryogenic and storable propellants is

$$\boxed{H_{EL} = 277\ M^{0.48}\ f_1\ f_2\ f_3} \quad \text{MYr}$$

Technical Quality Factor f_2 :

Out of the various possibilities it has been verified that *not* the type of propellant or the specific impulse has the major impact on development costs, but the number of test firings performed during the engine development and qualification program, resp. the engine reliability standard required. There is a strong relationship between the number of engine tests and the demonstrated reliability. This was assessed the first time in a paper from Pratt & Whitney (ref.9) and confirmed by the SSME (Space Shuttle Main Engine) Program, as depicted in FIG. 2-10. 730 tests had been made between June 1975 and the time of the first Shuttle launch in April 1981. The engine reliability at that time was stated to be 0.984, and it was increased by additional tests to more than 0.99. Until Nov. 1991 some 1900 engine tests had been made, including the operational launches.

Another example is the HM.7 engine (ARIANE 4 third stage) which in the original version was only test fired some 158 times (1968-72). The result was an engine failure at the 5th flight and two more failures until the 63rd flight, in spite of an increased number of engine qualification tests. The theoretical relationsship according to ref. 9 is shown in FIG. 2-12 which seems to be well confirmed by a number of

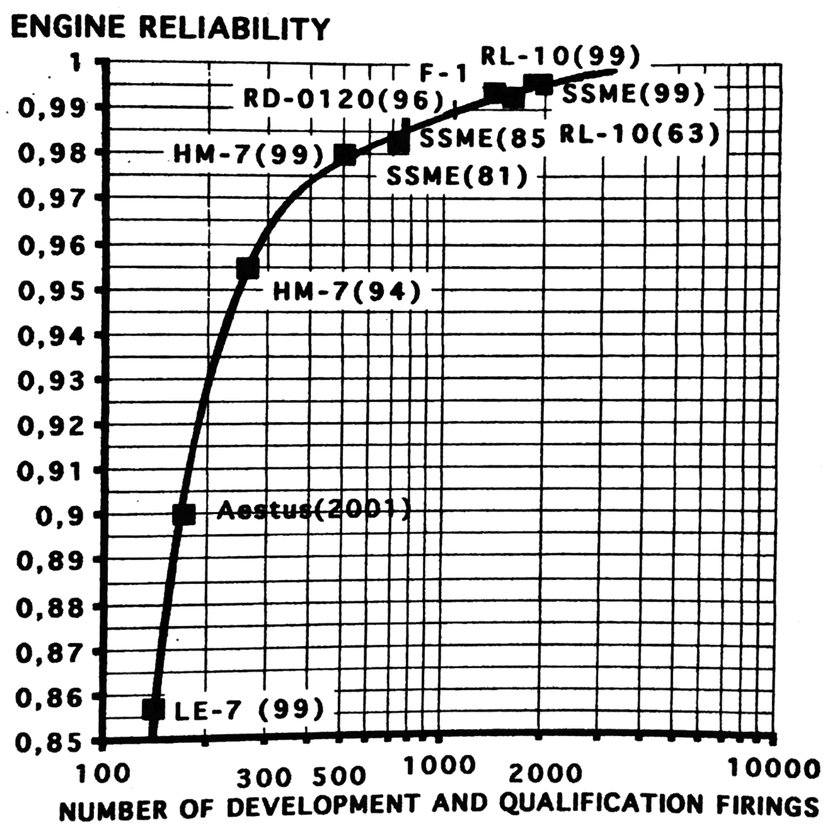


FIG. 2-10: **Liquid Propellant Rocket Engines' Reliability Model**

examples. The continuous improvement of the SSME reliability with the growing number of tests is shown (related to 100 % thrust level). The Japanese LE-7 engine performed only 140 system qualification firings before the first (successful) launch. However, the engine failed at the 7th launch, thus ending up with an effective reliability of 0.857. For the RL-10 engine the number of firing tests during development and qualification was 1600 according to NASA News Release No.62-97 (Sept. 62), including the initial engine models.

In comparison, jet engines are qualified through some 12000 endurance cycles before flight testing (ref.85) thus achieving an operational reliability of 0.9999.

Although it should be possible to reduce the number of test firings by application of modern probabilistic methods the original assessment seems to be well confirmed by practical experience in the past decades. In fact, the number of engine development and qualification test firings before the first flight operation differed considerably as the survey shows:

LE-7 (Japan)	140 tests	Vulcain-1(Europe)	250 tests
SSME (USA)	730 tests	RD-0120 (Russia)	800 tests
RL-10 (USA)	1600 tests	F-1 (USA)	1 437 tests.

For engines with ablative chambers the complexity is lower, and fewer number of tests can be expected. However, the Merlin 1A engine for FALCON-1 had only 130 tests before the first flight mission, and the system failed at the first launch in March 2006.

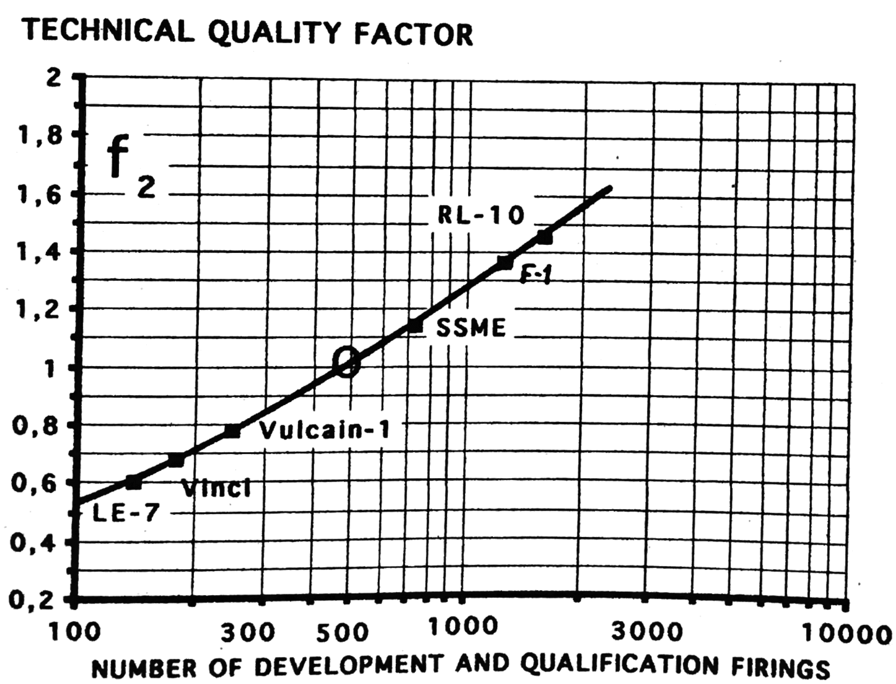


FIG. 2-11: **Technical Quality Factor f_2 for Liquid-Propellant Rocket Engines**
(with selected examples)

With this background of engine reliability vs. number of qualification firings it is possible to create a Technical Quality Factor for the rocket engine development CER, as depicted in FIG. 2-11. The reference point for the Quality Factor f_2 (f_2 = 1.0) has been set at 500 tests which represents an average value.

The resulting relation for the rocket engines' qualification factor is as follows:

$$\boxed{f_2 = 0.026\,(\ln N_Q)^2}$$

The effective operational engine reliability depends, however, not only on the number of qualification tests but also on the operational thrust level used. The values of FIGs. 2-10 and 2-11 refer to a 100% thrust level utilization during flight.

If the thrust level during flight operations is limited to some 95 or 90 % of the design value then the operational reliability increases. As shown in FIG. 2-12 the qualification effort and development cost can be reduced substantially by application of this strategy. The graph is based on results of Aerospace Corp. „Operations Design Model" prepared with data from the TITAN's rocket engines' operational flight history (ref. 95). A more detailed analysis on this aspect can be found in ref. 103.

The conclusion is that cost-engineering principles in this case would recommend „overdesigning" a rocket engine by some 10 %, compared to the flight thrust level requirement. This increases the initial development cost by about 6 % (according to FIGs. 2-08 and 2-09) but allows to reduce the number of qualification firings (at 100 % power level) to 50 %, resulting in a final cost reduction of 20 to 30 % (cf. FIG. 2-10).

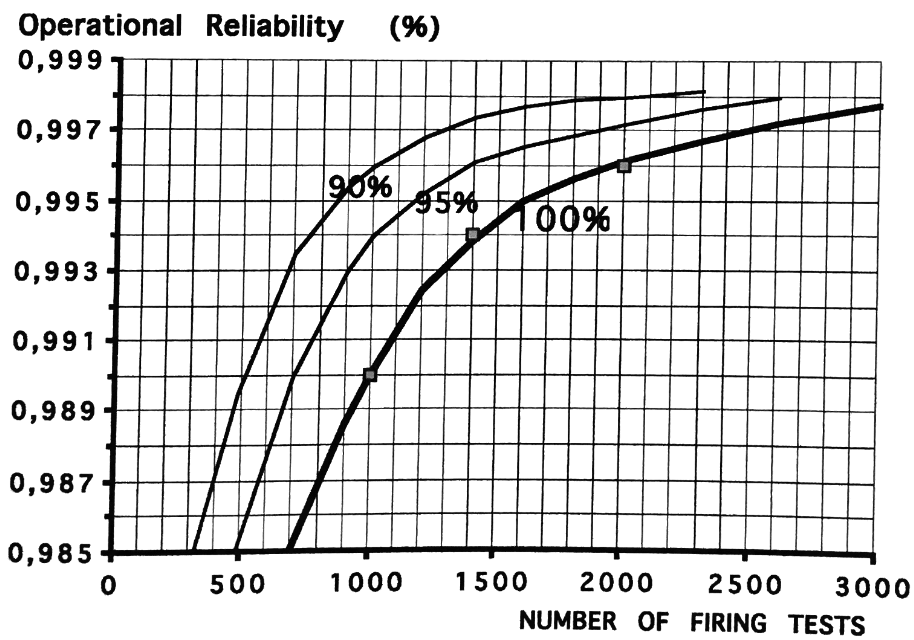


FIG. 2-12 : **Number of Qualification Tests vs. Reliability with Thrust Level Utilization during Flight (90 to 100 %)**

This principle is also supported by the practical flight experience with the SSME (Space Shuttle Main Engine): the reliability suffered by increasing the thrust level to 104% of the nominal level, and even more going to 109% power as shown in FIG. 2-13.

In order to realize higher reliability and lifetime in the future, new rocket engines are conceived with a reduced number of components (cf. TABLE 3-I) as well as to use hydrogen-resistant materials and powder metallurgy.

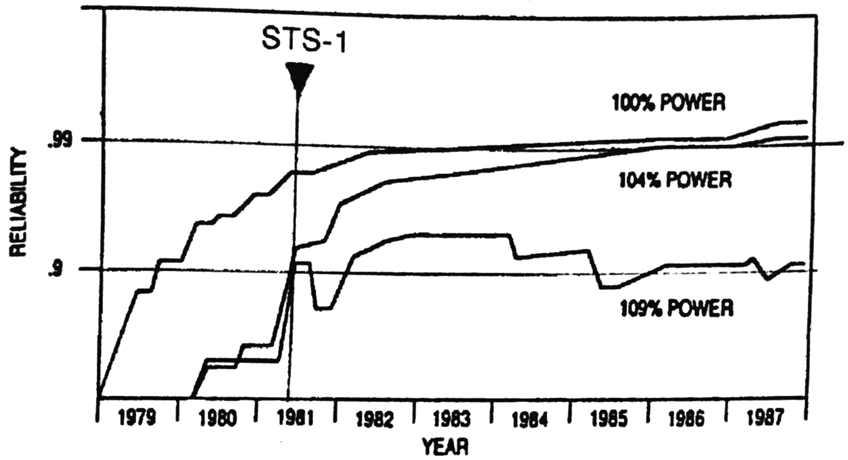


FIG. 2-13: **SSME Reliability vs.Time for three Power Levels** (ref. 89)

2.33 Pressure-fed Rocket Engines

Pressure-fed rocket engines are restricted to relatively low thrust levels (up to about 50 kN) and primarily used for small propulsion modules (kick stages) and for secondary propulsion systems (orbit and attitude control). The chamber

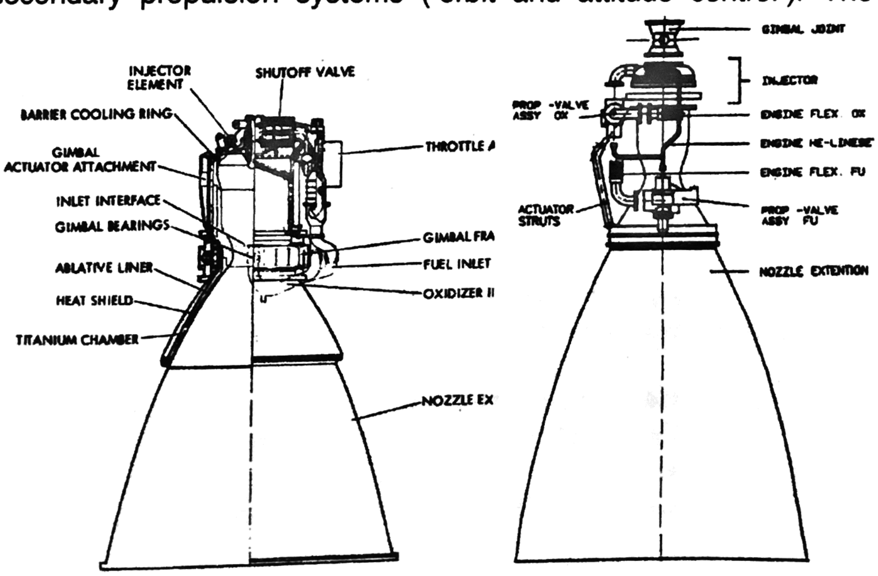


FIG. 2-14: **Examples of Pressure-fed Rocket Engines (LEM Ascent Engine and Aestus)**

pressure needs to be as low as possible in order to keep the propellant tank pressure and the related tank weight down. All engines are using „storable propellants“ like nitrogen tetroxyde and MMH or hydrazine mixtures. The deletion of the complex turbopump system in combination with a relatively low chamber pressure reduces the required development effort substantially. FIG. 2-14 shows as examples for this type of rocket engines the Lunar Module (LEM) Ascent Engine and the AESTUS engine for the ARIANE 5 third stage.

The CER for pressure-fed engines can be derived from seven reference projects in the thrust range from 0.4 to 50 kN, as shown in FIG. 2-15 :

$$\boxed{H_{EP} = 167 \; . \; M^{0.35} \; f_1 \; f_3} \qquad \text{MYr}$$

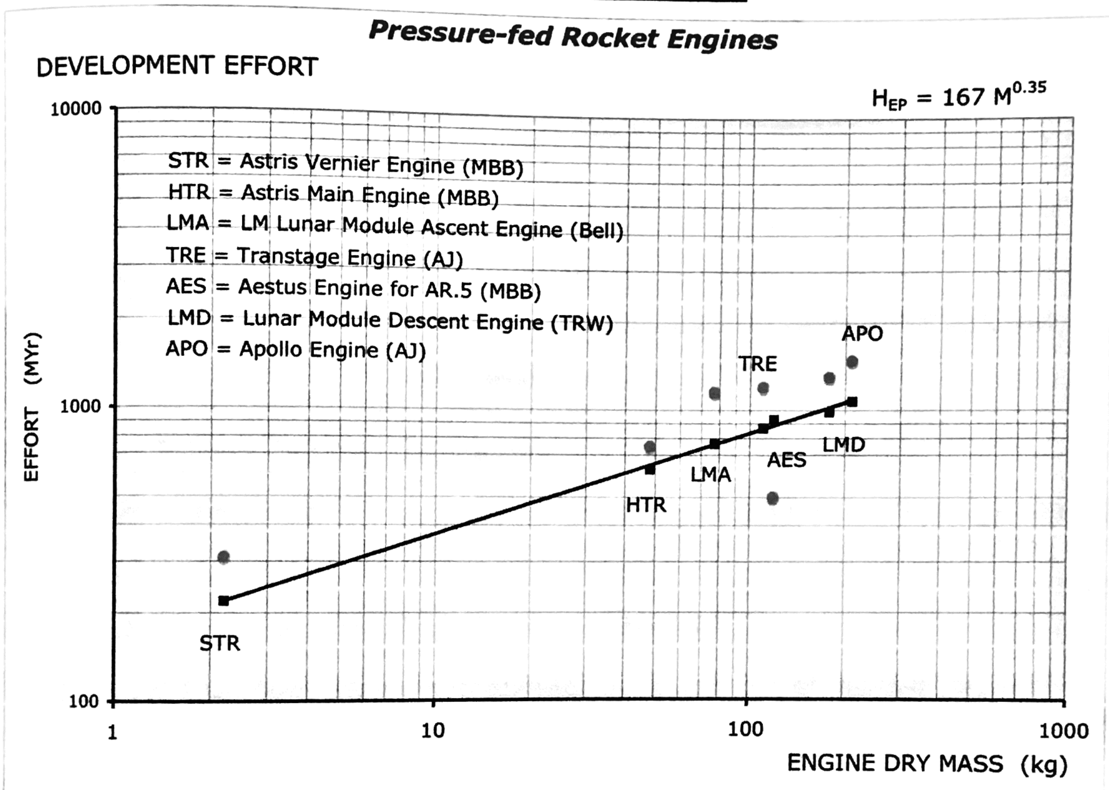


FIG. 2-15: **Basic CER for Pressure-fed Rocket Engines**
(Triangles: original data, squares: reference data with regression factors applied)

A Technical Quality Factor has not been established since the seven reference projects show a very good correlation to the basic CER. The data spread is only +/- 7 % (see FIG. 2-15). The reliability vs. number of qualification tests follows a similar trend as shown in FIG. 2-10. The AESTUS-engine had 171 development and qualification firings with 6 engines (ref.128). It failed on the 10th flight (hard ignition) and required 63 additional test fitings (in addition to 78 ignition tests) in order to qualify a modified ignition sequence (AW, 31.1.02).

2.34 Airbreathing Turbo- and Ramjet-Engines

This type of propulsion system has been considered in the past years for a variety of space launch systems. Either for the first stage propulsion of TSTO vehicles or even as the most advanced concept in a combined propulsion system for a winged SSTO vehicle (NASP).

2.341 Technical Options and Applications

(1) TURBOJET or TURBOFAN engines are the most common type of airbreathing engines; they are, however, limited to a speed of about Mach 3.5. A large number of such engines has been developed ; FIG. 2-16 shows the SNECMA M53 engine as a typical turbojet engine design.

While the specific impulse values of these engines are very high also the engine mass is large. The thrust-to-weight ratio - in contrast to rocket engines - does not

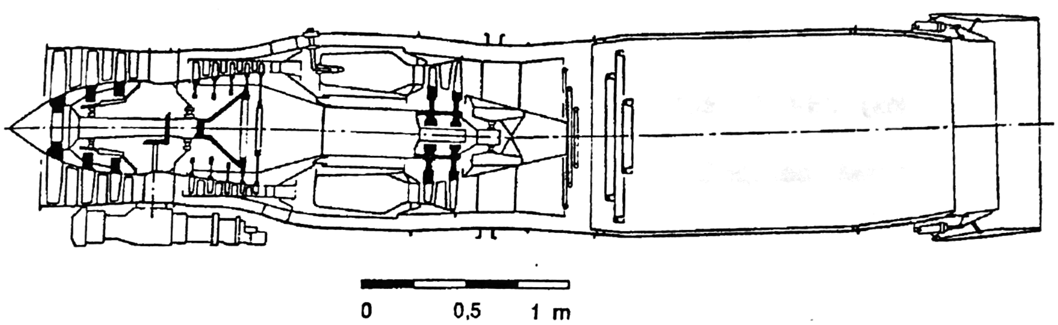


FIG. 2-16: **Typical Turbojet Engine Design** (SNECMA M53)

improve with size (or thrust level) but depends mainly on the technology applied. FIG. 2-17 shows the mass of turbojet engines and the thrust-to-weight ratio (8 to 10) vs. thrust level on ground. This compares to a range of 60 to 70 for rocket engines (SL thrust), or to 25 to 30 for ramjet engines.

(2) RAMJET ENGINES have been applied to a number of missile projects but not yet in a high-performance version for high-speed transport vehicles. Such hydrogen-powered ramjet engines have been studied in detail, and experimental work has been performed in the USA, in France and Germany (Hypersonics Technology Program). Ramjet propulsion can be applied up to Mach 7 with subsonic combustion. Since ramjet engines operate only above a speed of Mach 2 the combination with turbojet engines or rocket propulsion (i.e. air-ejector rocket) is required for the initial flight phase.

For flight speeds higher than Mach 7 a supersonic combustion system is required, called *Scramjet - Propulsion* which can be used in principle up to a speed of Mach 15. However, for technical and cost-engineering reasons it is probably better to switch already at Mach 12 to rocket propulsion.

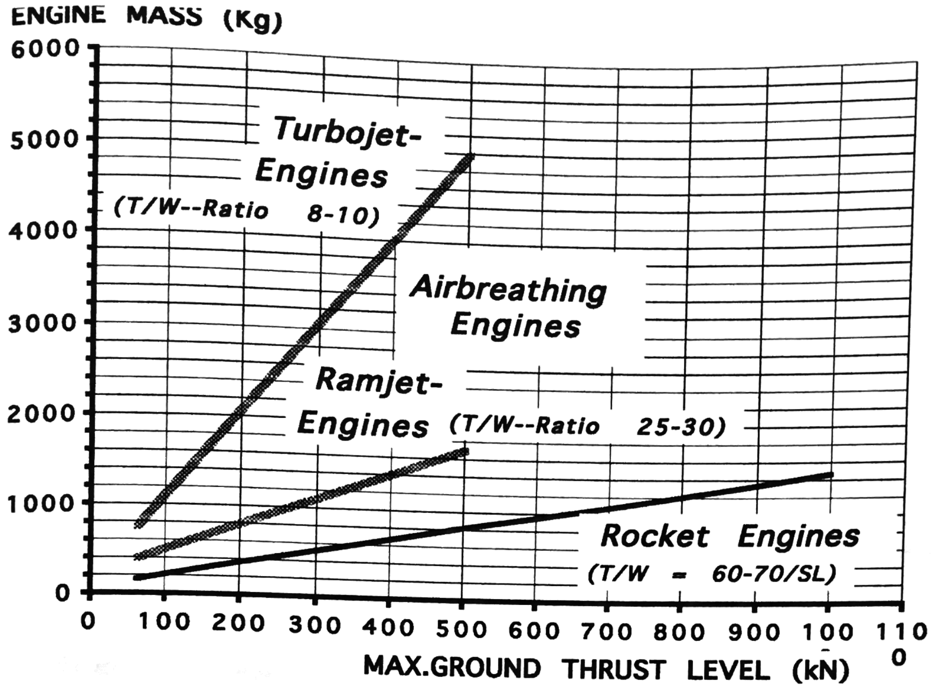


FIG. 2-17: **Turbojet Engines', Ramjet and Rocket Engines' Mass Comparison and Thrust-to-Weight Ratio vs. SL Thrust Level**

For the Turbo-Ramjet combination which is very suitable for longer cruise flight requirements, two options are feasible: the integrated or tandem configuration (requiring a separate diverter duct) or the parallel engine arrangement with common air-intake. The second option allows an independent optimization of both the turbojet engine and the ramjet engine, also the ramjet duct can be used above Mach 1 to produce hot air for base drag reduction; this concept is shown in FIG. 2-18. A major problem is the variable high-temperature air intake and the asymmetric nozzle which requires both variable throat and thrust vector control mechanisms.

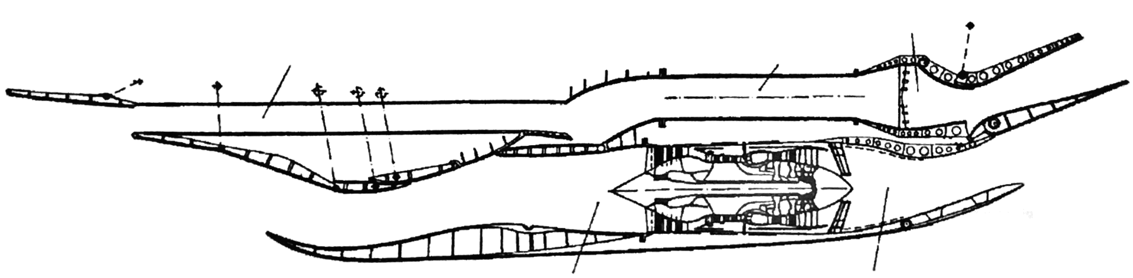


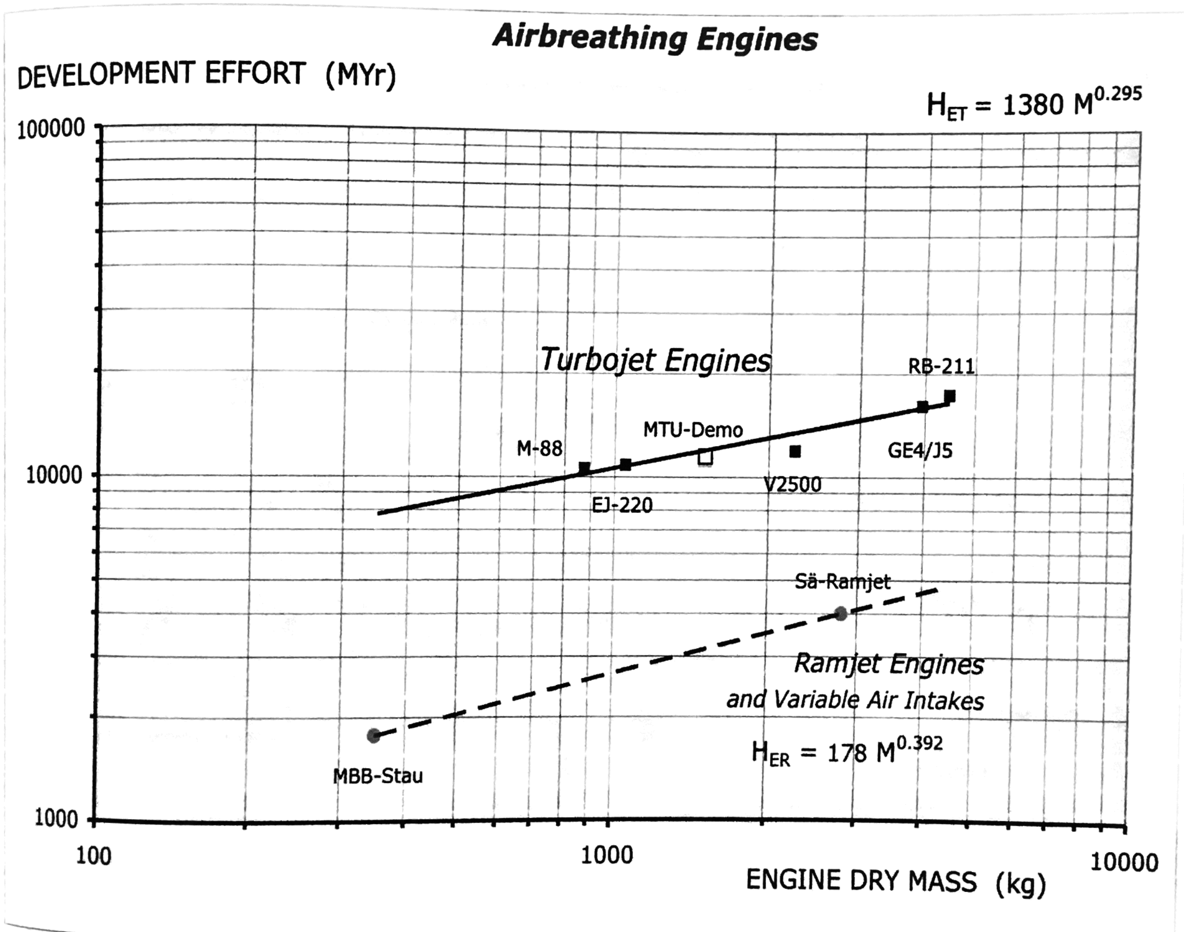
FIG. 2-18: **Typical Turbo-Ramjet Combined Propulsion System for Speeds up to Mach 7** (MBB / Dasa Concept Design)

2.342 Cost Estimation

For Turbojet and Turbofan Engines it is possible to establish a CER from five realized engines plus three detailed project cost studies, as shown in FIG. 2-19. A Technical Quality Factor has not yet been derived since the number of reference projects is relatively small and the spread of the cost values is within +/- 13 %. The complete CER is

$$\boxed{H_{ET} = 1380 \cdot M^{0.295} \quad f_1 \quad f_3} \qquad \text{MYr}$$

For Ramjet engines no CER can be established at this time because of the complete lack of realized reference projects. The only way for a provisional approach is the use of CERs which have been derived for „hot structures". In fact the variable large air intakes required for a launch vehicle are part of the engine



2-19: **Basic CER for Airbreathing Engines with Reference Projects**
(Solid Squares = Realized Engines, Hollow Squares = Detailed Project/Cost Studies)

and the vehicle design, representing very complex hot structures. In FIG. 2-19 which shows the CER for Turbojet engines, the lower parallel line indicates the best possible cost trend for ramjets. The *specific* development cost are only half as much as for turbojet engines since the mechanical combustor design is relatively simple (no turbomachinery). This preliminary CER is valid for both the engine with nozzle as well as for the air intake. Both have to be dealt with separately so that the total cost of the ramjet engine plus air intake assembly will become similar to the development cost of turbojet engines.

The number of qualification tests for new jet engines is some 12000 endurance cycles before flight testing. This is much more than for rocket engines and leads, accordingly, to a much higher reliability of about 0.9999.

2.4 Vehicle Systems' Development CERs

2.41 Large Solid-Propellant Rocket Boosters

This group of propulsion systems comprises large solid-motor Boosters which are used mostly as pairs representing a launch vehicle propulsion stage, such as in case of the TITAN III/ IV-family, the Space Shuttle or ARIANE 5. As a typical example, the ARIANE 5- Solid Rocket Boosters are shown in FIG. 2-20.

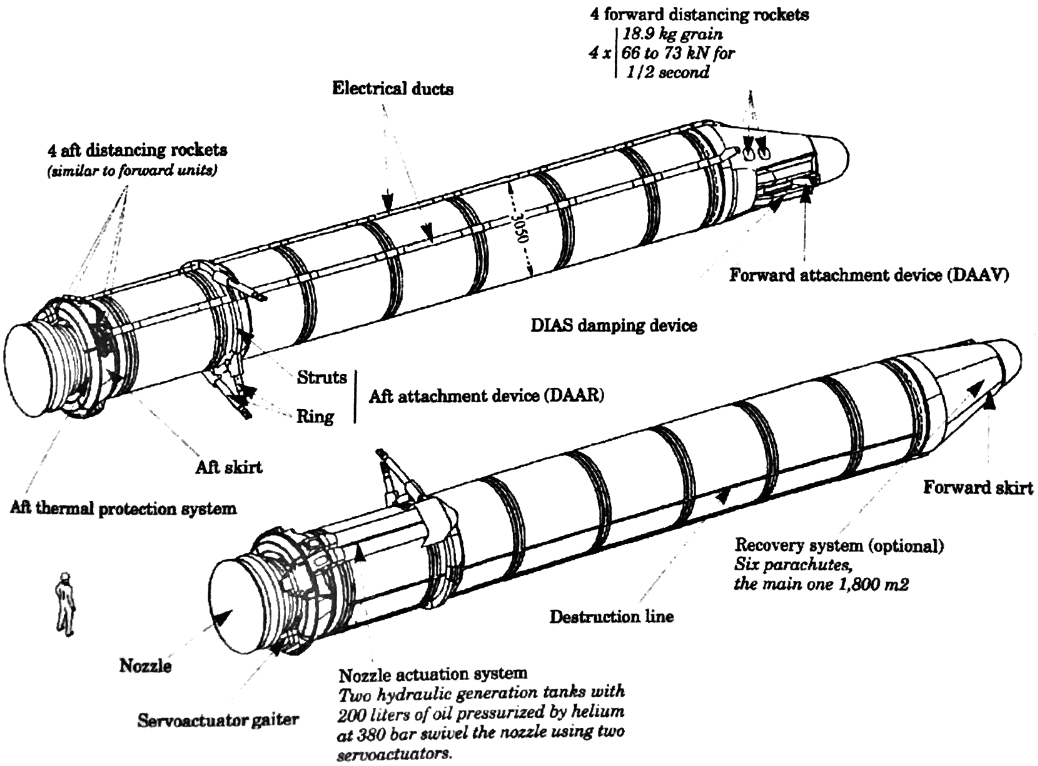


FIG. 2-20 : **ARIANE 5 Solid-Propellant Rocket Boosters**

The large solid-propellant motors need to be equipped with a steerable nozzle or fluid-injection systems for attitude control. They have additional mechanical equipment such as attachment and separation systems, and an aerodynamic shroud.

For *reusable* Boosters (such as used for the Space Shuttle) additionally parachutes plus recovery and flotation equipment is required, as well as location systems and instrumentation. The design for re-use also implies higher structural design factors, thus higher mass. The SRB recovery from sea adds damage from impact and salt water. Special recovery ships have to be built in this case which increases the development cost and later contributes much to the direct and indirect operations cost. The net mass of fully equipped Boosters is 25 to 30 % higher than the mass of the basic rocket motor as shown in FIG. 2-05.

For the definition of the basic CER there are five reference projects available, as shown in FIG. 2-21. The resulting cost estimation relation is as follows:

$$\boxed{H_{VR} = 10.4\ M^{0.60}\ f_1\ f_3} \quad \text{MYr}$$

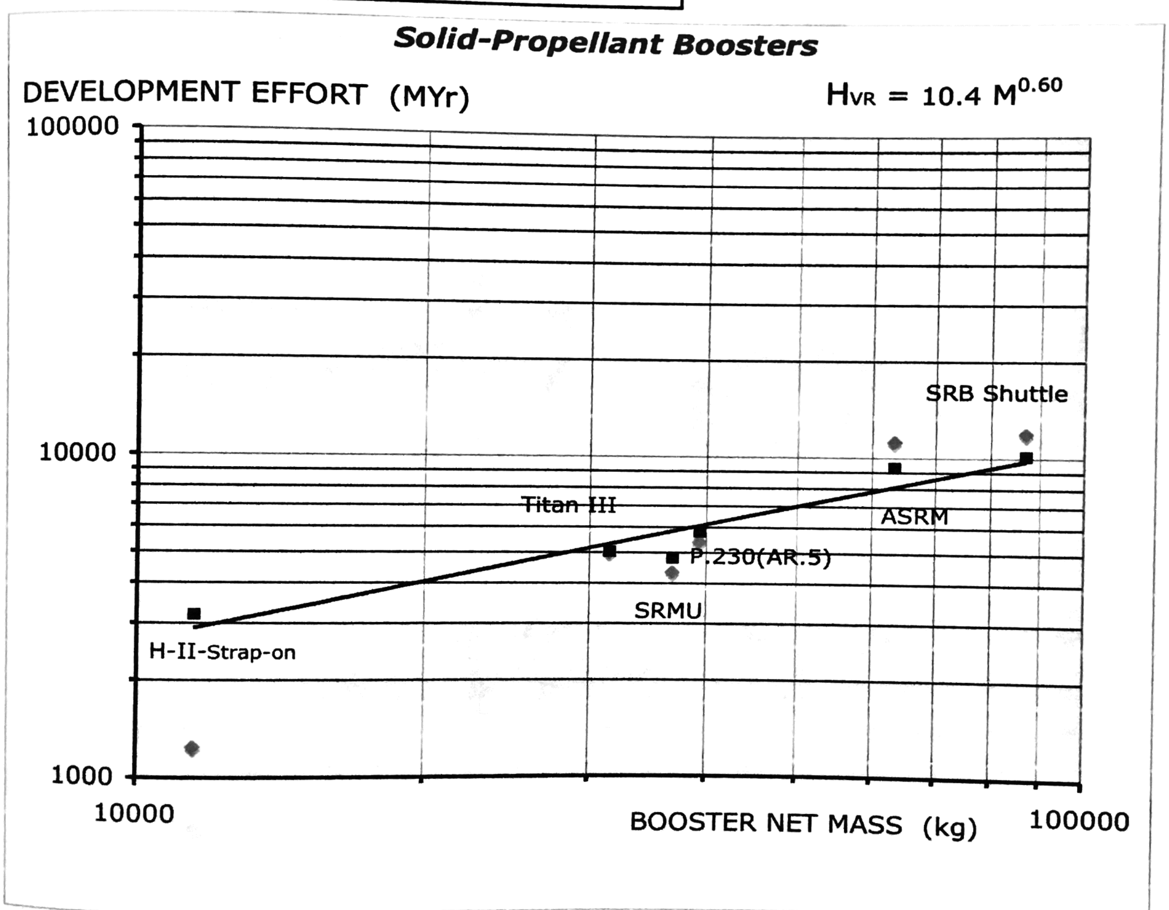


FIG. 2-21: **Solid-Propellant Booster Systems CER with Reference Projects**
(Red rhombs are original data, black squares are the reference points after regression process)

A Technical Quality factor has not yet been defined for this group of systems due to the small number of projects. However, the data fit is satisfactory, taking into account that the ASRM (Advanced Solid Rocket Motor for the Shuttle) did not reach operational status. The Shuttle Booster cost are higher due to the (partial) recovery mode it was designed for.

2.42 Liquid Propellant Propulsion Systems/Modules

This group of space systems comprises both propulsion modules as well as spacecraft-integrated bipropellant systems. „Propulsion Modules" are are propulsion systems with their own basic structure (where all elements are attached to), but no external (load-carrying) structure, no own power supply and no own intelligence equipment like telemetry or guidance and control - these functions are taken over by a separate module (like in case of the ARIANE launch vehicle) or by another system (i.e. the spacecraft, as in case of the GALILEO Retro Propulsion Module (RPM). The ARIANE 5 Propulsion Module EPS or L.9.5 and the GALILEO Spacecraft (Jupiter Orbiter) Retro Propulsion Module are shown in FIG. 2-22 as illustration for the design options of propulsion modules.

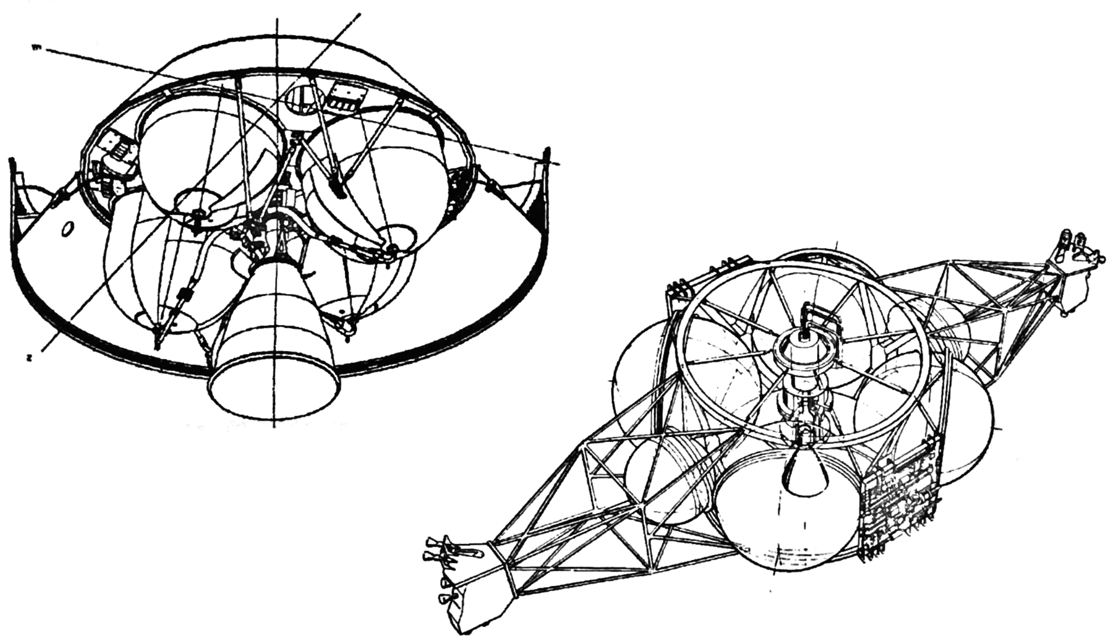


FIG. 2-22 : **Propulsion Module Examples: ARIANE 5 EPS and GALILEO RPM (Retro Propulsion Module)**

Another assistance in the definition of propulsion modules and integrated propulsion systems is provided in FIG. 2-23 with the dry mass vs. propellant mass of :

(A) Spacecraft-integrated Propulsion Systems (such as the roll control system (SCA) of ARIANE 5, and

(B) Propulsion Modules, such as the GALILEO Retro Propulsion Module (RPM)

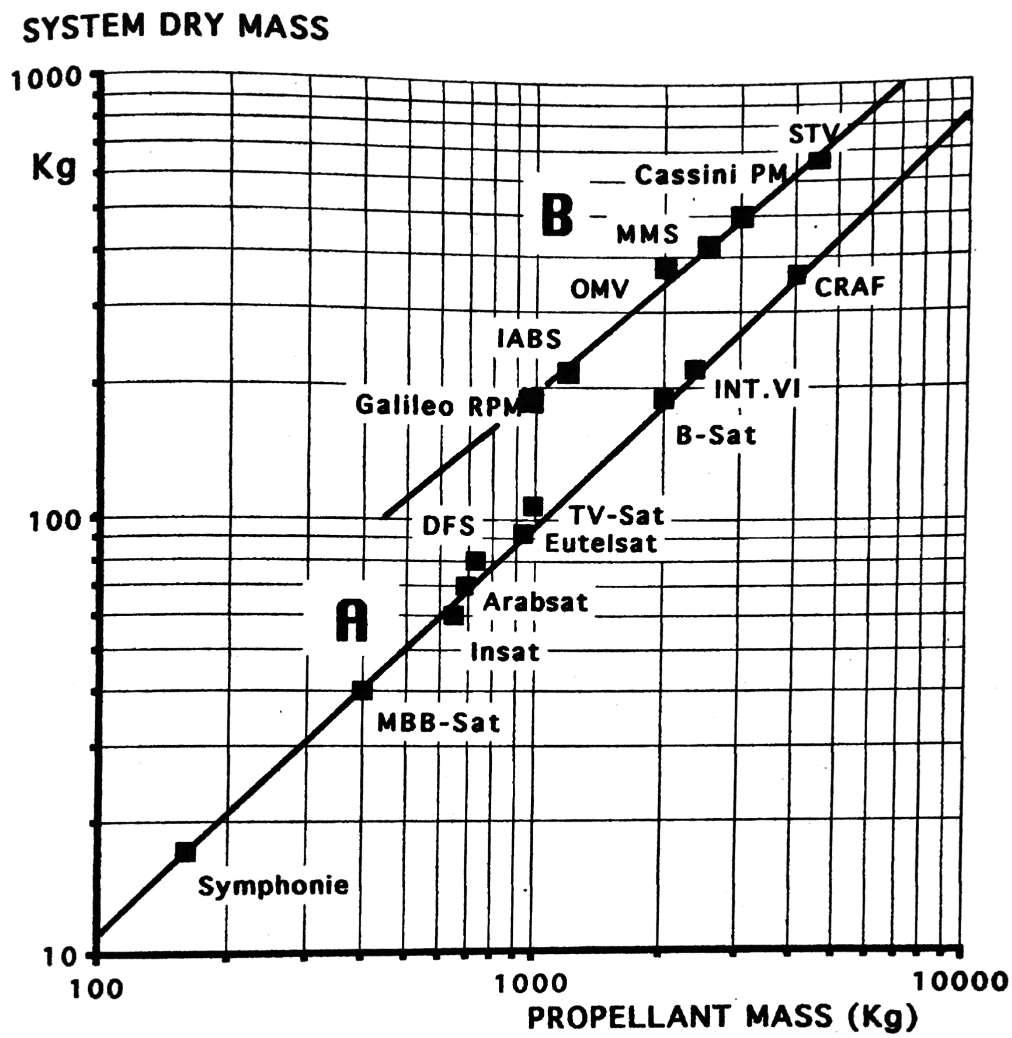


FIG. 2-23 : **Dry Mass Trend of Spacecraft-Integrated Propulsion Systems (A) and Propulsion Modules (B)** - Storable Propellants

The basic CER has been established on the background of 5 reference projects plus the contracted cost for the European ATV (Automated Transfer Vehicle). The resulting development effort for Propulsion Modules is substantially lower than for complete stage vehicles (next chapter). The CER for propulsion modules has been defined as

$$\boxed{H_{VP} = 14.2 \cdot M^{0.577} \quad f_1 \quad f_3} \qquad \text{MYr}$$

The reference data spread related to the basic CER is only +/ - 5 %, as depicted in FIG. 2-24. Exceptions are the GLL-RPM and the STV-Project. The STV, however, is a type 3-estimate (cf. chapter 2.12) and the GLL Propulsion Module is exceptional since it was the first unified propulsion system with very high redundancy and high verification effort for a long-term (12 year) interplanetary mission.

A specific quality factor f_2 for this group of technical systems has not (yet) been established. One important factor or cost driver may be the system's required operational lifetime.

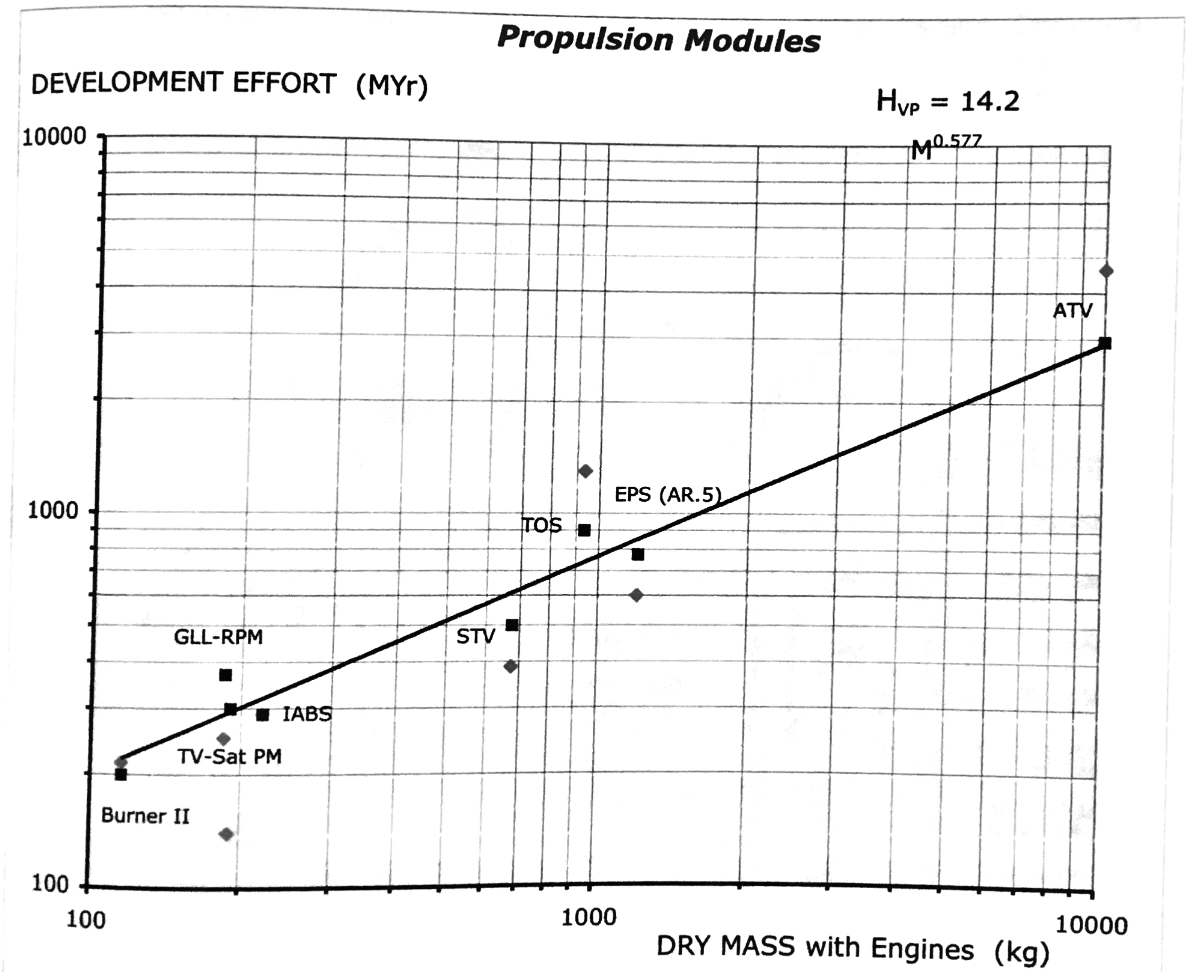


FIG. 2-24: **Basic CER for Propulsion Systems/ Modules with five (7) Reference Projects**

The dry mass - only in this case - includes the weight of the engines. Engine development cost, however, are NOT included in the CER-derived cost. The use of existing engines is assumed. If this is not the case the engine development cost have to be estimated separately according to the CER of chapter 2.33.

2.43 Expendable Ballistic Stages and Transfer Vehicles

This group of space systems comprises a large family of vehicles used as launch stages, upper stages and Orbital Transfer Vehicles. The vehicle DRY MASS serves as reference for the CER, however, without the main propulsion engine(s). Since there often existing engines are being used, these have to be considered separately. In addition, there are single-engine vehicles, or such with multiple engines, a design feature which influences strongly the development cost of the complete vehicle. As examples FIG. 2-25 shows the largest vehicle built up to now, the SATURN V first stage, and the CENTAUR upper stage, the first cryogenic vehicle developed.

FIG. 2-26 shows the Dry Mass Fraction and the Net Mass Mass Fraction (Dry mass plus residuals, maneuver propellants and reserve) for vehicles with cryogenic propellants (LOX/ LH_2) illustrating the strong influence of the propellant mass

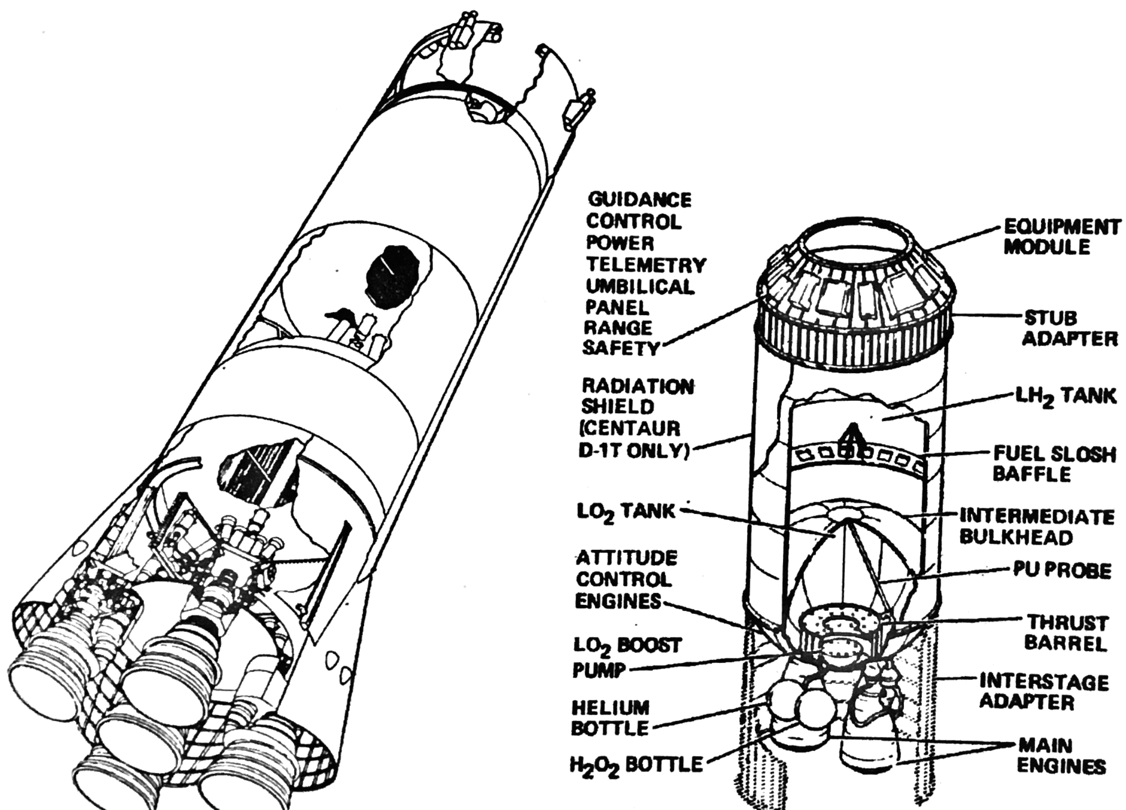


FIG. 2-25 : **Examples of Ballistic Expendable Stage Vehicles** (SATURN V-S-IC- Stage and the CENTAUR Upper Stage and Transfer Vehicle)

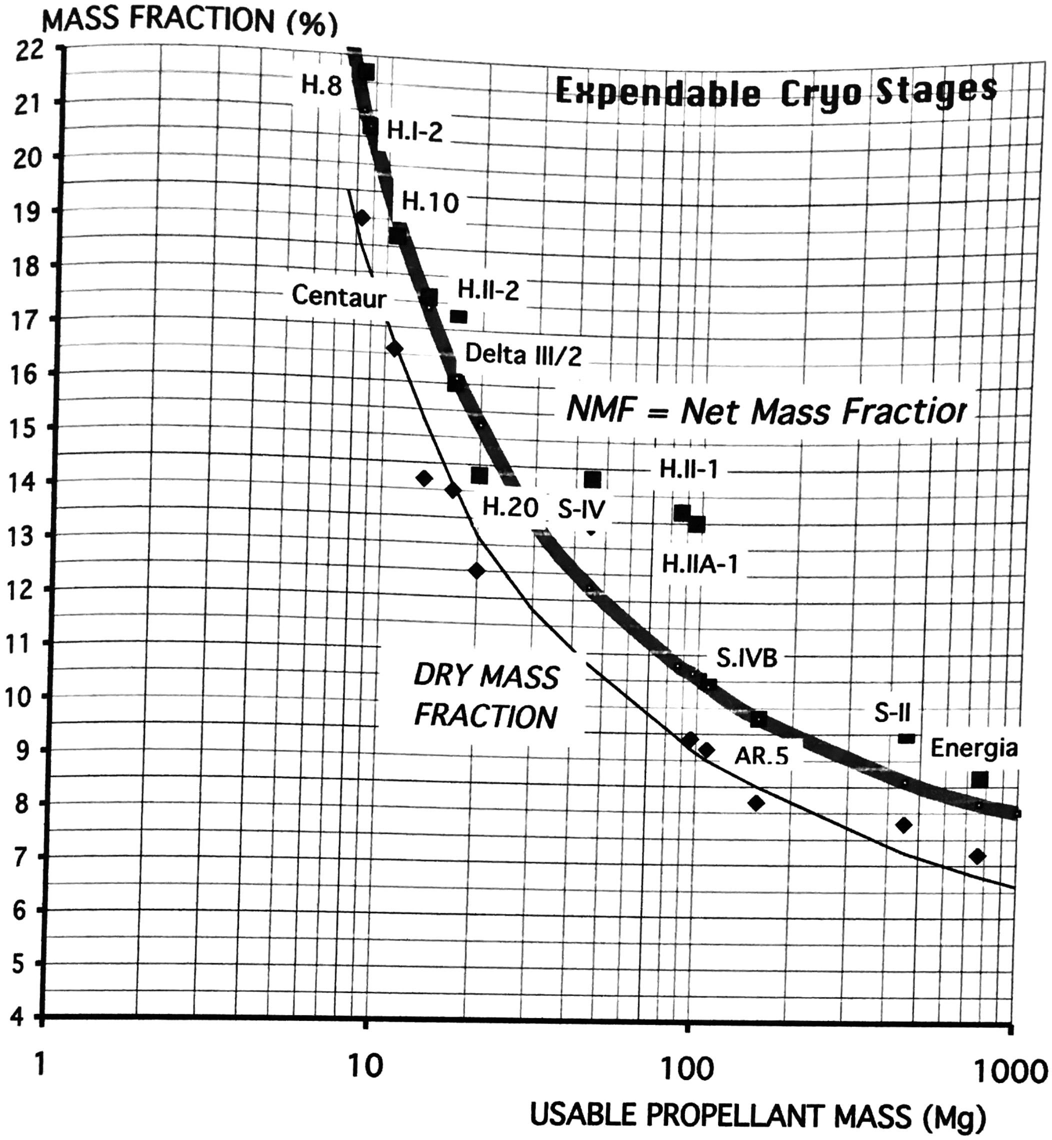


FIG. 2-26 : **Net Mass Fraction and Dry Mass Fraction (%) of Expendable Vehicles with Cryo Propellants vs. Size (Propellant Mass)**

For this type of vehicles exists a relatively good data base which comprises 15 projects, as shown in FIG. 2-27. This larger data base resulted in an improved CER which is defined as follows :

$$\boxed{\mathbf{H}_{VE} = 100 \cdot M^{0.555} \quad f_1 \quad f_2 \quad f_3} \qquad \text{MYr}$$

Both, vehicles with cryogenic propellants (LOX/ LH_2) as well as vehicles with storable - or better medium-energy-propellants - are covered by this CER. The

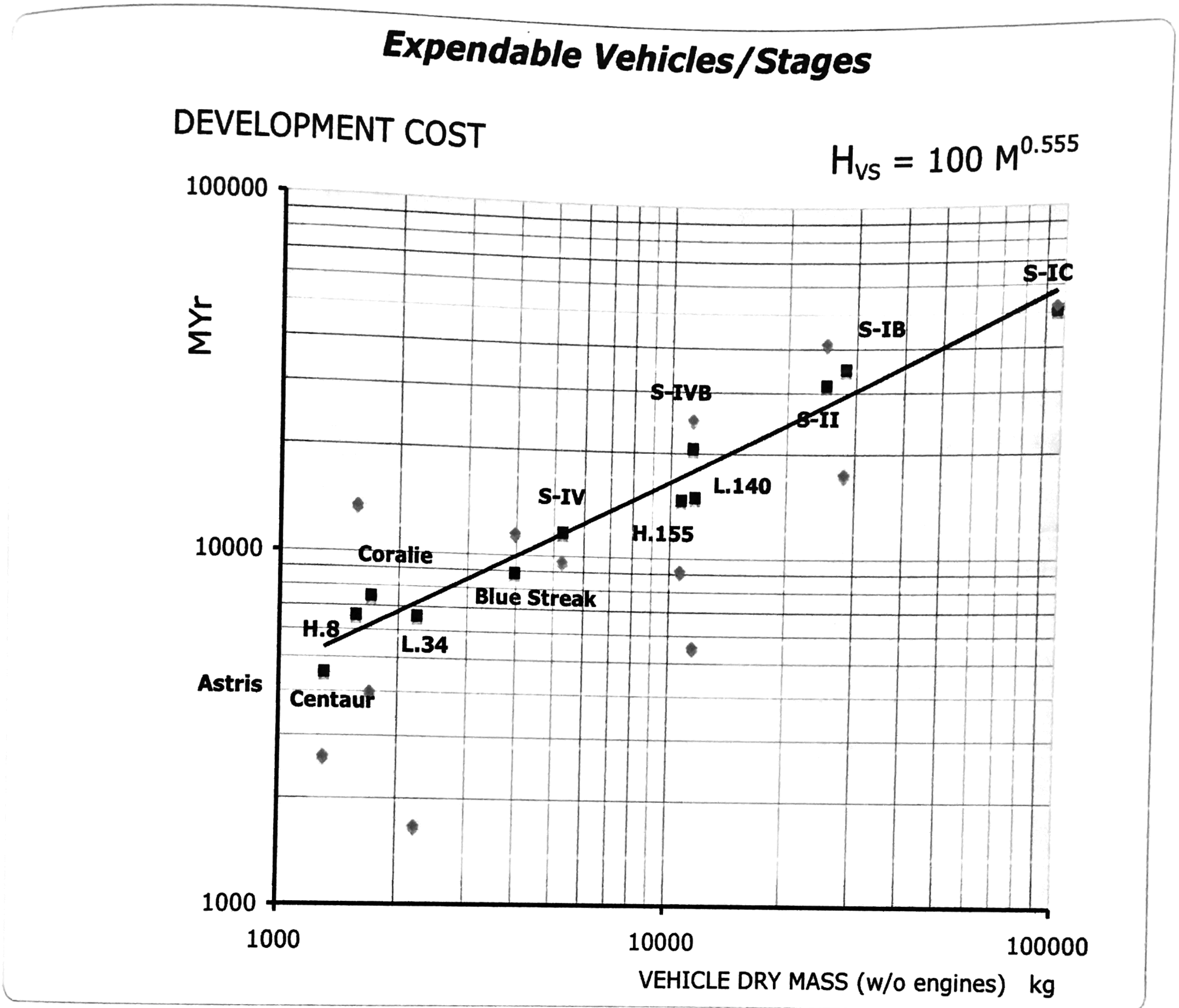


FIG. 2-27: **Basic CER for Expendable Stage Vehicles with the Reference Projects** (Square points are normalized values; triangles the original cost data)

higher costs of cryogenic stages are taken care of by the inherently higher dry mass. There is also a relatively great cost data spread which is caused by the level of technology employed, the safety margins, tank pressures, etc., but this is taken into account by the selected Technical Quality Factor. Without the f_2-Factor the large data spread would not allow to establish a CER. With f_2 as defined here, the accuracy, resp. spread range of the reference projects with respect to the CER is +16/ -18 %.

Technical Quality Factor :

For the Technical Quality Factor f_2 the vehicle net mass fraction (NMF) is being used. Net mass is the dry mass plus residuals and gases at cut-off , here without the engine(s) mass, related to the (usable) propellant mass:

$$k^* = (M_n - M_e) / M_p.$$

The Technical Quality Factor is defined as the ratio of the average NMF to the specific NMF:

$$f_2 = k_{ref} / k_{eff}$$

with k_{ref} = average reference net mass fraction as defined in FIGs. 2-28 and 2-29.
k_{eff} = effective net mass fraction of the vehicle concept or the existing vehicle.

For the sake of simplicity the net mass fraction figures have been subdivided in two: one for vehicles using Hydrogen/Oxygen, and the other for all other propellant combinations often referred to as „storable propellants" or „medium-energy propellants" such as LOX/ Kerosene and UDMH/ N_2O_4 (FIG. 2-29).

Clearly the vehicle dry mass is a measure for the technology employed, and the technology level is a cost driver. In addition, this definition of a mass-related

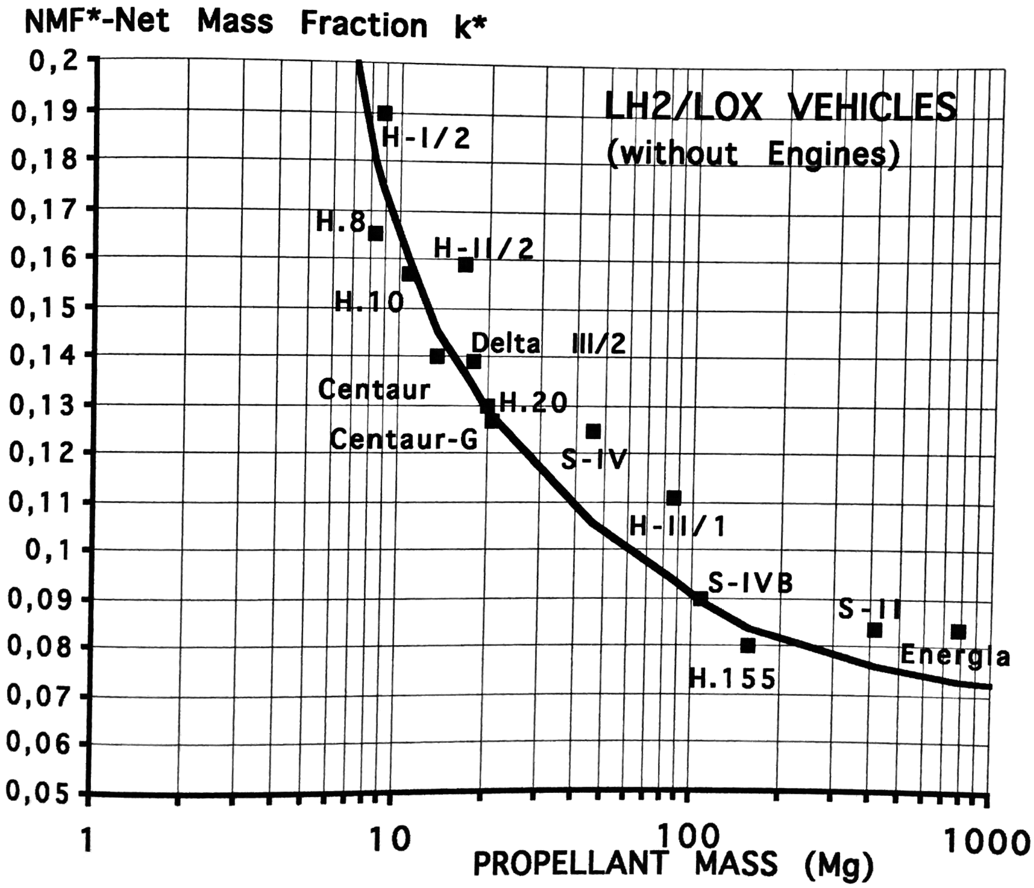


FIG. 2-28: **Reference Curve for Quality Factor f_2 for Ballistic Expendable LOX/LH_2 Vehicles, based on the Net Mass Fraction k***

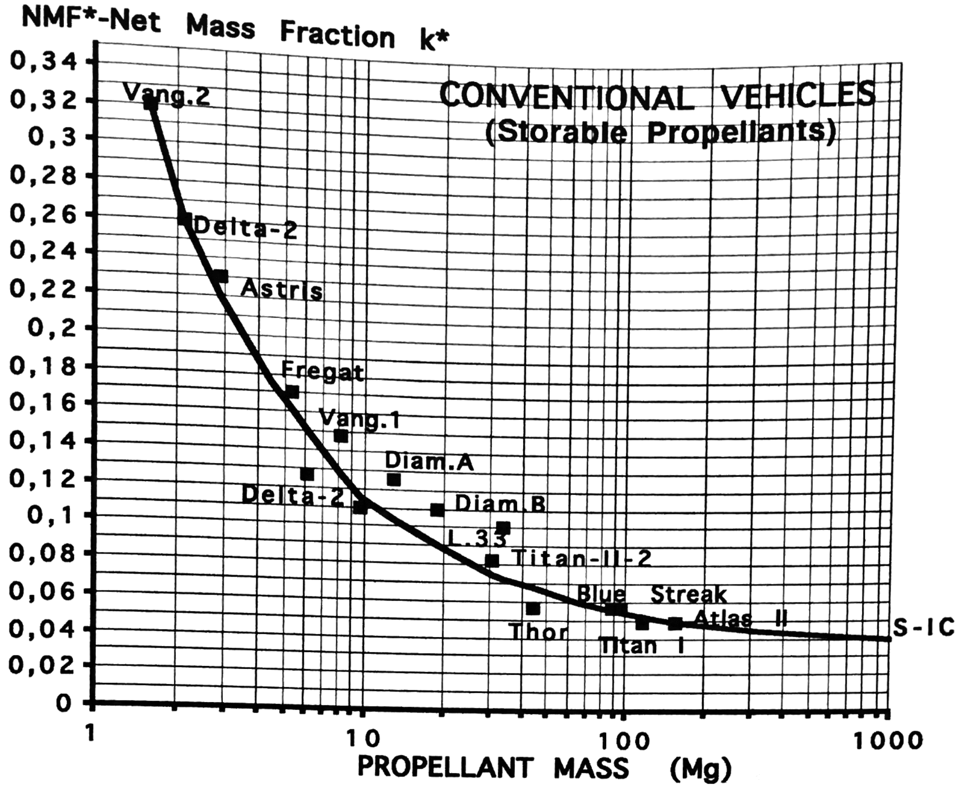


FIG. 2-29: **Reference Curve for the Quality Factor f_2 for Vehicles with Medium-energy or Storable Propellants, based on the Net Mass Fraction k*** (excluding engines)

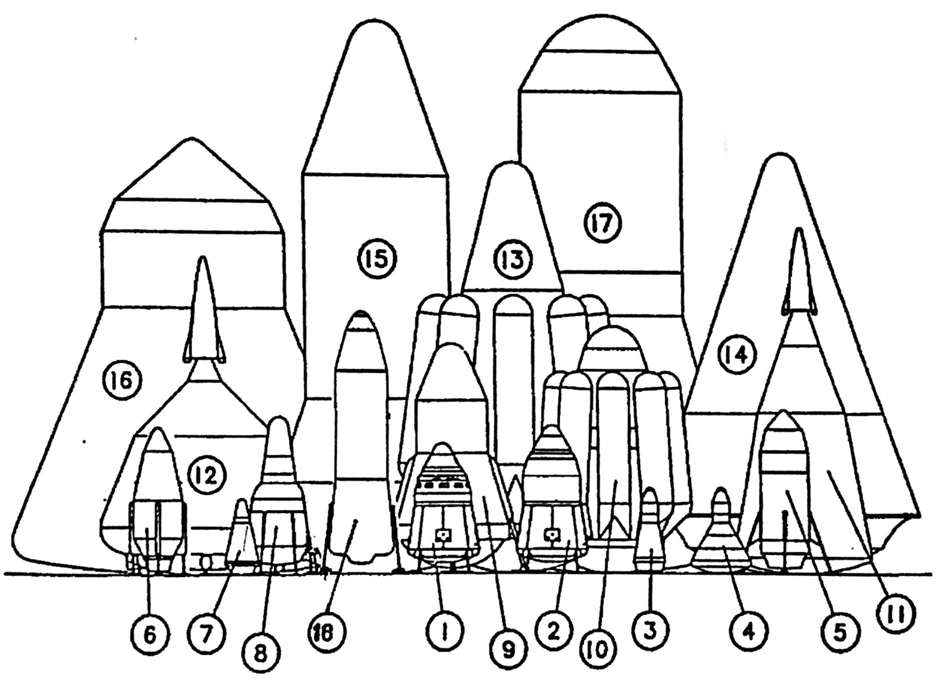
quality factor avoids the situation of one specific stage design to become more expensive by application of a low-tech heavy design (because the basic CER is dry mass - related).

In other words: Costs are increasing with dry mass for scaled vehicles. For a point design, however, with constant propellant mass, a higher dry mass due to simpler (existing) technology leads to *lower* development cost.

The drive for high-tech solutions and materials decreases dry mass and launch mass, but **it increases costs.** This may be acceptable in some cases but normally in space transportation the vehicle design (and operations) must be targeted for minimum cost. Therefore, in a cost-engineered optimized design a careful trade-off between dry mass vs. development cost has to be performed.

2.44 Reusable Ballistic Launch Vehicles

Ballistic reusable launch systems have been considered for a long time as candidates for the next generation of cost-efficient launch vehicles. The first concepts were conceived already in the 1962-1963 period: ROOST by Phil Bono and NEXUS by Krafft Ehricke. Quite a number of project studies have been performed in the past decades on this type of advanced launch vehicles as illustrated in FIG. 2-30 (ref. 53). Two-stage, single-stage and booster-assisted vehicles have been studied - however primarily either as too small vehicles with marginal performance (No.1 to 8), or as very large vehicles - too large for being a realistic option (No.13 to 17). The more recent detailed vehicle designs in the medium-size class were the Japanese KANKOH MARU and the *MBB* BETA II Concept with rear re-entry (heat-shield integrated engines), the McDonnell Douglas DC-Y and the DASA Festip Concept FSSC-3 with front re-entry, all with 8 Mg LEO payload and some 600 Mg launch mass (FIGs. 2-31 and 2-32).



1) Phoenix E (1985) PacAm
2) Phoenix C (1982) PacAm
3) Phoenix L (1974) Hudson
4) Phoenix L' (1976) Hudson
5) S-IVB (1965) Douglas
6) SASSTO (1967) Douglas
7) ATV (1972) NASA Marshall
8) BETA (1969) MBB-Germany
9) Hyperion (1969) Douglas
10) Pegasus (1969) Douglas
11) Unnamed (1970) NASA-OART-MAD
12) SERV (1971) Chrysler
13) ROMBUS (1964) Douglas
14) NEXUS (1964) General Dynamics
15) Unnamed (1983) Rockwell
16) Unnamed (1977) NASA-JSC
17) "Big Onion" (1976) Boeing
18) BETA II (1986) MBB

FIG. 2-30: **Historic Overview of Reusable Ballistic Vehicle Concepts**

The front re-entry mode in combination with aerodynamic control surfaces allows a larger cross-range for landing, does require, however, a 180° turn maneuver of the vehicle before the final landing approach.

The interest in this type of vehicles as a candidate for future launch systems has been revived with the test flights of the DC-X vehicle by McDonnell Douglas in 1993-95 and a number of studies confirming the feasibility of single-stage(to LEO) reusable vehicles. Inspite of the fact that no such vehicles have been realized yet they are covered in this report as an example that and how new vehicles types can be dealt with in a cost model - and not only historic type vehicles as sometimes stated as disadvantage of statistics-based cost models.

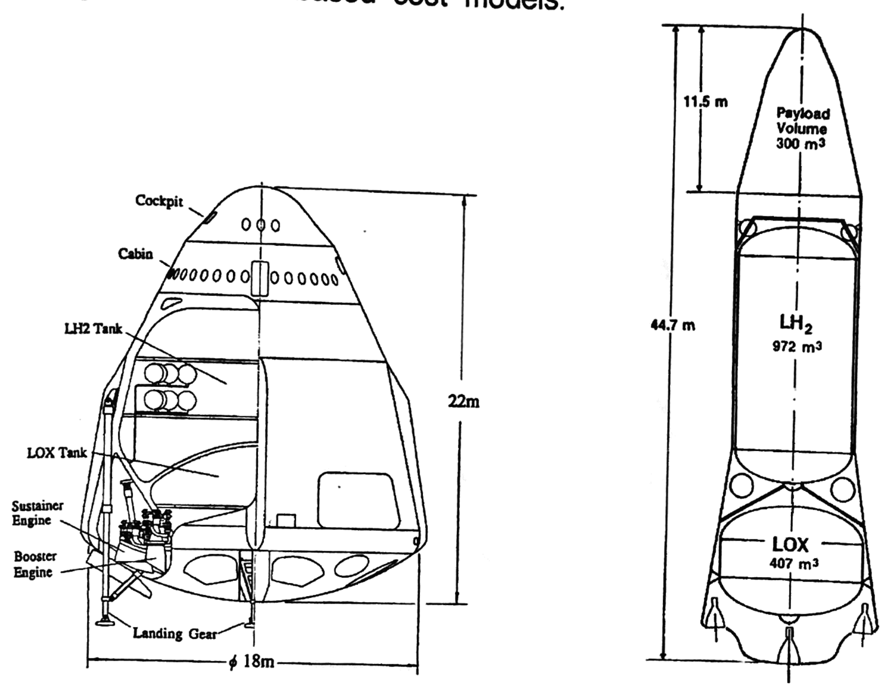


FIG. 2-31: **Ballistic Reusable SSTO-Vehicle Concepts with Rear Re-Entry Mode** (KANKOH MARU with heat shield-covered engines, and BETA IIA with a plug-cluster engine arrangement)

The required vehicle net mass (including orbital and landing propellants) is an important costing parameter. Therefore, the vehicle dry mass fractions and net mass fractions have been assembled from the major project studies performed (FIG. 2-33). The net mass of reusable ballistic vehicles is about 50 % higher than for expendable vehicles with the same propellant mass. This is due to higher safety factors in the structural design (required by the reusability and repeated load cycles)), thermal protection for re-entry, additional equipment for integrated check-out or health control systems and increased redundancy. The higher net mass also requires a larger propellant mass (for the same payload) which again increases the net mass.

The NMF and DMF curves of FIG. 2-33 represent an nominal design reference. The actual values vary, depending on the vehicle design principles (i.e. modularity

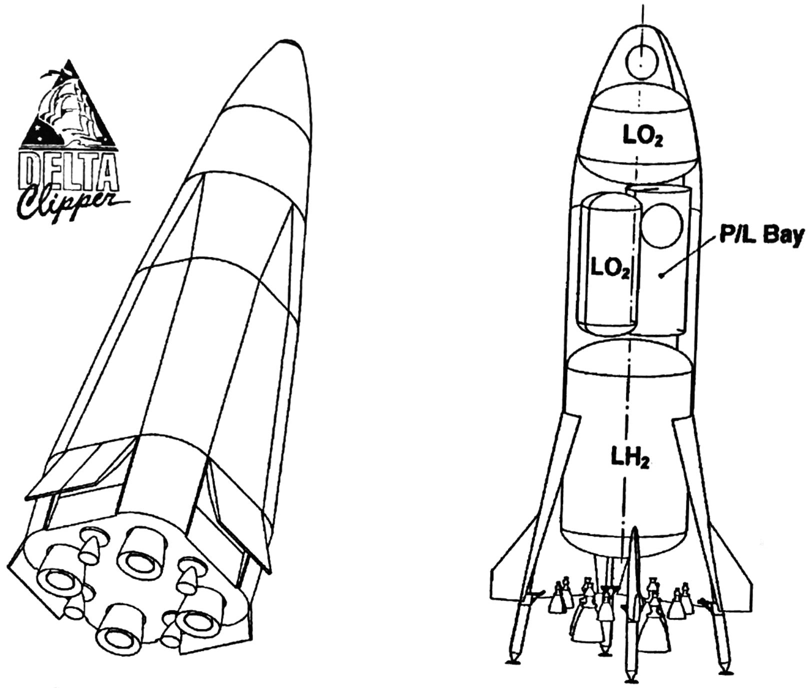


FIG. 2-32: **Ballistic SSTO-Concepts for Front Re-Entry Mode with Aerodynamic Control Surfaces** (DC-Y and FESTIP FSSC-3)

will increase the values), the structural safety factors employed, and the type and number of rocket engines (a multi-engine concept will increase the NMF value due to the lower thrust-to-weight ratio, compared to a single large engine). The difference between dry mass and net mass is 22 to 25 %, related to the vehicle dry mass. It includes the propellants for orbital operations (orbit injection, descent impulse), attitude control and for the active landing maneuver (retro impulse).

TABLE 2-I : **Safety Design Factors for Launch Vehicles** (ref.121)

	Non-pressurized Structures	**Pressurized Structures**	**Lines/Ducts** (> 4 cm dia.)
Expendable Launch Vehicles	1.1	1.25	1.25
Space Shuttle Orbiter	1.35	1.8	1.5
Reusable Launch Vehicles	1.35 - 1.5	1.8 - 2.0	2.5
Commercial Aircraft	1.5	2.0	2.5

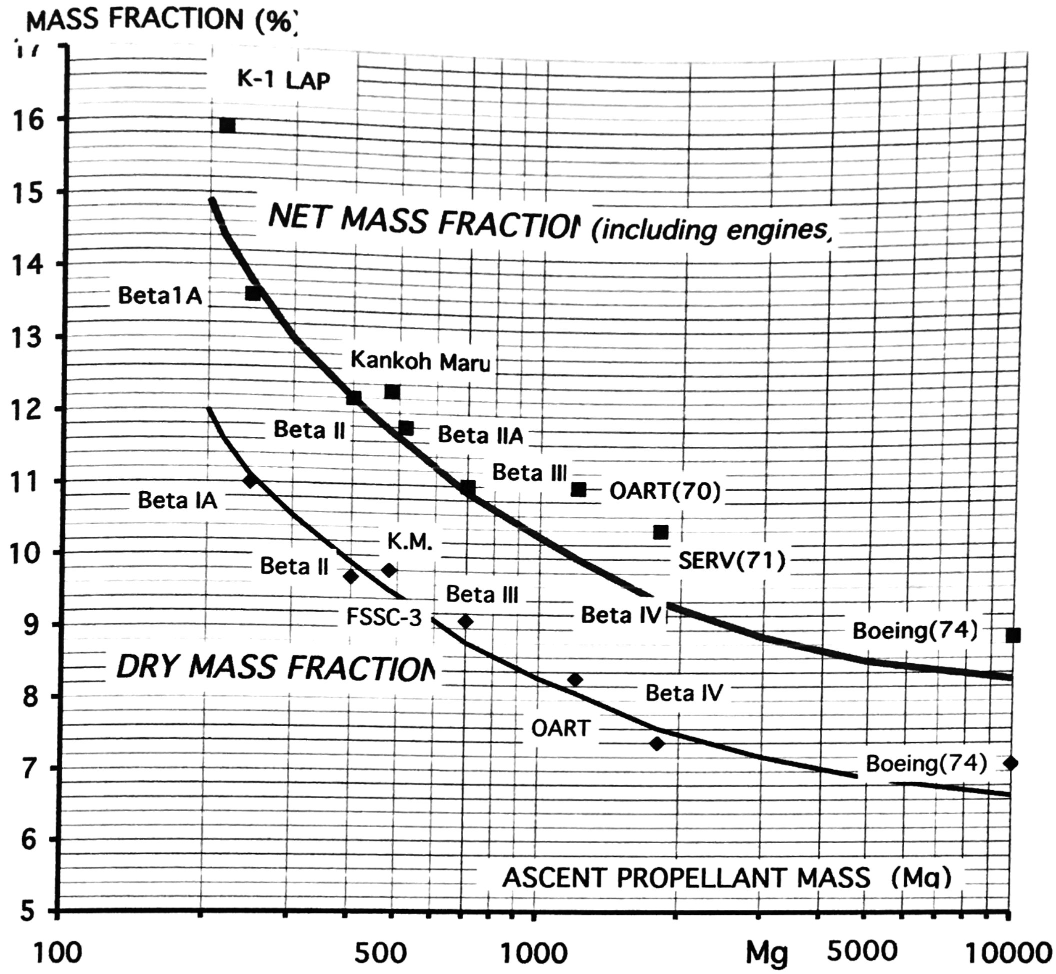


FIG. 2-33: **Net Mass Fraction (NMF) and Dry Mass Fraction (DMF) of Ballistic Reusable Vehicles with LH2/LOX** (incl. engines)

In case of a reusable SSTO vehicle about twice the propellant mass is required for the same payload compared to an expendable vehicle. From FIG. 2-33 it can be concluded that the larger propellant mass and the inherently higher specific mass results in about 180 % higher net mass of the reusable vehicle (total Net Mass, including the engines). Part of this higher weight is structure mass, due to the required higher safety factors for RLVs compared to ELVs, as shown in TABLE 2-I.

With liquid hydrogen as fuel single-stage (SSTO) vehicles to LEO are feasible, and most concepts are using this baseline. With storable propellants and their lower performance a two-stage configuration is required for LEO missions, as conceived for the KISTLER K-1 Vehicle (FIG. 2-64). In this case conventional existing technology was baselined (i.e. parachute recovery of the stages) with accordingly reduced development cost. Since the vehicle development has not been completed, no final development cost figures are available as reference data. As a purely commercial venture the development cost should be in the order of 25 to 30 % of BAU cost for a government contract.

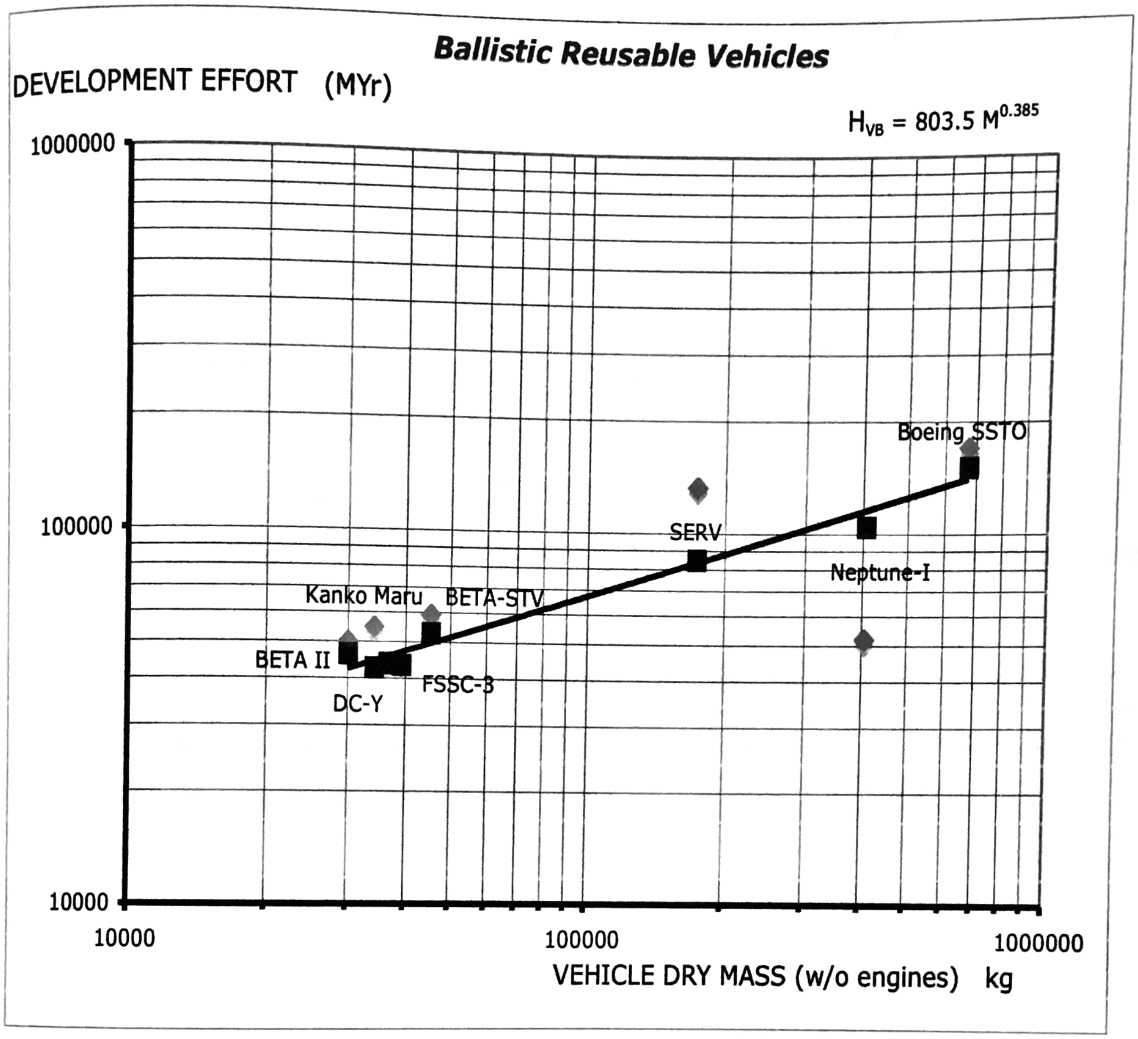


FIG. 2-32: **Basic CER for Ballistic Reusable Launch Vehicles**

For the establishment of a preliminary CER the detailed development cost estimates of eight ballistic reusable project studies (adding a 15 % cost margin according to the TransCost ground rules) resulted in a plausible result, in between the trends of winged orbital vehicles and advanced aircraft. In addition, the result of a detailed cost analysis using the subsystem-based PRICE-H computer model for the FSSC-3 concept does fit quite well as shown in FIG. 2-32. The costs for the large vehicles are somewhat questionable since no experience with such large-size vehicles exists.

The preliminary CER for the development cost of Ballistic Reusable Vehicles (excluding engine mass and development cost) is accordingly

$$\boxed{H_{VB} = 803.5\ M^{0.385}\ f_1\ f_2\ f_3} \qquad \text{MYr}$$

The f_2 -definition is the same as for expendable vehicles:

$$f_2 = k_{ref} / k_{eff}$$

with k_{ref} = average reference net mass fraction
k_{eff} = effective net mass fraction of the vehicle concept or the existing vehicle.

The f_2-Factor is related to the Net Mass Fraction (NMF) in comparison to the nominal reference value as shown in FIG. 2-33.

2.45 Winged Orbital Rocket Vehicles

This group of space vehicles comprises all rocket-propelled winged vehicles with orbital capability (i.e. designed for re-entry) and horizontal landing on a airfield. Winged re-entry vehicles such as the Shuttle ORBITER, the Russian BURAN and the European HERMES and the Japanese HOPE projects do *NOT* belong to this group although the external configuration looks similar to rocket-propelled launch vehicles. The difference is the lack of an essential propulsion capability (propellant mass) and - on the other hand - excluding longer-term orbital experimentation capability.

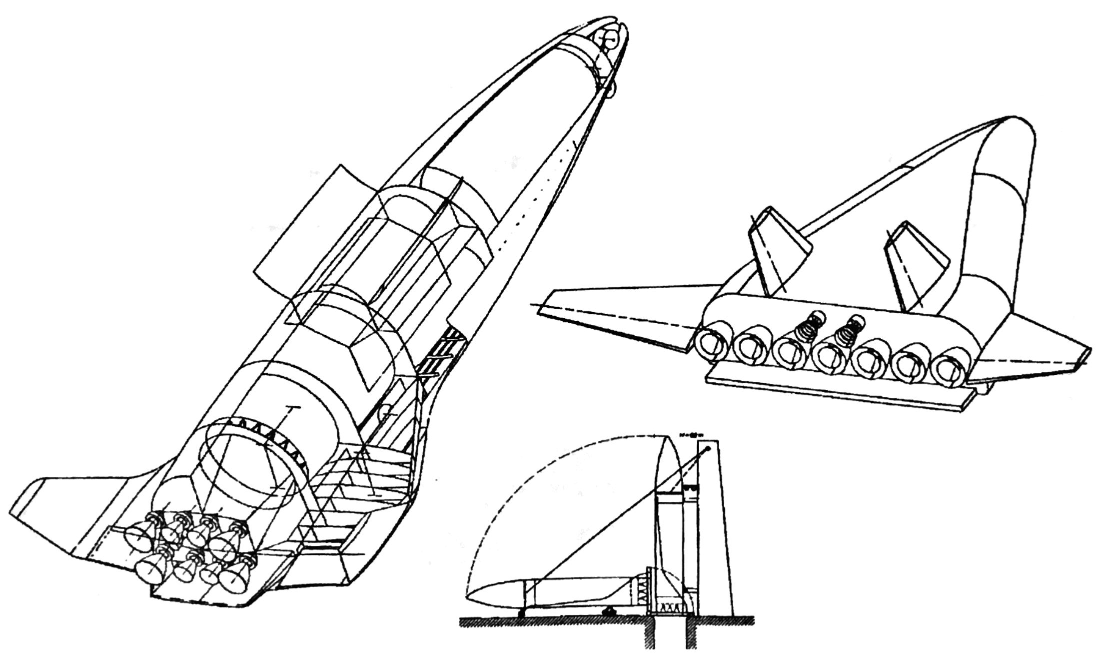


FIG. 2-35: **Winged SSTO Launch Vehicle Configurations**

This type of vehicles is covered by „Crewed Space Systems", chapter 2.49. The following different vehicle concept options are covered by the preliminary CER conceived for such vehicles:

(1) SSTO's with Vertical Take-off (VTO-HL)

The preferred vehicle configuration as studied by different companies and institutions is the conventional wing-body design as shown in FIG. 2-35 (left). The other option shown is a lifting-body configuration similar to that one of the X-33 test vehicle which had been conceived by Lockheed Martin's Skunk Works in California for the „Venture Star" SSTO. The X-33 vehicle should have achieved a speed of Mach 12-13 with a linear aerospike rocket engine. The development cost were estimated to 1.4 Billion $, however, the development was terminated due to cost and schedule increases (cf. ch. 2.63).

(2) SSTO's with Horizontal Take-off (HTOL) and Launch Assist System

To possibilities exist for a horizontal launch assist system: (a) launch sled, with or without its own propulsion, and (b) a carrier aircraft.

The sled option is very limited in the launch azimuth flexibility. Without own propulsion and high separation speed (reqiring a long track) there is no real performance advantage: The reduction of thrust level for a horizontal launch (and thus engine mass) is balanced by a higher delta-V demand due to the turn maneuver(s) and longer flight in the denser atmosphere, increasing the total vehicle mass. The turn maneuver also leads to high wing loads with impact on the dry mass. There seems to exist a vehicle size limit to some 600 Mg GLOW for practical reasons.

The other option of a carrier aircraft provides ideal launch azimuth conditions but the total SSTO vehicle mass is limited presently to some 250 Mg, the maximum cargo capability of the world's largest aircraft, the Antonow AN-225 (FIG. 2-36). This vehicle size, however, does not provide a satisfactory payload and mission capability.

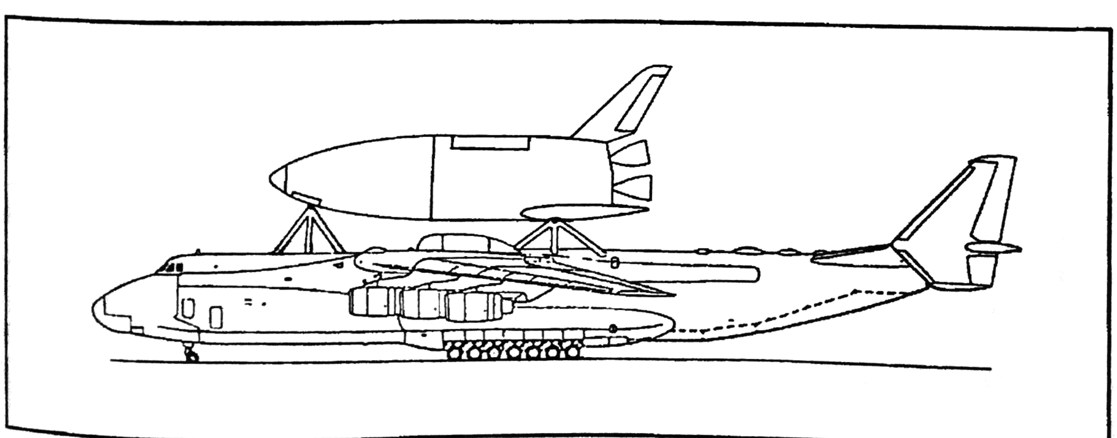


FIG. 2-36: **Aircraft-launched RLV** (BAe Interim HOTOL Study "Hotrock"

(3) Winged Second Stage of Two-Stage RLVs

The first stage of the TSTO Vehicle can either be expendable, a ballistic RLV or a Flyback Vehicle (chapter 2.46). The winged second stage vehicle can also be combined with a high-speed aircraft, as shown by the SÄNGER Project (FIG. 2-43). Two-stage launch vehicles with *vertical* launch mode always require rocket propulsion. There are two concept options: Parallel staging or Tandem staging (FIG.

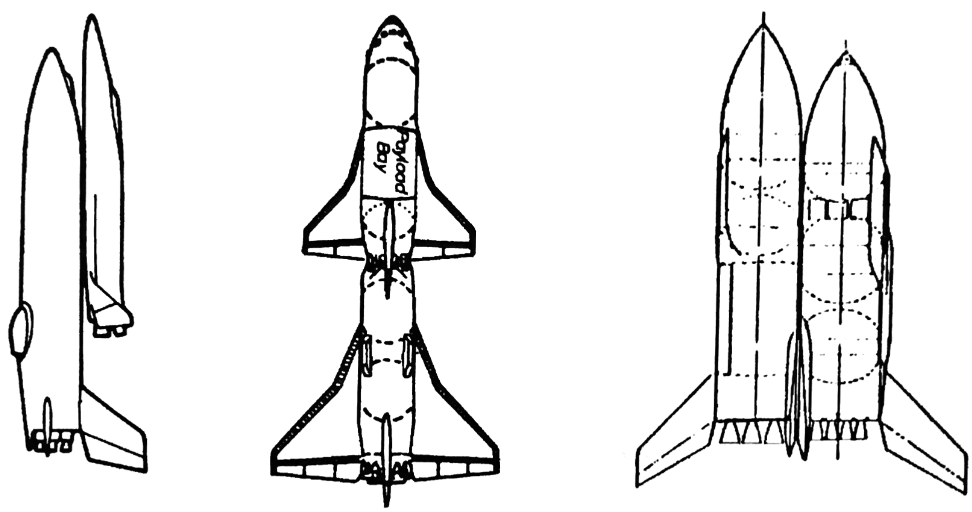


FIG. 2-37: **Two-stage Winged Launch Vehicles with Parallel and Tandem Arrangement of the Stages**

2-37). A special variant are the „bimese" or „triamese" configurations (not shown) where the first and second stages have the same external geometry but are different internally due to the cargo bay in the upper stage and the larger propellant tanks in the first stage. Whether this special concept provides a cost advantage is an open issue since the performance is degraded compared to optimized stage sizes. Also propellant transfer during flight from the first stage to the second stage vehicle is required to make it an efficient system.

In comparison to ballistic RLVs winged vehicles exhibit not only a definitely higher net mass - due to the additional weight of wings, aerodynamic control surfaces and the required actuator / power systems - but also a much greater sensitivity vs. vehicle size as shown in FIG. 2-38. This trend has been clearly confirmed by more than a dozen detailed project studies (cf. FIGs. 2-39 and -40). FIG. 2-39 shows the net mass values of winged rocket vehicle projects vs. propellant mass plus payload: in this case - different to expendable vehicles - the payload cargo bay is located in the center of the vehicle and it influences the vehicle size, and accordingly the mass and costs. In addition, there is a diffence between automated and piloted vehicles. All vehicles are using LOX/ LH_2 propellants.

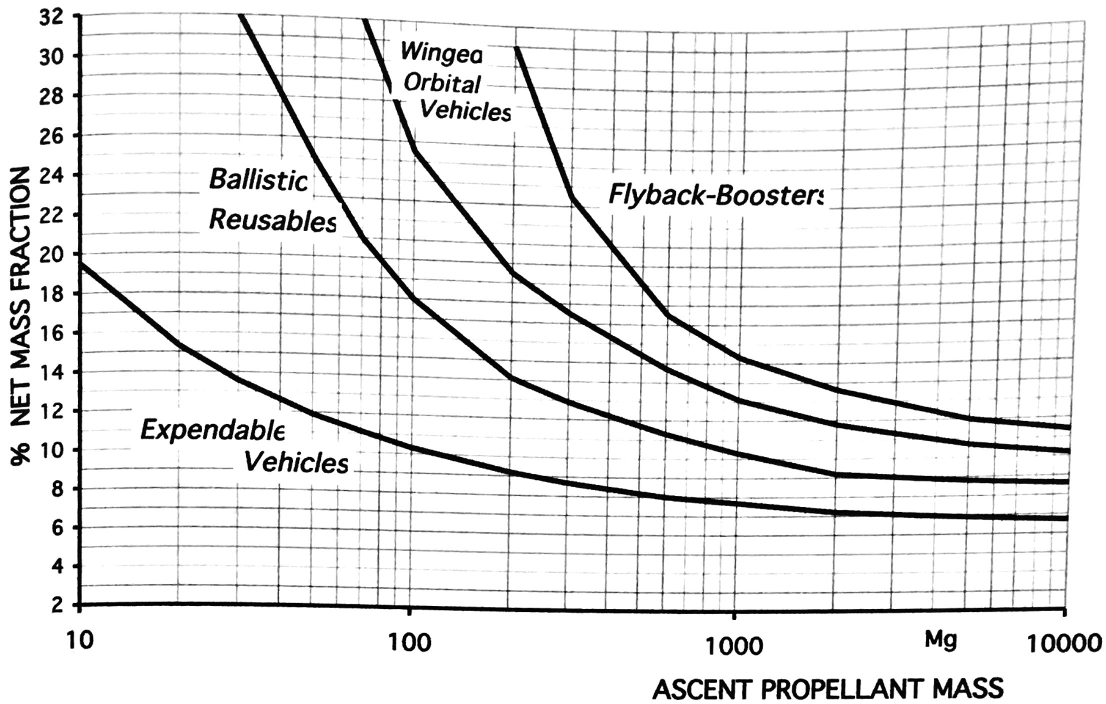


FIG. 2-38: **Comparison of Net Mass Fraction Trends vs.Vehicle Size (LOX/LH_2-Vehicles, unmanned)**

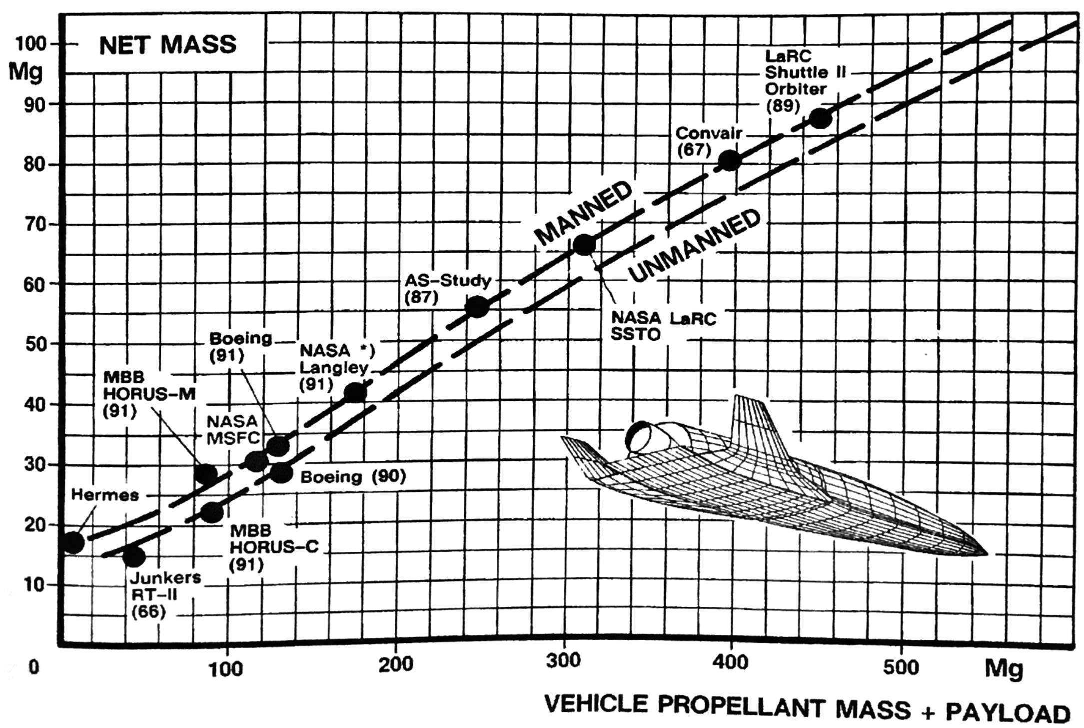


FIG. 2-39: **Net Mass of Winged RLVs (Rocket Propulsion) with and without Crew Cabin**

FIG. 2-40 provides a more detailed overview about the Net Mass Fractions of available detailed project studies on winged orbital RLVs. It becomes evident how strongly the NMF depends on the vehicle size - in terms of propellant mass. The curve represents a median line; variations are due to the level of technology assumed, the mass margin included, etc. The difference of the net mass to the vehicle dry mass is about 12 % (10 to 14%), i.e. only half as much as for ballistic RLVs, the overall NMF values are, however, 30 to 40 % higher.

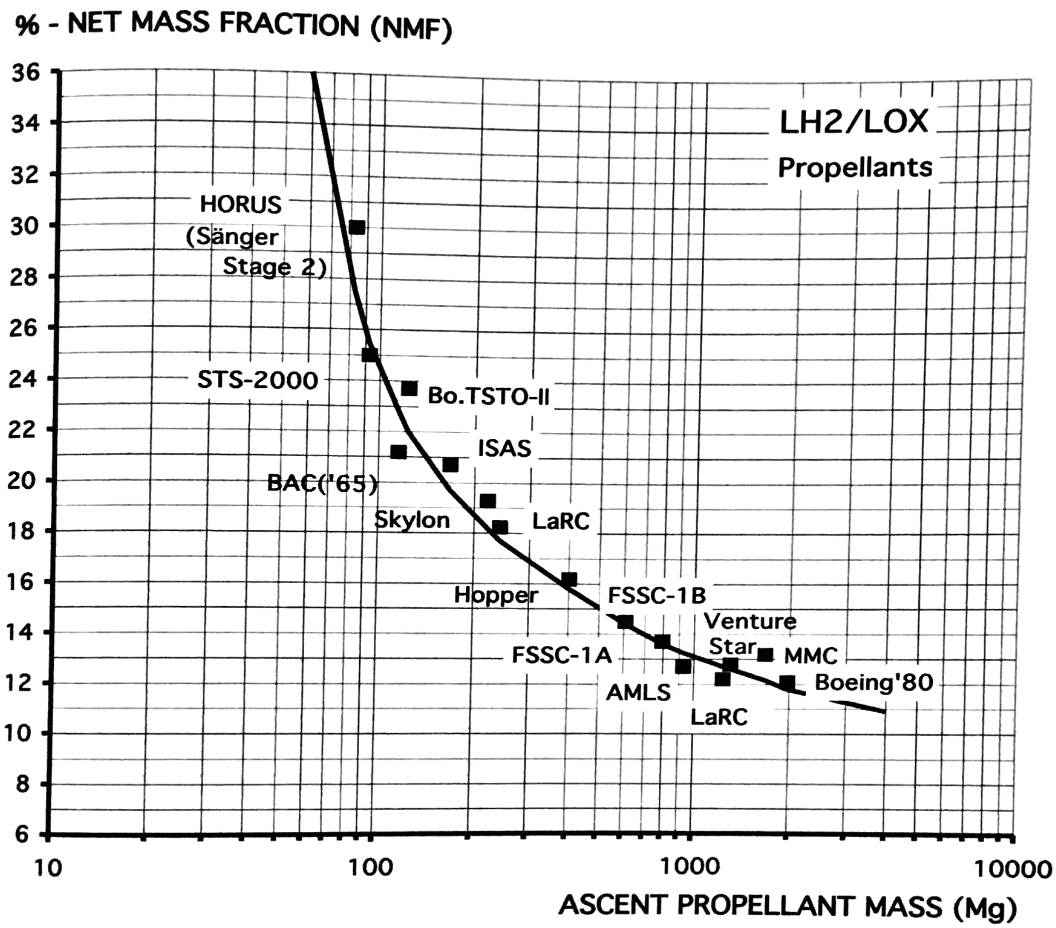


FIG. 2-40: **Net Mass Fraction of Winged Orbital Vehicles vs. Propellant Mass** (Unmanned Vehicles with Internal Cargo Bay, incl.engines)

The establishment of a CER for Winged Orbital RLVs is difficult since there are no realized projects yet, except the somewhat different Shuttle Orbiter as the first realized vehicle of this kind. The first project was the DYNA-SOAR Vehicle (1960). The development cost estimate at that time was much too low due to the lack of experience. The cost growth then led to the termination of the project in 1963. The HL-20 reference point resulted from a NASA Langley Study with Lockheed Skunk Works about development as usual (BAU) and a commercial development (ref. 46).

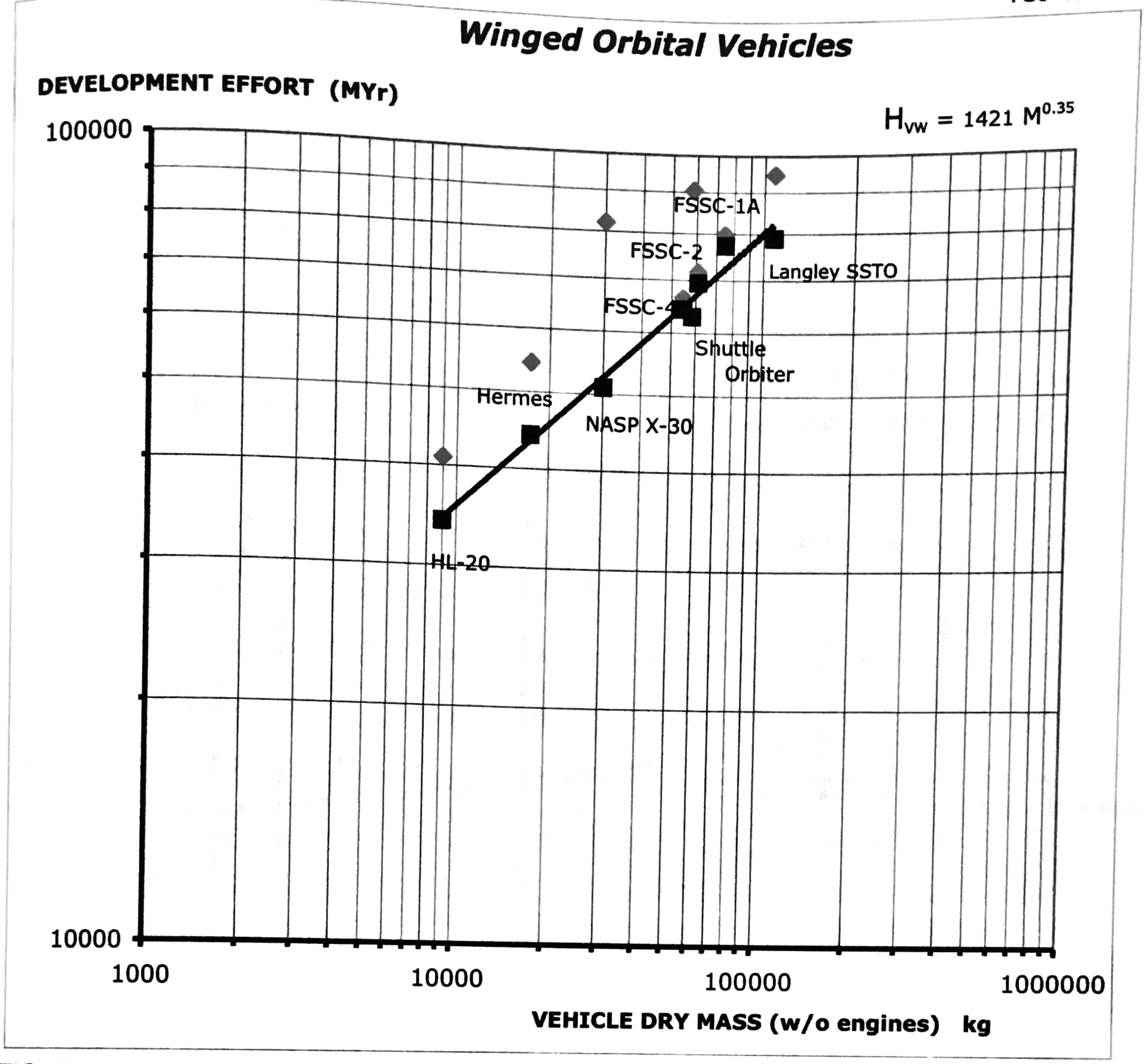


FIG. 2-41: **Preliminary Basic CER for Winged Orbital RLVs with 8 Reference Projects**

The HERMES development cost are based on the detailed industrial proposal for Phase C/D, plus 15 % for unforeseen problems and delays, as discussed in chapter 2.11. The FSSC-values resulted from detailed vehicle design studies and a subsystem-based cost estimate (plus 15%). With these reference points a preliminary CER has be established as follows :

$$\boxed{H_{VW} = 1421 \; M^{0.35} \; f_1 \; f_2 \; f_3} \qquad \text{MYr}$$

The maximum deviation, resp. range of +/ - 20% from the basic CER could only be achieved by application of a Technical Quality Factor.

Technical Quality Factor f2 :

The Technical Quality Factor is related to the vehicle net mass since this is the main cost driver for winged vehicles. The median net mass fraction reference of FIG. 2-40 serves as reference. In this case the vehicle net mass includes the engine(s), and the related mass is propellant plus payload:

$$\text{NMF } \varepsilon^* = M_N / (M_P + M_{P/L}) .$$

The Technical Quality Factor is defined as

$$f_2 = (\varepsilon^* / \varepsilon)^2$$

with ε^* as a nominal value taken from FIG. 2-40, and ε being the NMF-value of the project under evaluation (automated vehices without crew cabin/ cockpit).

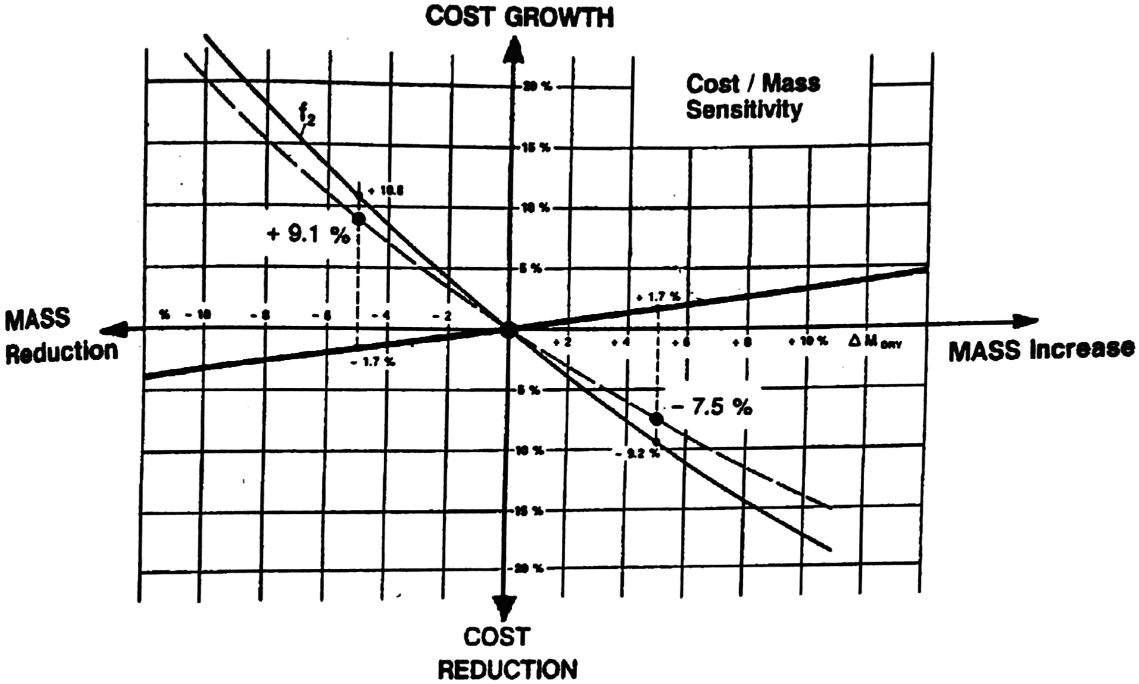


FIG. 2-42: **Quality Factor f_2 Impact on Cost-to-Mass Relationship**

This definition takes into account the effect of using high-tech materials and procedures which reduce the net mass but increase cost. This is illustrated in FIG. 2-42: The diagram shows that for a given vehicle design the (assumed) mass reduction of 5 % with the basic CER would lead to a cost reduction of some 1.7 %. With the application of the f_2-Factor, however, the cost will not decrease but increase by some 9.1 %, as expected. The same applies if - by simpler technology - the weight is allowed to go up by some 5 %, then a cost reduction of some 7.5% can be expected. This is valid for a vehicle with the same size (same propellant mass).

Crewed vehicles require a larger dry mass compared to unmanned vehicles. This is due to the additionally required pressurized cabin, the life support and crew equipment, also resulting in higher power demand and increased power subsystem mass. These requirements result either in a larger vehicle (in case the same payload capability is required as for the unmanned system), or to a 3.5 to 5.5 Mg payload reduction for a given vehicle with a crew of 2 to 5 persons, including pilot(s). The alternative concept of avoiding the integrated cockpit approach was conceived in 1990 (ref. 29) for the SÄNGER-HORUS orbital vehicle and also adopted by NASA in 1994 for the SHUTTLE follow-on vehicle studies (ref. 59). Passenger accomodation would be realized by an autonomous pressurized module which is placed into the cargo bay.

2.46 HTO First Stage Vehicles, Advanced Aircraft and Aerospaceplanes

2.461 Vehicle Characteristics

This group of vehicles comprises advanced (supersonic) aircraft as well as launch vehicle first stages with airbreathing propulsion (turbojet engines) and take-off from a conventional runway. The second stage is carried on top (Sänger concept, with LH_2

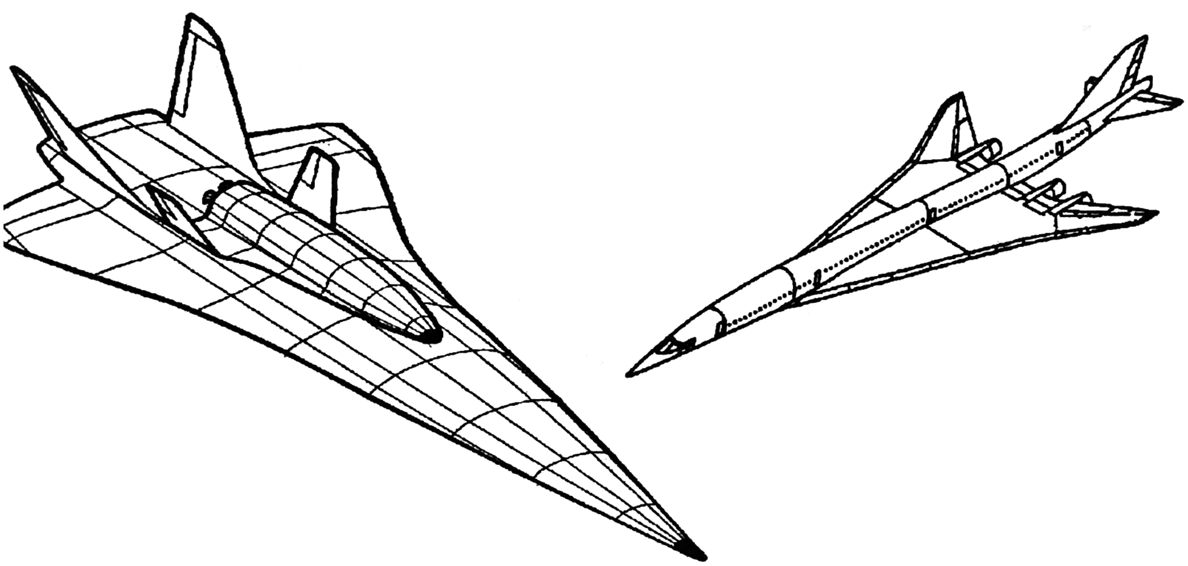


FIG. 2-43a: **SÄNGER II TSTO Concept (Mach 6.8 HST) and Supersonic Transport Aircraft (McDonell Douglas, Mach 2.8 SST)**

propellant) or underneath the fuselage (Boeing BETA II concept, using Kerosene). FIG. 2-43b depicts these representative vehicles which provide great flexibility for launch from a variety of sites, for cruise to the appropriate latitude, and for launching the upper stage in any azimut direction.

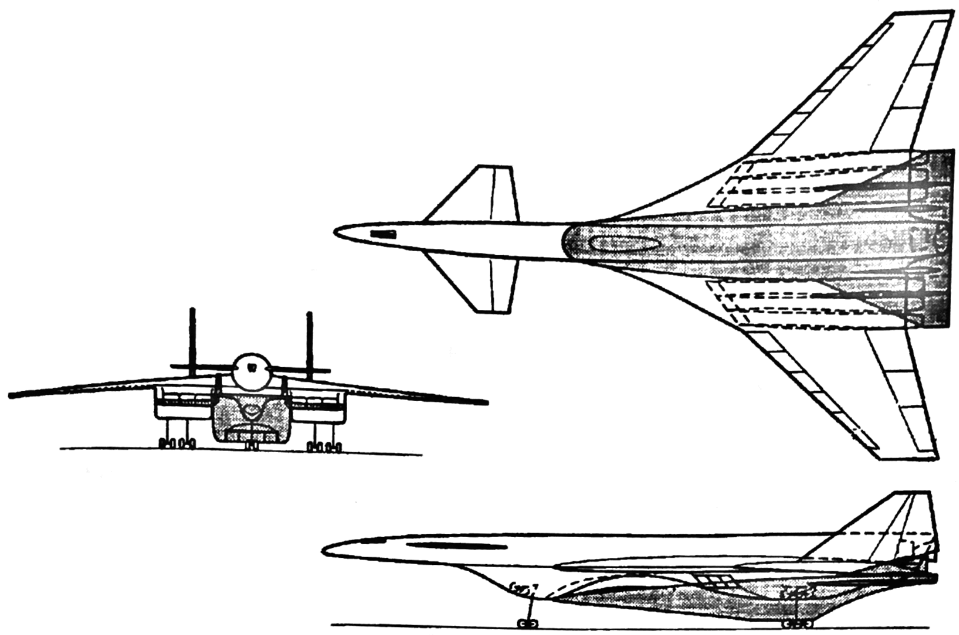


FIG. 2-43b: **Boeing BETA II TSTO (Mach 3.5 SST) Space Launch Vehicle Concept**

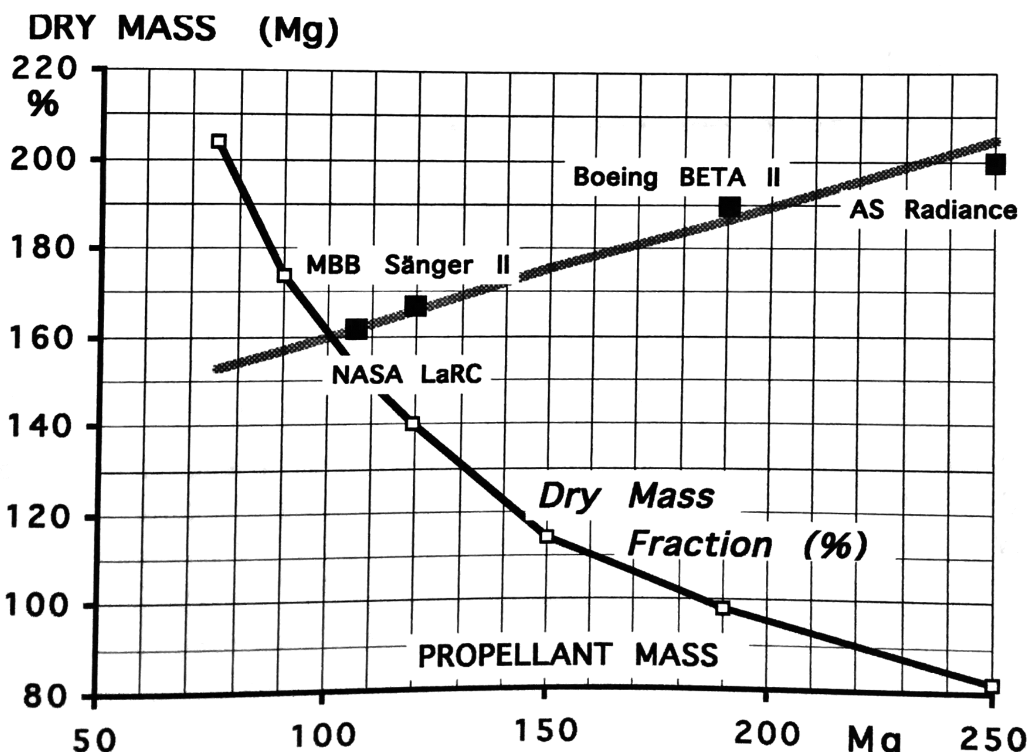


FIG. 2-44 : **Dry Mass (Mg) and Dry Mass Fraction (%) of HTO First Stage Vehicles with Airbreathing Propulsion**

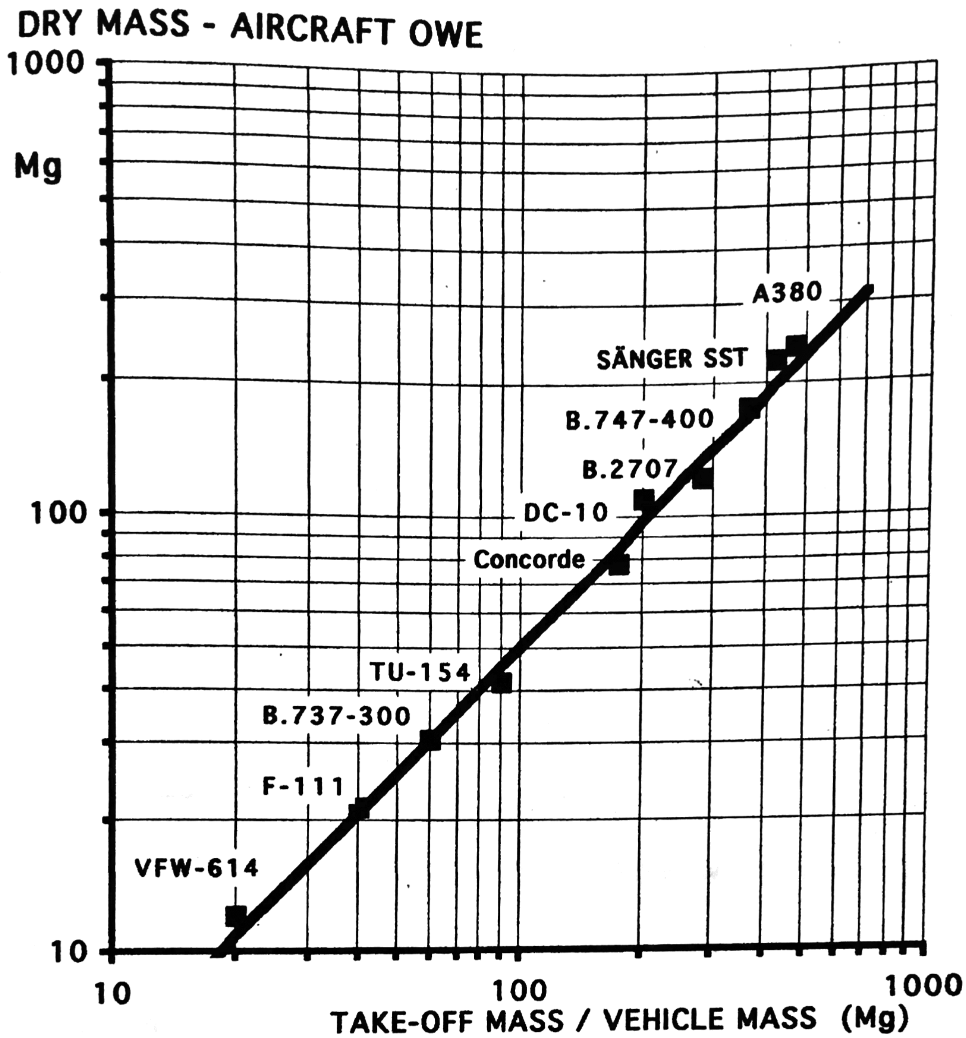


FIG. 2-45: **Aircraft Dry Mass (OWE) vs. Take-off Mass**

2.462 Cost Estimation and Technical Quality Factor

Development costs of nine advanced aircraft projects have been retrieved which provide the basis for the developments cost CER as shown in FIG. 2-46. This CER for Advanced Aircraft should also be applicable for future HTO First Stage Vehicles with airbreathing propulsion :

$$\boxed{H_{VA} = 2880\ M^{0.241}\ f_1\ f_2\ f_3} \qquad \text{MYr}$$

Two commercial subsonic airplane development projects exhibit the same cost / mass trend but on a considerably lower level (one third) compared to the mostly

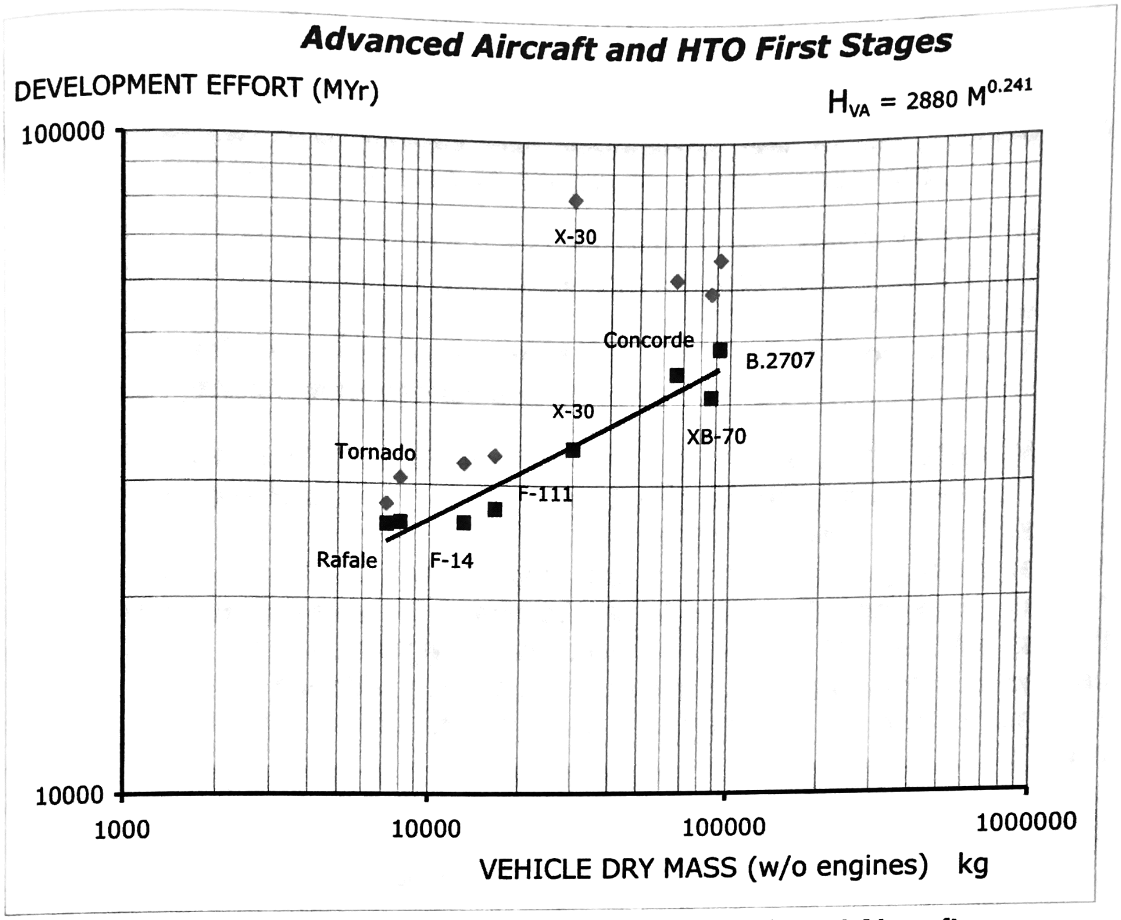


FIG. 2-46: **Reference Projects and Basic CER for High-Speed Aircraft**

military government-funded projects of Mach 2+ aircraft. The much lower development effort for commercial projects to a certain extent is caused by the existing experience for subsonic aircraft but to a larger part due to the independent industrial development approach without detailed external requirements and control procedures. It is shown later (chapter 2.6) that also for space projects a development cost reduction by a factor three to four seems feasible by the application of commercial - industrial development approaches.

Technical Quality Factor (TQF):
Although only few reference data are available, the first approach for a TQF for high-speed winged vehicles is the application of the maximum speed capability. An f_2-Factor has been derived with a Mach exponent of 0.15 :

$$f_2 = Ma^{0,15} .$$

This TQF allows to make the Shuttle Orbiter with Mach 25 (at re-entry) to fit with this CER. The TQF must be considered as preliminary only until confirmed or modified by future high-speed aircraft projects.

2.463 AerospacePlanes (SSTO Launch Vehicles with Combined Turbojet/ Ramjet / Scramjet-Propulsion)

These aircraft-like HTO vehicles with airbreathing and rocket propulsion are sometimes described as the ultimate concept for space transportation.

The US NASP Program (1986 to 93) was dedicated to this goal conceiving the X-30 demonstrator vehicle (no payload, no operational mission requirements). Some 2.5 Billion $ were spent in the definition phase while the X-30 vehicle mass increased from initially 23 to 160 Mg - with further trend upwards. The total cost of the 15-year development program were estimated to some 15 Billion $ ('91).

Another study was performed in Europe at that time: the ESA-WLC Study (Winged Launcher Configurations) by MBB/Dasa and BAe (ref. 91). An *operational* SSTO vehicle with airbreathing combined Rocket-Ramjet-Scramjet-Rocket propulsion was studied in detail (FIG. 2-47). The results of this up to now probably most realistic study were not encouraging: The required use of liquid hydrogen results in large propellant tanks and, accordingly, in a large vehicle size: 80 m long, 670 Mg take-off mass, the maximum practible size for an HTO vehicle. The calculated LEO payload as verified by an optimized ascent trajectory calculation with the ALTOS computer program was 4 Mg. This assumed already the application of advanced materials which were only in the laboratory development stage. The vehicle dry mass was calculated with 152 Mg, including 11 % margin. The high dry mass and the low payload makes this vehicle concept extremely sensitive to any mass change (weight growth).

The high dry mass results from the vehicle size, the thermal protection and the complex propulsion system, requiring a highly variable air intake (plus protection at re-entry !). From the Isp-values airbreathing propulsion looks very attractive, but the

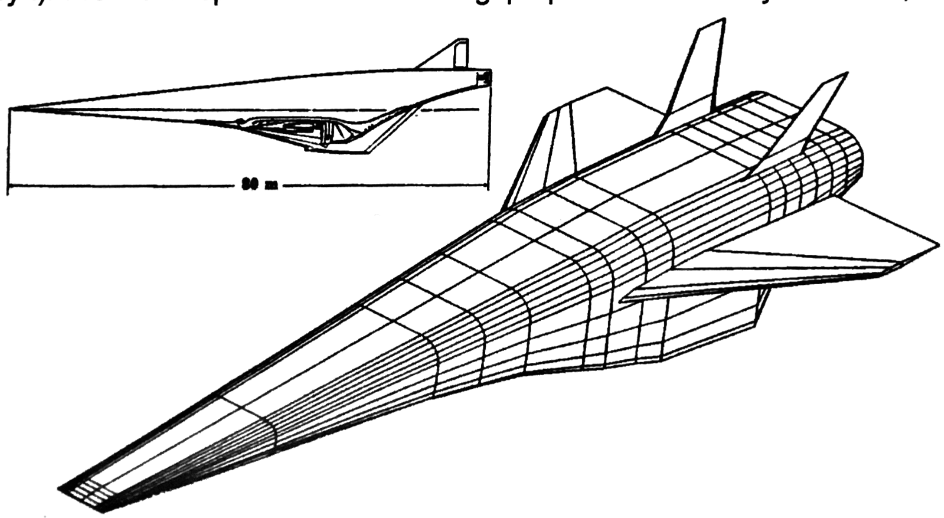


FIG. 2-47 : **Winged SSTO-Vehicle with Combined Propulsion** (ESA-WLC Operational Aerospace Plane Concept) long ascent trajectory within the atmosphere also means drag losses, increasing the total delta-V requirement. The key problem, however, is the large vehicle mass to be placed into orbit: Since the dry mass is the most important cost driver, it is doubtful that such a vehicle -even if all technological problems could be solved - is competitive economically. The NMF of the above-mentioned SSTO Vehicle would be 32.4 % at 503 Mg propellant mass assuming the application of advanced technologies. A special problem in addition is the inherent integration of propulsion and vehicle structure with thermal control. A separate cost assessment is, therefore, not feasible, given by example, the large and complex air intake.

Using the CER for advanced aircraft development for the Mach 25 SSTO-vehicle with 152 000 kg dry mass, the result is some 140 000 MYr or 32 Billion $ (year 2003 value). There are, however, not only the extraordinary development cost but it has been shown in ref. 55 that the specific transportation cost to LEO would be about 50 % higher for such an SSTO compared to a two-stage winged launch vehicle with an airbreathing first stage (MBB SÄNGER and Boeing BETA II Concepts). Similar results are discussed in ref. 92.

2.47 VTO First Stage-Fly-back Rocket Vehicles

The winged first stage concept with return to the launch site by means of turbojet engines is a preferred option for TSTO launch vehicles. In this case aircraft requirements are added to the VTO launch vehicle requirements which makes it a complex and expensive vehicle concept. These vehicles reach a velocity of Mach 5 to 9 at stage separation before cruising back to the launch site using turbojet

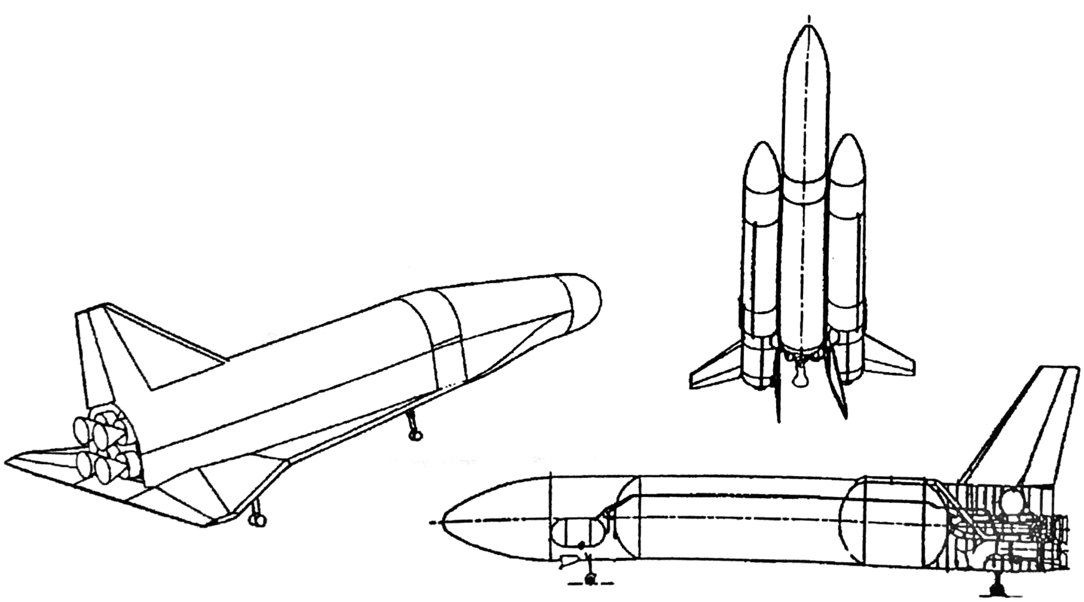


FIG. 2-48: **Winged Rocket Boosters with Fly-back Turbojet Engines (LFBBs)**

engines. Only in case of a low separation speed (Mach 3) a glide-back flight to the launch site is feasible without auxiliary air-breathing engines. FIG. 2-48 shows a typical fly-back vehicle design which can be used in single or twin configuration to launch a reusable or expendable second stage.

The NMF-chart for this vehicle type, assembled from a number of detailed project studies, is shown in FIG. 2-49. Compared to Orbital Vehicles (FIG. 2-40) the net mass values are substantially higher, especially for vehicles with less than 800 Mg propellant mass (+ 25 to 45 %). This is due to the turbojet engines and the fuel required for the return flight, but also caused by the more aircraft-specific design requirements. The difference between net mass and dry mass is some 22 % (21 to 23 %) taking into account the flyback propellants, residuals and reserve

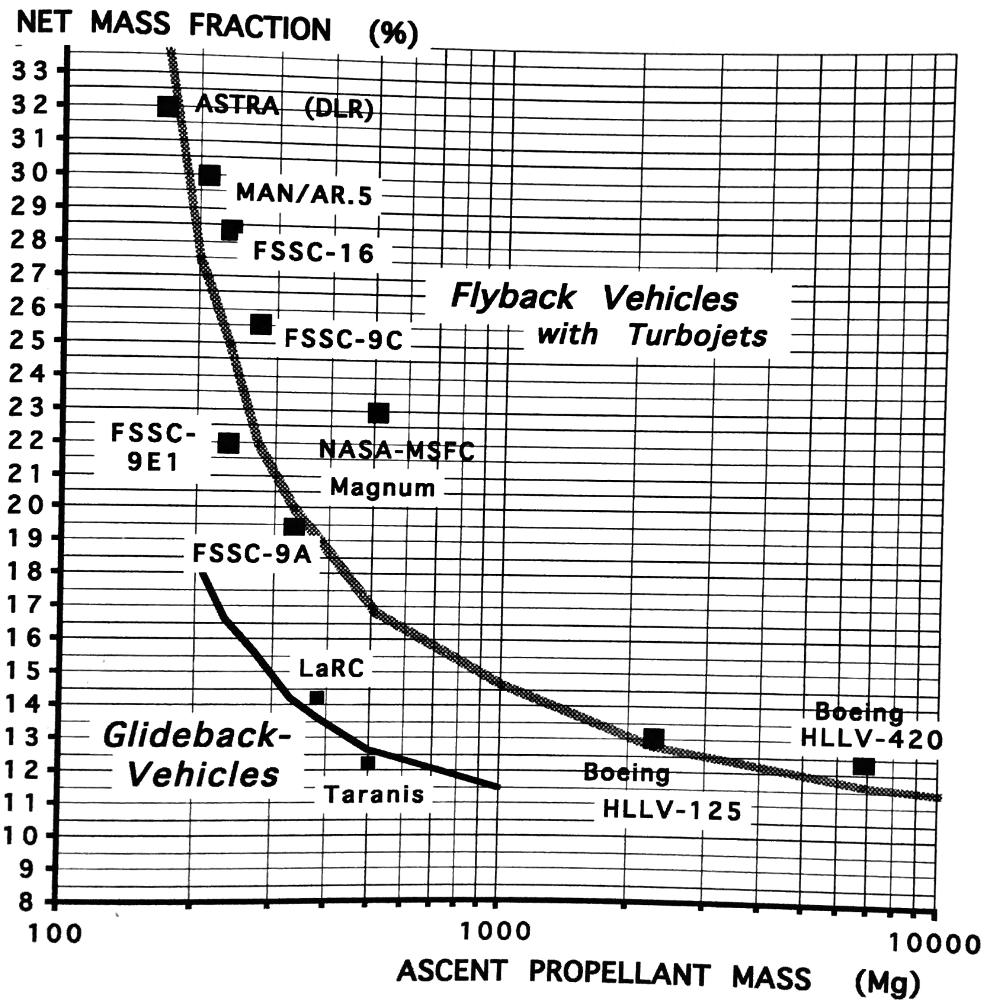


FIG. 2-49: **Net Mass Fraction of Winged VTO Rocket-propelled First Stage Vehicles (Fly-back Boosters)**

propellants. FIG. 2-49 also shows a net mass curve for „Glideback-Vehicles". Due to a low separation speed (i.e. Mach 3) the relatively short range allows for avoiding additional turbojets and fuel for either flying back to the launch site or to land at another remote airport. Accordingly the difference between net mass and dry mass in this case is only some 10 %. This value and accordingly the NMF is influenced by the separation speed and the required amount of fly-back propellants.

The main difference is that transport aircraft need a large internal payload volume while the Fly-back Booster carry the payload externally. The relatively heavy turbojet engines are dimensioned for the take-off mass while in case of Fly-back Boosters only small engines are are required for the cruise flight of the empty vehicle. Also the wings need only to be sized for this case.

For conventional aircraft and HTOL-vehicles the maximum flight velocity (Mach-number) does not seem to have a major impact on the dry mass, as the data. points for subsonic aircraft, the CONCORDE (Mach 2.1), the SST (Mach 2.7) and the SÄNGER EHTV Version (Mach 4.4) in FIG. 2-45 suggest.

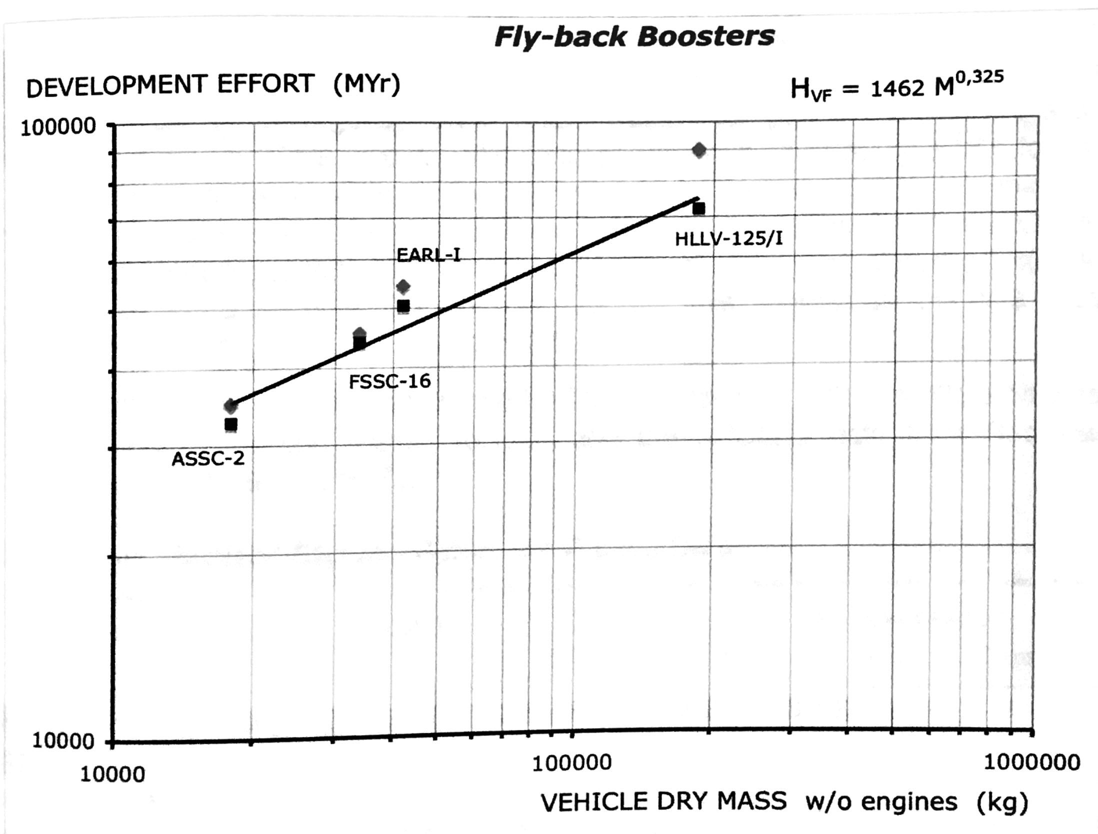


FIG. 2-50: **CER for Flyback Boosters (LFBBs) based on four reference projects**

Since no fly-back vehicles of this type have yet been realized only a preliminary CER approach is feasible as an interpolation between the basic CER of Winged Orbital Vehicles (FIG . 2-41) and of High-speed Aircraft (FIG. 2-46). The result is plausible since Fly-back vehicles are more complex than aircraft but less demanding than Orbital Vehicles with the re-entry maneuver including the thermal loads. FIG. 2-50 shows this approach with the preliminary CER for the development cost set between the other two basic CERs as mentioned. The resulting CER for Flyback-Boosters has been defined as

$$\boxed{\mathbf{H}_{VF} = 1462 \ M^{0.325} \ f_1 \ f_3} \quad \text{MYr}$$

A certain confirmation are the development cost values derived from the detailed project studies on EARL , FSSC-16 and the ASTRA ASSC-2 LFBB projects with the subsystem-based PRICE-H Model. A Technical Quality Factor f_2 has not yet been defined. This requires some more project data points.

2.48 Crewed Ballistic Re-entry Capsules

Manned space activities started with ballistic capsules like VOSTOK, MERCURY, GEMINI and APOLLO. Such systems also became of renewed interest as a rescue vehicle or emergency crew return vehicle as part of the Space Station operations program. More recently, this concept has been selected by NASA for the ORION crew exploration vehicle. For this reason a special CER for manned ballistic capsules has been established. FIG. 2-51 shows the APOLLO CM (Command Module) as the most advanced project of this type. Together with the other well-known projects MERCURY and GEMINI this represents the data base for the CER with the values shown in TABLE 2-III.

The MERCURY data are taken from the NASA Publication SP-4201, the GEMINI data as published in Aviation Week, 16.1.67, and the APOLLO data from Rockwell Document SD-71-35-1 (June 72).

TABLE 2-II: **Ballistic Capsule Reference Project Data**

	DRY MASS Kg	CREW	DEVELOPMENT PERIOD	DEVELOPMENT COST (M$)	MISSION PERIOD = MYr	MISSION PERIOD Days
MERCURY	900 *)	1	58-63	262	9 300	2
GEMINI	2 000**)	2	62-65	450	15 000	15
APOLLO CM	5 380 *)	3	62-66	1 116	37 200	20
ORION CEV	7 800**)	6 (4)	06-13	(3 900)	(16 000)	20

*) without escape tower **) 3000 kg with Propulsion and Adapter

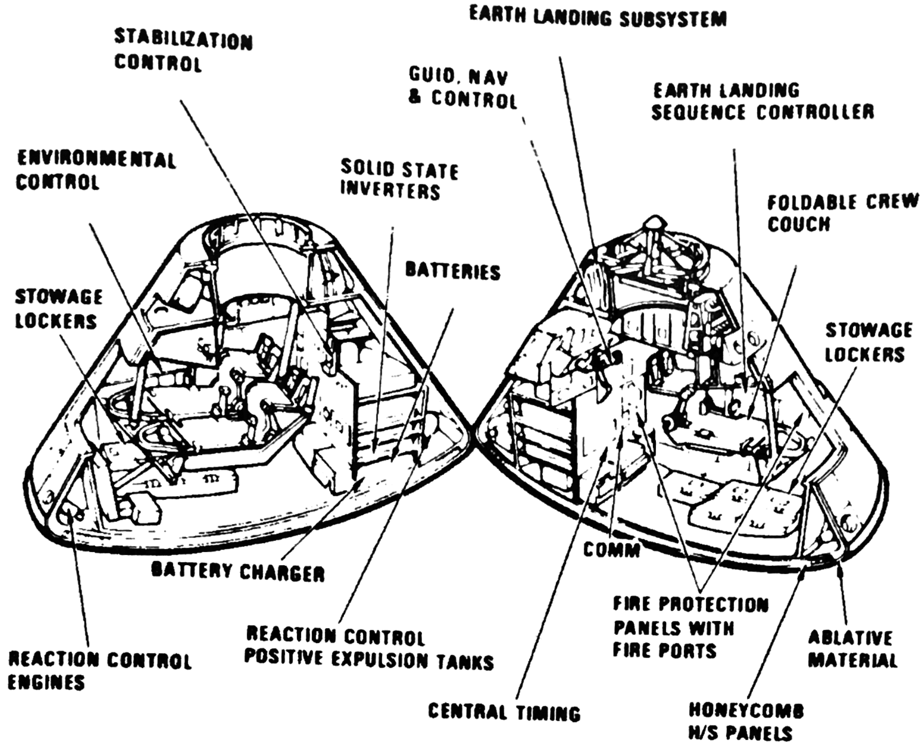


FIG. 2-51 : **The APOLLO CM Crewed Ballistic Capsule**

Using the MYr-values from Table 2-III would produce a good-looking CER curve, but a wrong one ! A cost trend would result being almost proportional to the dry mass, or showing constant specific cost (cost per kg). This, however, is unrealistic compared to all other space systems. All kinds of projects exhibit a reduction of specific cost vs. size (mass): the famous „scale factor", which has a very good explanation: many activities and subsystems are not or very little dependent on the vehicle size. Therefore, larger systems always must be more cost -effective.

The higher cost of the APOLLO CM compared to MERCURY and GEMINI can be traced to the new development of an advanced communication system for lunar distances (with an automatically Earth-pointing antenna dish), as well as to the life support and power supply systems which have been designed for a 60 man-days capability.

FIG. 2-52 shows the basic CER resulting from the application of the Quality Factor. The numerical value is

$$\boxed{H_{VC} = 436 \ M^{0.408} \ f_1 \ f_2 \ f_3} \quad \text{MYr}$$

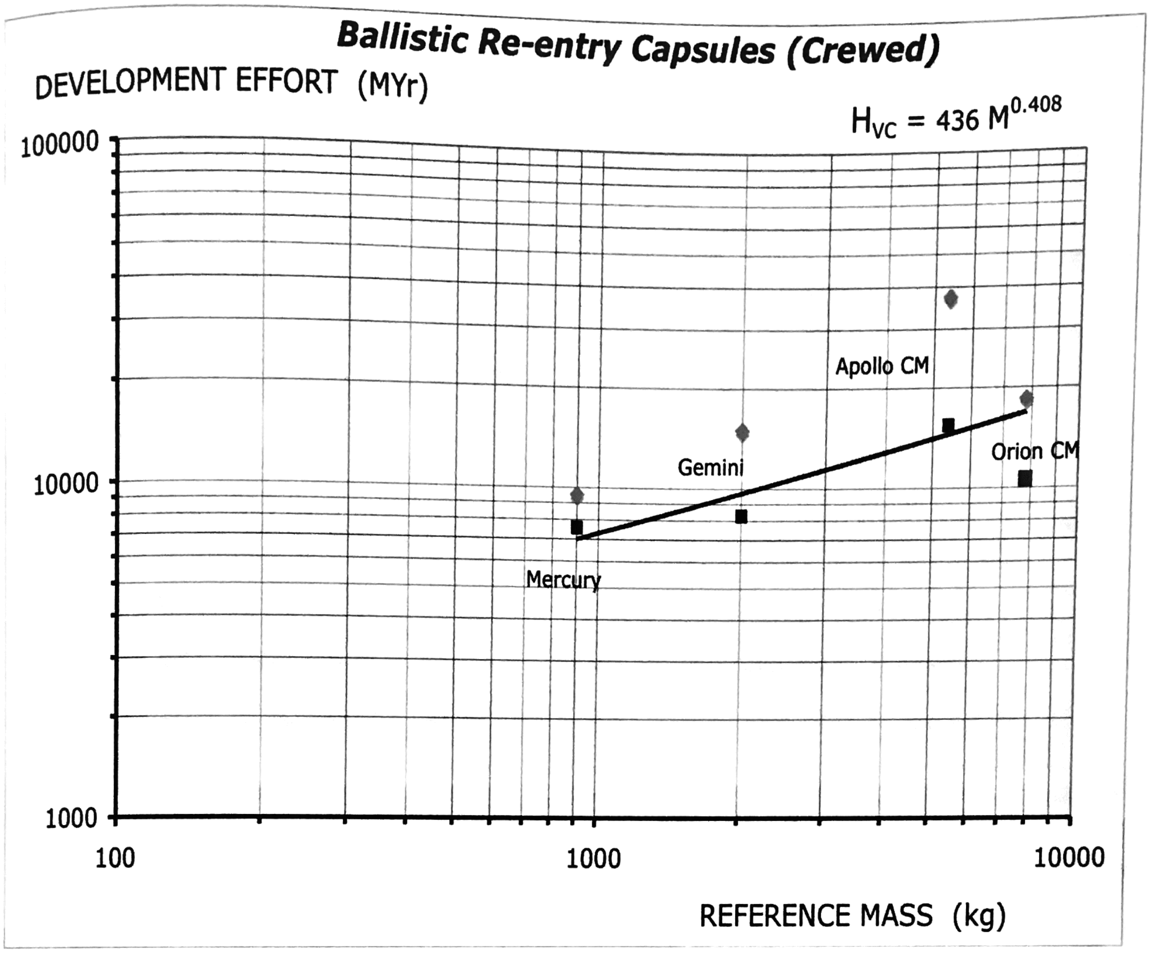


FIG. 2-52: **Reference Data and Resulting CER for Crewed Re-Entry Capsules**
(ORION value is proposal value; final cost tbd.)

Technical Quality Factor (TQF):

By trying different approaches the best result was achieved with a quality factor using the product of crew number and lifetime (days) with a power factor of 0.15:

$$f_2 = (N \cdot T_M)^{0.15}$$

with N = crew number, T = max. mission design lifetime

With this quality factor the following regression factors can be derived for the reference projects:

MERCURY	$(1 \times 2)^{0.15}$	= 1.11
GEMINI	$(2 \times 15)^{0.15}$	= 1.67
APOLLO	$(3 \times 20)^{0.15}$	= 1.85

Using these quality factor values as regression factors for the historic reference data is resulting in a much better CER with a realistic cost -vs.-mass trend and a cost level which is comparable to other projects and reference projets' range within +/ - 20 % of the basic CER.

The development cost for the ORION CEV in US$ assumed in 2006 are 3.5 times higher than for APOLLO, but translated into MYr they are only 50 %. The usual increase by 15 to 20 % must be added to the proposal value, but it still represents a great challenge to management and full utilization of existing experience.

2.49 Crewed Space Systems

This group of vehicles is of preliminary nature and comprises all those manned space systems which have n o t been covered by the previous CER's, such as

(a) Winged Re-entry Systems/ Crew Return and Rescue Systems

(b) Orbital Space Station Systems (Freeflyers, not attached Modules)

(c) Interorbital Spacecraft with Crew Cabin and (Lunar) Lander Vehicles

Only a limited number of quite different crewed space systerms have been realized but it seems that their common criteria of being systems for manned space operations allows to cover those by a common CER as illustrated in FIG. 2-56 by six reference projects.

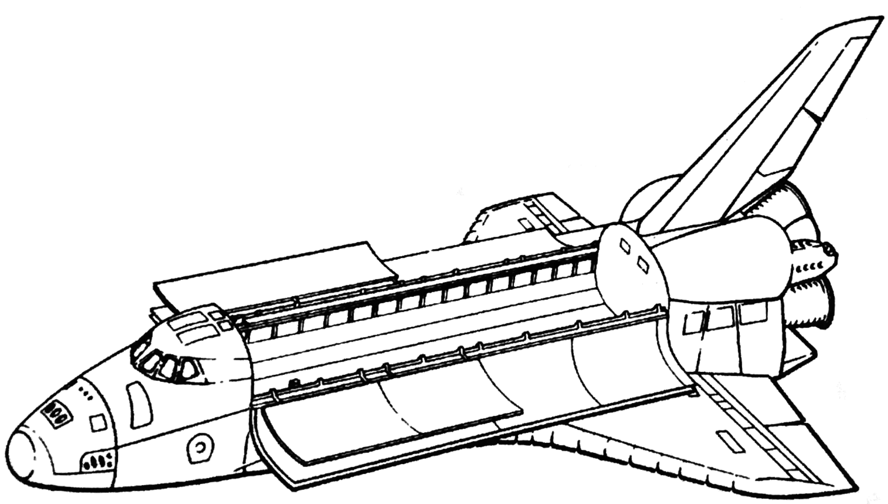


FIG. 2-53: **Shuttle ORBITER as Winged Re-entry Vehicles with Orbital Experimentation Capability**

These projects range from the complete GEMINI System (including equipment adapter and propulsion module) with 3000 kg to the International Space Station (ISS) with a total mass of 281 500 kg and development cost of more than 30 Billion USD (excluding transportation and operations cost).

Winged re-entry systems are one major systems group, with DYNASOAR (1963) as the first project of this kind. The main representatives are the Space Shuttle ORBITER (FIG. 2-53) and the Russian BURAN vehicles. Similar smaller-scale projects were the European HERMES Project and the Japanese HOPE Vehicle. Characteristic features are the lack of a major propulsive capability or propellant mass (= the difference to the group of Winged Orbital Vehicles) and instead an orbital laboratory and

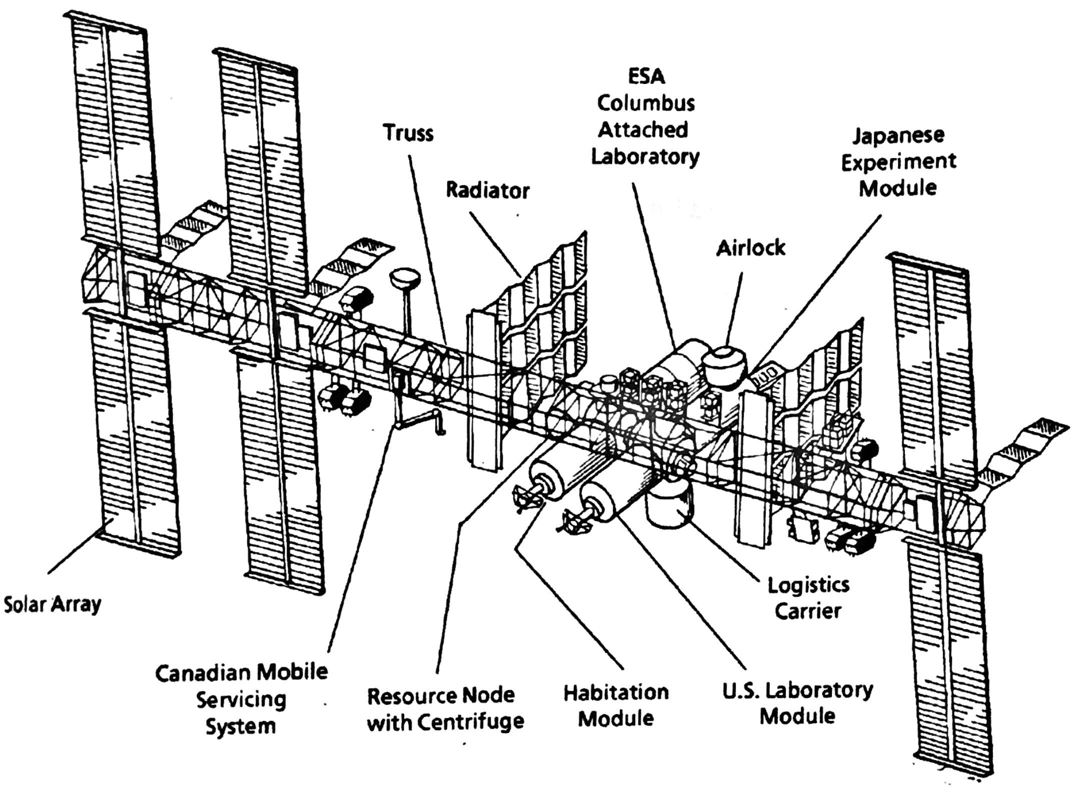


FIG. 2-54 : **Orbital Facility: The International Space Station (ISS)**

experimentation capability over one or two weeks in orbit. Other projects studied more recently are the so-called Crew Return Vehicle or Crew Rescue Vehicles (X-38).

The second type of crewed space systems are free-flying Orbital Facilities (complete systems, not attached Modules). The substantial development cost growth in case of the ISS (FIG. 2-54) to some 30 Billion USD was the typical result of repeated design changes during development, the initial management concept and a substantial schedule extension.

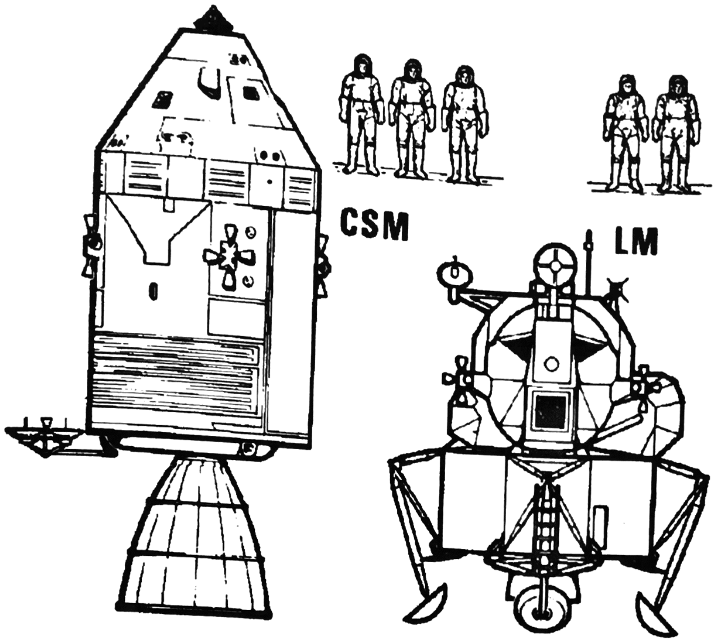


FIG. 2-55: **The APOLLO Space Vehicle and the Lunar Lander**

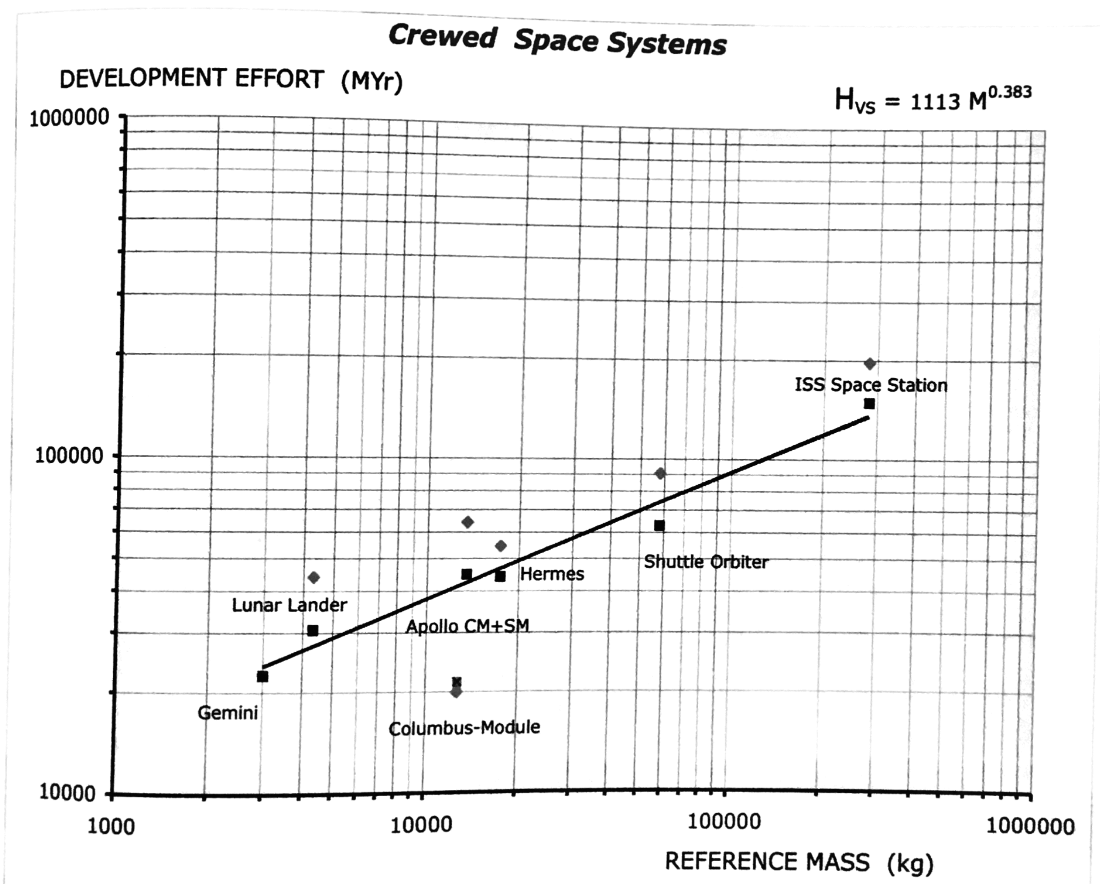


FIG. 2-56: **Reference Projects and Basic CER of Crewed Space Systems**

The third type of crewed space systems are Interorbital Space Transfer Vehicles and Landers, such as developed in the US APOLLO Program for transfer from Earth Orbit to Lunar Orbit and to the Lunar surface (FIG. 2-55). These vehicles comprise multiple and versatile propulsion capabilities.

The preliminary CER for crewed space systems has been defined as

$$\boxed{\mathbf{H}_{VS} = 1113\ M^{0.383}\ f_1\ \ f_3} \qquad \text{MYr}$$

Due to the different nature of the crewed space systems no TQF (Technical Quality Factor) has been established. The reference projects' data fit is very good without a TQF. FIG. 2-56 shows the basic CER derived from six reference projects of different nature.

2.5 Cost Impact of Development Program Preparation, Organization and Schedule

2.51 Project Definition and Preparation / Changes

2.511 Realistic Mass Estimates

It has already been mentioned that realistic mass estimates are important for a reliable cost estimation. Underestimation of the dry mass leads to too low cost. In the initial project study phases the mass estimate can never be really complete. Therefore, initial mass margins of 10 to 20 % should be applied in order to take care of all the additional small hardware items which cannot yet be identified in the conceptual project phase. The table in chapter 1.22 shows recommended mass margins which can decrease with the progress of the project definition.

The history of all space projects proves that a certain weight increase during the development phase is the normal case. FIG. 2-57 shows a typical example of project mass vs. time including a „Weight Reduction Program" which normally is an expensive exercise. The example shown is from the Lunar Landing Vehicle (LM) with a final 27 % mass growth for several reasons. One is the fact that it was the first project of this kind.

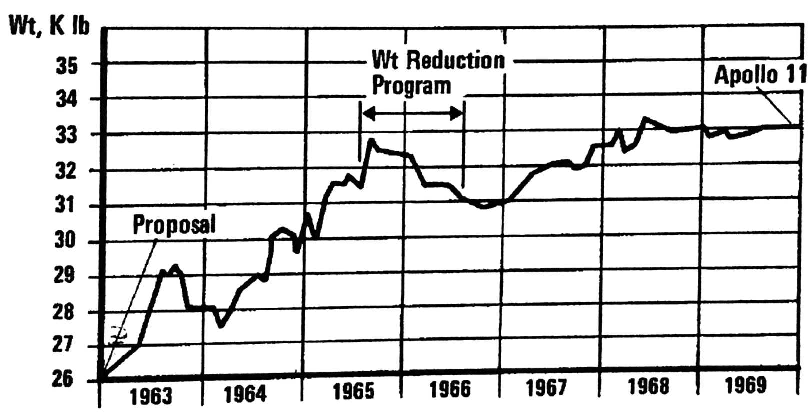


FIG. 2-57: **Vehicle Mass vs. Development Period** (Apollo Lunar Lander Vehicle)

2.512 Crewed vs. Automated Vehicles („Man-Rating")

In case of launch vehicles three cases can be distinguished (a) piloted vehicles with cockpit, (b) cargo vehicles, and (c) vehicles with crew module in the cargo bay. This raises the question for the difference in design requirements, a subject that was dealt with at the „Human Rating Workshop" at the NASA Johnson Space Center with some 120 participants from NASA, DoD and the Aerospace Industry on Nov.19-22, 1991. The main result was the conclusion that all launch vehicles have to be designed for maximum safety and reliability, independent of carrying a crew or cargo.

Another statement was : „Acceptable safety risks must be balanced against cost to make design decisions". New reusable launch vehicle concepts normally favour automated systems without cockpit, but foresee the possibility of a pressurized crew compartment which can be placed into the cargo compartment.

2.513 Project System / Subsystem Definition Standard

Another „must" for avoiding later project cost increases is the careful and detailed definition of the project itself with its subsystems and internal / external interfaces. The detailed project definition must be complete and with suffient detail before the start of the hardware phase. The uncertainty with respect to project schedule and cost risk increases with lower technology readiness status of the different subsystems and components.

TABLE 2-III shows the definition of the „Technology Readiness Levels". For a safe project development all subsystems and components should have reached at least level 6 at the start of Phase C/D. If this is not completely the case then specific activities should be performed in a pre-development program.

Without such a preceding technology development and verification activity substantial cost growth during project development can occur. This has been proven by several Shuttle subsystems as well as by a number of different spacecraft projects.

TABLE 2-III : **Definition of Technology Readiness Levels**

TRL 1	Basic principles observed and reported („Basic research")
TRL 2	Technology concept and 7 or application formulated („Applied research")
TRL 3	Analytical and experimental critical functions and/or characteristics demonstrated ("proof-of-concept")
TRL 4	Component and / or breadboard validation in laboratory environment (First step of dedicated technology development)
TRL 5	Component and / or breadboard validation in relevant environment (project-oriented)
TRL 6	System / subsystem model or prototype demonstration in a relevant environment (ground or space)
TRL7	System prototype demonstration in a space environment (Subsystem development)
TRL8	Actual system completed and „flight qualified" through test and demonstration (ground or flight)
TRL9	Actual system „flight proven" through successful mission operations

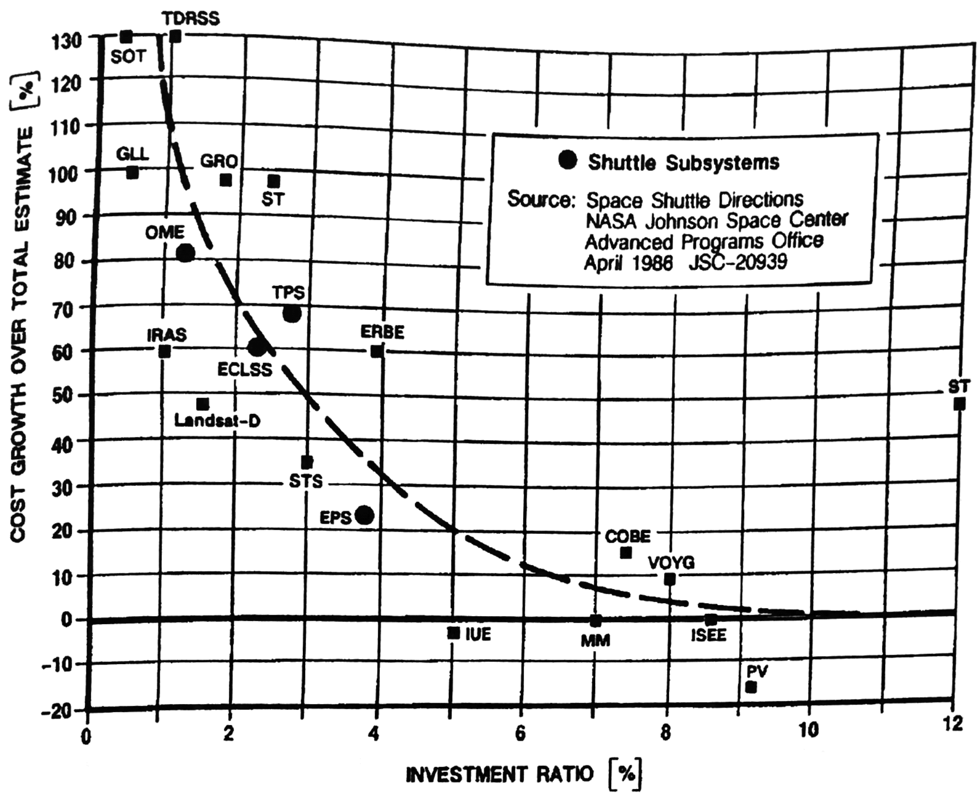


FIG. 2-58: **Historical Cost Growth of Spacecraft and Space Shuttle Subsystems vs. Investment before Project Development Program Start**

FIG. 2-58 illustrates the fact that a later cost growth can be avoided by a timely technology development and verification program investment of few percent, compared to the cost of the dedicated project development.

2.514 Technical Changes during Development

There are two kinds of changes:

a) customer-requested changes
(change or supplement of specifications), and

b) internal changes, required by unsuccessful technical approaches or poor interface definition.

The second type of changes can be minimized by a good project preparation, as described in the previous chapter.

The first kind of changes is difficult to avoid but it must be made clear what the cost will be before the change is implemented. Many changes are started without any idea of the cost impact. Afterwards the question comes up: who is responsible for the change, who pays ??

2.52 Cost Impact of Program Organization and Type of Contract

2.521 Program Organization

The organization principle for the development of a complex technical project is well known: it requires a clear-cut prime contractor / subcontractor relationship with well defined responsibilities. As simple as this well-proven rule is, so more surprising is the fact that often this rule is ignored in favour of several „parallel" contractors without a strong prime contractor, mainly for political or „prestige" reasons.

It can easily be proven how any other organization principle with several parallel contractors (or „ co-contractor" principle) with coordination by the customer or an additional organization leads to higher project cost. A more recent example for this type of project organization was the first phase of the US Space Station Program for many years, until it became evident that an industrial prime contractor is unvoidable. Sometimes the argument is raised that the subcontract overhead cost charged by the prime contractor could be saved by this type of organization, but this is self-deception since the additional costs will then show up on the customer side - some-times hidden in the general budget, but certainly not disappearing.

From historic cases an empirical model can be established, based on the number of parallel or associated contractor organizations. Sometimes such relations are hidden for good reasons but it can be determined by revealing the command lines, the number of „coordination committees" and meetings (and the number of meeting participants !). This indicates already why such an organization means project cost growth: more manpower, more interfaces, planned and unplanned parallel activities and - accordingly - schedule delays.

FIG. 2-59 shows the empirical model and the reference points for the project cost increase with the number or parallel organizations. The resulting cost growth factor is

$$\boxed{f_7 = n^{0.2}}$$

with n = number of participating parallel organizations.

It is not that bad as the sometimes mentioned n - relation, and it does NOT refer to the number of companies or countries working jointly on a project. If they are organized strictly according to the prime/ subcontractor principle then no essential cost growth should occur.

Numerical examples for the derivation of the empirical factor f_7 are, by example, the development of the ELDO (European Launcher Development Organization) launch vehicle program „Europa I/ II" by 9 national organizations in parallel, and specifically the development of the third stage (ASTRIS) by an „Arbeitsgemeinschaft" (Working Group) of two companies (no prime). The *TRANSCOST* -Model would give cost of some 408 M.DM while the effective total cost were 484 M.DM. The cost growth factor was $f_7 = 1.19$, including some other secondary effects. The author's participation in this project allowed the insight which is rarely available in other cases. The ELDO Program is also an example that the lack of a strong industrial prime contractor does not only increase cost but also can lead to failure. A similar mistake is, not to clearly separate between contractor and prime contractor tasks, i.e. that the prime contractor (agency) assumes a partial role of a prime contractor while the prime contractor (company) is reduced to an engineering assistant role.

Another reference point for the empirical cost growth factor from the spacecraft area is the German AZUR Satellite Project with 6 companies working under parallel contracts from the customer organization. The „normal" cost comparable to other satellite projects at that time should have been some 41 M.DM while the effective cost were 60 M.DM, or a cost growth factor of 1.46.

A more recent example for cost saving by a clear contractor / prime contractor relationship is the reorganized Space Shuttle Operations prime contract: Instead of giving 5 contracts to 5 different companies working in parallel in 1996 one single contract was awarded by NASA to the „United Space Alliance"., reducing the annual cost from 3.2 to 2.43 Billion $, or by 31.6 %[1]

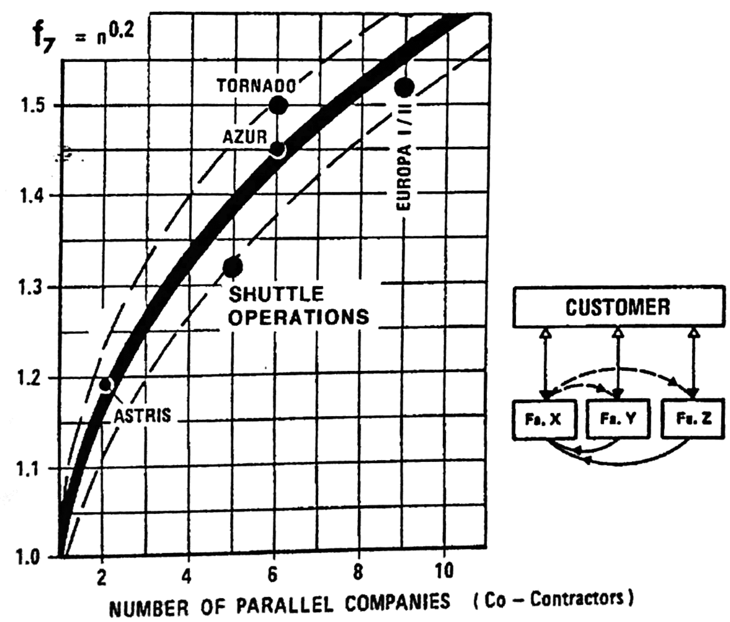


FIG. 2-59 : **Cost Growth Factor for Parallel Contractor Organizations**

1 according to Aviation Week, 22.4.96

2.522 Type of Contract

The contract conditions can have an impact on the project cost, depending on the type of contract. There are the following major contract types:

(a) Cost plus a Percentage Fee (5 to 7 %)
(b) Cost plus Fixed Fee
(c) Cost plus Award Fee
(d) Firm Fixed Price.

(a) The „Cost plus ..“-type of contract is no challenge for cost limitation - more so to the contrary. A percentage fee is a challenge for the contractor to increase the contract volume and cost as much as possible.

(b) The Fixed -Fee -Contract tries to solve the problem of a wrong incentive by eliminating the fee on cost overruns.

(c) The Award Fee Contract based on schedule milestones, technical performance and final cost is a strong means of cost management, especially if an award fee is guaranteed for cost savings. This contract type seems to be the only one which can help to avoid cost overruns, or even can lead to cost reduction. For a contractor this provides the chance for higher profit (up to 15%), and the realized percentage between 0 and 15 % finally shows the performance of the company.

(d) The Firm Fixed Price Contract looks to be the most simple one, but in reality it is one of the most difficult one for both contract partners. A certain risk provision required for development contracts, is normally not agreed by the customer, resp. cannot be requested by the contractor for competitive reasons. The FFP contract requires a detailed definition of the development subject and conditions. Any deviation from the basic contract requires a new „Contract Change Agreement“ which is a lot of bureaucracy on both sides. This procedure can induce critical delays in the projects schedule (with the additional problem „who caused the delay ?). For these reasons the FFP Contract *for development tasks* finally can become an expensive one for the customer. By no means it does prevent cost increases, as some people believe.

The FFP is more applicable for studies and for production contracts.

2.523 Project Management and Reporting /Reviews

The Project Manager and his staff is responsible for the implementation of the technical requirements and performance but at the same time for the basic proposal cost estimate and the later adherance to it. The project management team is also responsible for preparing the regular progress reports and review meetings. The size of the project management team and its staff must be carefully determined: it should be as small as possible and have some education in cost engineering. The required team size (and cost) are strongly influenced by the contractual reporting requirements (scope and frequency). The larger the custumer's project team, the larges the contractor's team has to be; therefore, it's size has a two-fold impact on the program cost.

2.53 Cost Impact of Development Schedule and Funding Profile

2.531 Major Schedule / Cost Factors

The development schedule and/ or the related annual funding has a major impact on the project cost. Each delay compared to the initial „optimum" schedule increases cost. There are multiple reasons for a schedule delay such as

- -- requirement changes during development,
- -- technical changes as „improvements",
- -- technical component/ software failures,
- -- changes in management structure or personnel,
- -- funding limitations (per budget year).

Since no project is immune against these impacts almost no program yet has been accomplished on the schedule initially planned for (except if a sufficient margin had been included at the beginning).

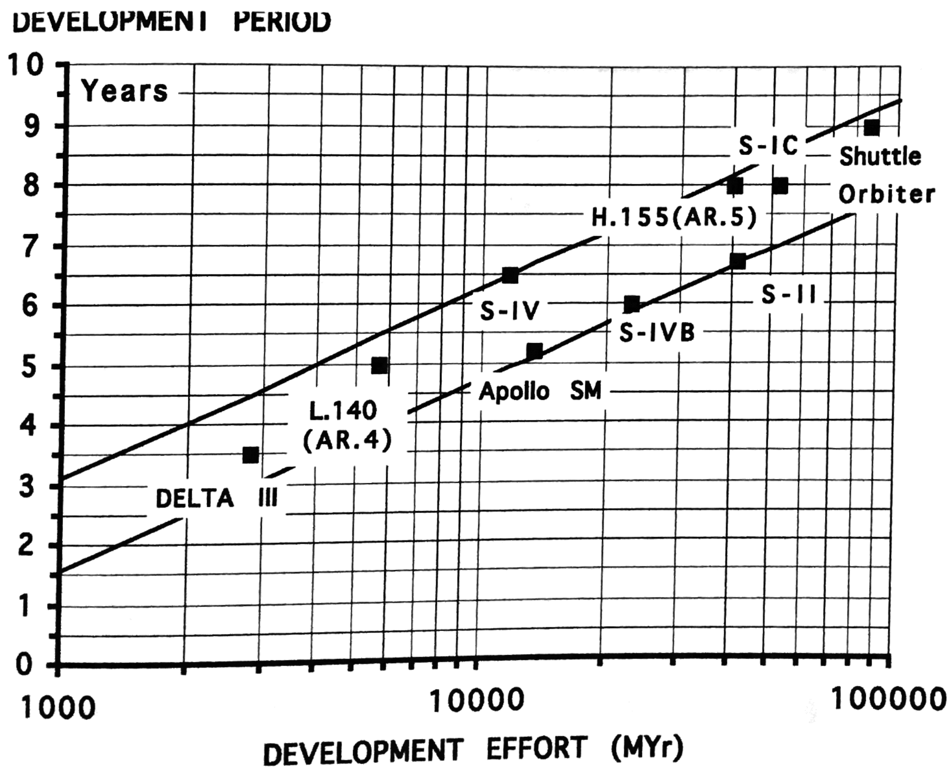


FIG. 2-60: **Development Period until First Flight vs.Total Development Effort**

2.532 Program Duration and Funding Profile

The optimum duration of a development program which results in minimum cost depends on the vehicle type and the totel development effort. FIG. 2-60 shows the trend for a number of vehicle systems for the period Start of Phase C/D until the first flight. This is not the complete development period but up to the key milestone. The complete development period depends on the success of the first flight and the number of required qualification flights before the launch system is declared „operational".

However, it is not only the program duration which has a major cost impact but in addition the funding profile, i.e. the funding distribution over the development period. There exists an „optimum funding profile" according to the inherent logic of a development activity. In case of unsufficient annual funding in comparison to the original budget requirements, or putting a limit on the annual expenditures which artificially extends the development period, a cost increase is unavoidable.

FIG. 2-61 shows the funding profiles of different launch vehicle stages in comparison to the optimum profile. It is evident that under-funding in the first years (or a delayed progress of work due to poor planning and/or unforeseen technical difficulties increases cost in the second half of the project, and/ or requires a schedule extension.

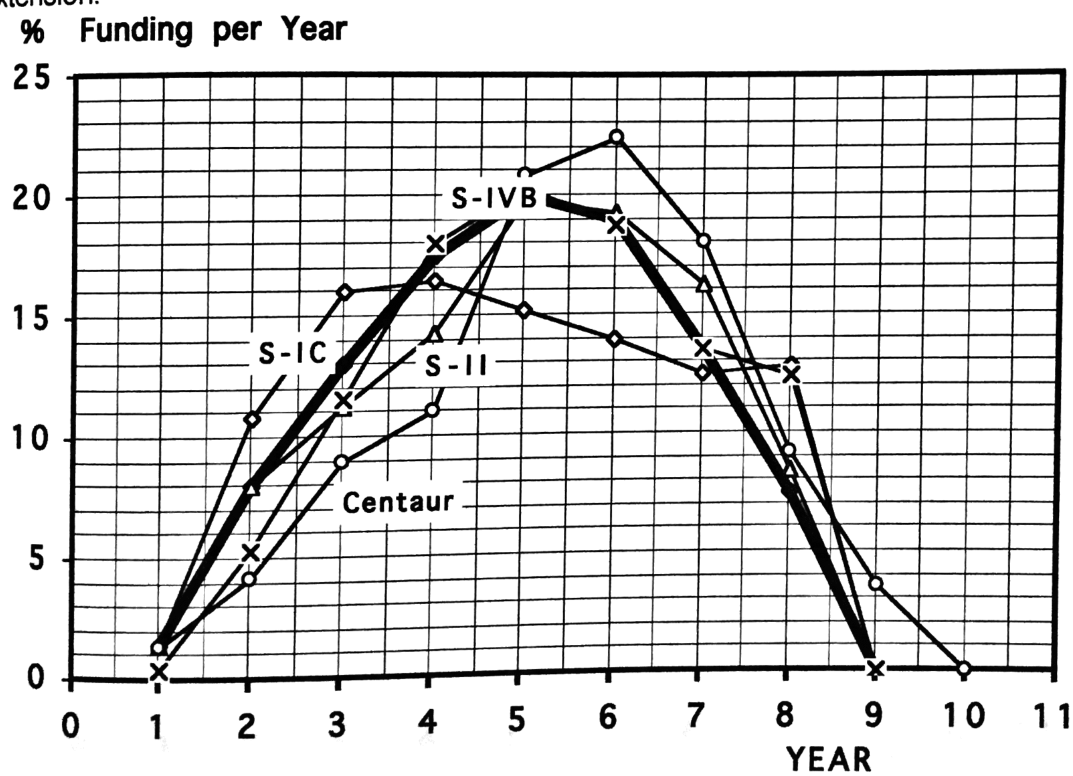


FIG. 2-61: **Historic Project Funding Profiles in Comparison to the „Ideal" Contribution vs. Development Years** (Solid Curve)

Three examples: The SATURN S-IC Stage was planned (and initially funded) as a seven year program, but delays in the years 5 and 6 caused extension and even double funding and manpower in the 8th year, compared to a „normal" 8-year-project profile.

The CENTAUR vehicle development was originally conceived as a 4 to 5 year project - underestimating substantially the required development effort for this first hydrogen-fueled vehicle. Only after 4 years and the first launch ending in a failure a substantial funding increase and schedule extension was realized - far above the „normal" project implementation plan for such a project.

In case of ARIANE 5 a wrong budget planning forced ESA/ CNES to take a bank credit to cover the funding gap - with associated bank interest cost - because a delay of the development schedule would have been even more expensive.

2.533 The Optimum Development Schedule

FIG. 2-62 shows „standard" development cost or budget profiles for 6, 7, 8 and 9-year programs from program start (Phase C/D) to the first launch. If the budgets become limited below the ideal profile this normally will lead to increased costs in the second part of the project and a delay of the first launch (schedule extension). A total cost growth is unavoidable not only because of the annual inflation or hour rate increase but also because of less efficient work procedures or team utilization.

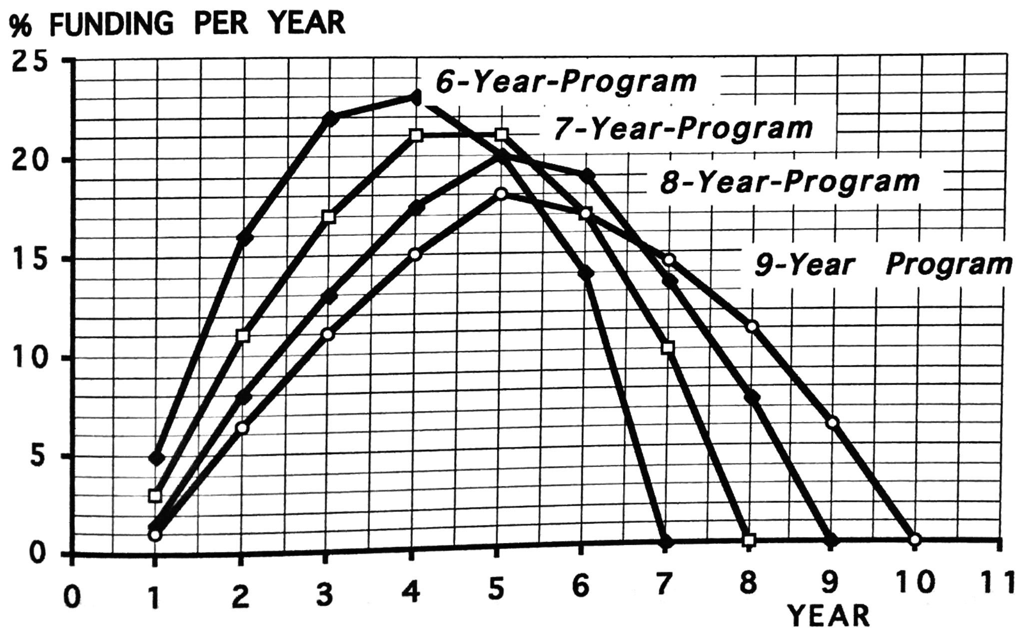


FIG. 2-62: **Ideal Funding Profiles for 6, 7, 8 and 9-Year Development Programs** (Start of Phase C/D until First Launch)

What is the „ideal" development period for a space launch system ? Certainly it should be as short as possible, and this depends on the organization and the available resources. If we look into historic cases then vehicles requiring 8 000 to 12 000 MYr should be planned for a development period of 5 to 6 years, vehicles with 15 to 25 000 MYr for a period of 6 to 7 years, and vehicles between 35 and 50 000 MYr for a period of 7 to 8 years (in the past there has been usually a development schedule extension by one year or two because of technical or funding problems). The Space Shuttle ORBITER, originally planned with some 70 000 MYr and a 7.5 year schedule, was extended to a 9-year schedule requiring some 85 500 MYr. This, however, is not surprising since it was the first vehicle of its type - a winged manned reusable re-entry glider. The ARIANE 5 development took 13 month more than planned (+15 %), and the cost did grow for this reason by some 3000 MYr (ca.8 %) until the first launch plus additional costs for technical changes. The Japanese H-II launch vehicle had a development schedule extension by some 19 %, and a cost growth by some 15 %.

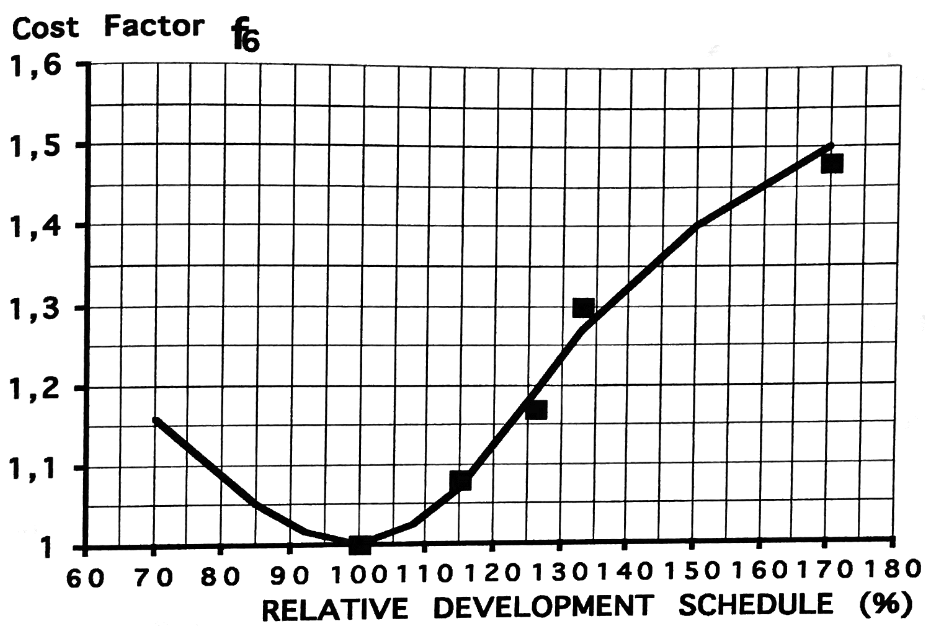


FIG. 2-63: **Cost Growth by Deviation from the Optimum Schedule (= 100 %)**

As a first empirical approach it can be derived from history that a delay or schedule extension by some 20 % will cause a cost increase of 10 to 15 %, and a delay by some 40 % a cost growth of 30 to 35 %. This is illustrated in FIG. 2-63. As indicated, also an accelerated schedule will cause higher cost due to the overtime and/ or shift activities to be paid for, and some additional parallel work. However, this applies only to a conventional development program. In case of reduced effort, as discussed in the next chapter, schedule and cost can be reduced, compared to a BAU-program.

2.6 Strategies for Reduced Development Effort

2.61 Business-as-Usual (BAU) Cost Drivers

The development effort as defined in the *TRANSCOST* CER's is based on and assumes the traditional relations and conditions of a development contract between a governmental agency (NASA, ESA, NASDA etc.) and an industrial contractor. Due to traditions developed and control functions required for public funds the overall development scheme requires additional efforts and time which is not necessarily required in case of industrial projects with commercial funding.

The past space vehicle development under contract of governmental agencies have been relatively expensive. The cost drivers in this case can be summarized as follows :

(1) Duplication of management teams at customer and contractor side
-- (engaging each other largely), requiring more personnel on the contractor's side in order to „cover" the government oversight.

(2) Micromanagement Procedures:
-- Overspecification (unnecessary requirements)
-- Regular (weekly) Progress Reports,
-- Excessive project reviews[2] and documentation
-- Preoccupation with traceability, and other „...ilities"
-- High bureaucratic effort for contract changes and supplements.

(3) Aversion to risk,
-- instead of a step-by-step approach and verification, individual test programs for each component.

(4) Extended Schedule by
-- Long acquisition cycle with „dead periods" in between
-- Frequent „stops and starts" by funding fluctuations
-- Design changes / Approval process.

(5) „One of a kind" designs, according to the preference of the project manager(s) instead of using existing (may-be non-optimum) systems and hardware.

(6) Sequencial small quantity production contracts
-- instead of one longer term contract/ order.

It is widely accepted that the high development cost must be / could be lowered by reduction of the extensive control procedures of past development contracts. Using

[2] ESA Newsletter „OnStation", Dec.1999, proudly reports that for the X-38 CRV contributions by ESA 22 Design Reviews have been held within 21 months

the previous experience and features of „Rapid Prototyping", „Skunk Work Principles" or „Concurrent Engineering" in future governmental contracts could lead to a 33 % cost reduction according to an analysis of NASA Comptroller' s Office (ref. 94).

A Study by NASA's Langley Research Center with Lockheed (ref. 46) on the development program for a Lifting Body Vehicle (HL-20) resulted in a potential 55% development cost reduction compared to traditional BAU methods. The time schedule was reduced in this example by 20 %.

It can be concluded that through revised procedures and introduction of some modern development features the development cost of governmentally contracted projects could be reduced to 50 - 70 % of the traditional BAU cost.

2.62 Commercial-Industrial Development Features for Cost Reduction

There are only few examples from the past and those limited to relatively small projects. It is difficult or impossible to finance larger projects in the space transportation area commercially because of the long development periods which delay the payback period to normally unacceptable time periods.

One major successful efforts of this kind was the SPACEHAB development. It was implemented for 105 Mio.$ while the NASA estimate for development and construction of 2 units was in the range of several hundred million Dollars.

Another example in the spacecraft area is the German SPAS Project (Shuttle Pallet Satellite), an industrial in-house project of MBB /Dasa, now part of EADS, with 1500 kg mass (without payload instruments) realized in 1980-82 with 50 % own funding. The total development cost of this novel satellite type designed for short but repeated missions were only 15 M.DM or 68 MYr - a small fraction of the estimated cost under a governmental contract.

More experience with purely commercial-industrial development projects exists in the aircraft industry. By comparison of military transport aircraft and civil passenger aircraft development as shown in FIG. 2-46 it can be concluded that commercial airliners are developed at a cost level of 35 to 40 % compared to a military aircraft with government (USAF) contract. The first major commercial venture in the launch vehicle area is the development of the ballistic reusable K-1

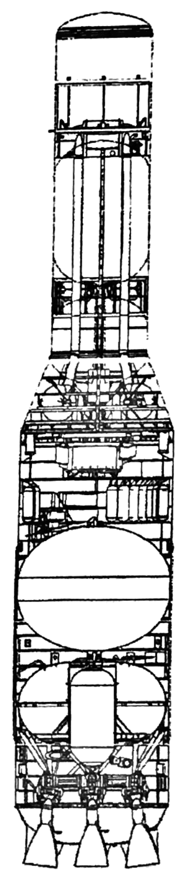


FIG. 2-64: **K-1 Vehicle**

Vehicle by Kistler Aerospace, Kirkland, WA,USA (FIG. 2-64). The two-stage vehicle with 380 Mg launch mass uses conventional existing technology but nevertheless represents a risky venture due to the limited test and verification effort enforced by the funding situation. The development cost until the first flight (planned to take place in Woomera/ Australia) are estimated to about 1 Billion US Dollars. This would represent some 20 % of a conventional governmental BAU development. The funding has mainly raised from private investors by issuing shares of the company so that no credit payback is required. The development finally may be completed eith help of NASA funds for n operational ISS delivery service.

From these examples it can be concluded that a strictly commercial-industrial development can be performed with 20 to 40 % of the cost of a traditional BAU-project. The lower value applies to projects with existing technology. The more new technology is involved the more expensive and risky the project will be.

TABLE 2-IV: **Lockheed Skunk Works Management Rules**

-- Strong Program Manager with Complete Authority
-- Strong, *SMALL* Customer Program Office
-- Small Number of Good People on Contractor Team
-- Simplified Drawing Release System
-- Minimal Documentation - Only Important Work Documented
-- Regular Cost Reviews - No Surprises
-- Contractor Responsible for Subcontractors
-- Eliminate Duplication of Inspections
-- Contractor Testing of Final Product in Flight
-- Stable Requirements
-- Timely and Adequate Funding
-- Mutual Trust between Contractor and Customer
-- Outside Access Strictly Controlled
— Reward/ Compensation Based on Excellence

The Lockheed „Skunk Works" activities have become known as a way of cost-efficient development for military (classified) projects. The Skunk Works Management Rules - as shown in TABLE 2-IV - do not really include special or new features; it is the strict applications of these rules that makes the approach a successful one.

The basic ground rules for a cost-efficient development strategy have already been discussed in previous chapters:

(1) High level of project definition - including a cost-optimized design from the very beginning;

(2) Maximum use of existing elements (COTS),

(3) New technologies' verification to be completed before development start,

(4) Prime/ Subcontractor Project organization with clear-cut responsibilities and a contract with award clauses;

(5) Fit of schedule and funding profile as required for optimum work sequence;

(6) Issue of subsystem specifications as late as possible (to avoid or reduce changes) and competitive procurement;

(7) Early experimental and model test philosophy employed.

The last point means that a careful trade-off between theoretical design/ calculation efforts and experimental testing must be performed. „Rapid Prototyping" is the designation of the strategy where time-consuming and expensive detailed design and theoretical verification efforts are replaced by early construction and testing (sometimes testing to failure) in order to verify the design. This strategy was used generally during WWII for military projects and has been always the strategy of Russian space projects.

FIG. 2-65 shows a cost comparison of the „Rapid Prototyping" strategy with the standard practice as conceived by McDonnell Douglas with the Delta Clipper experience (ref. 51). Essential is a short schedule which has proven its feasibility in previous projects: the SR-71 „Blackbird" Mach 3.5 aircraft flew only 30 month after contract award (with a lot of new technology); the THOR - IRBM

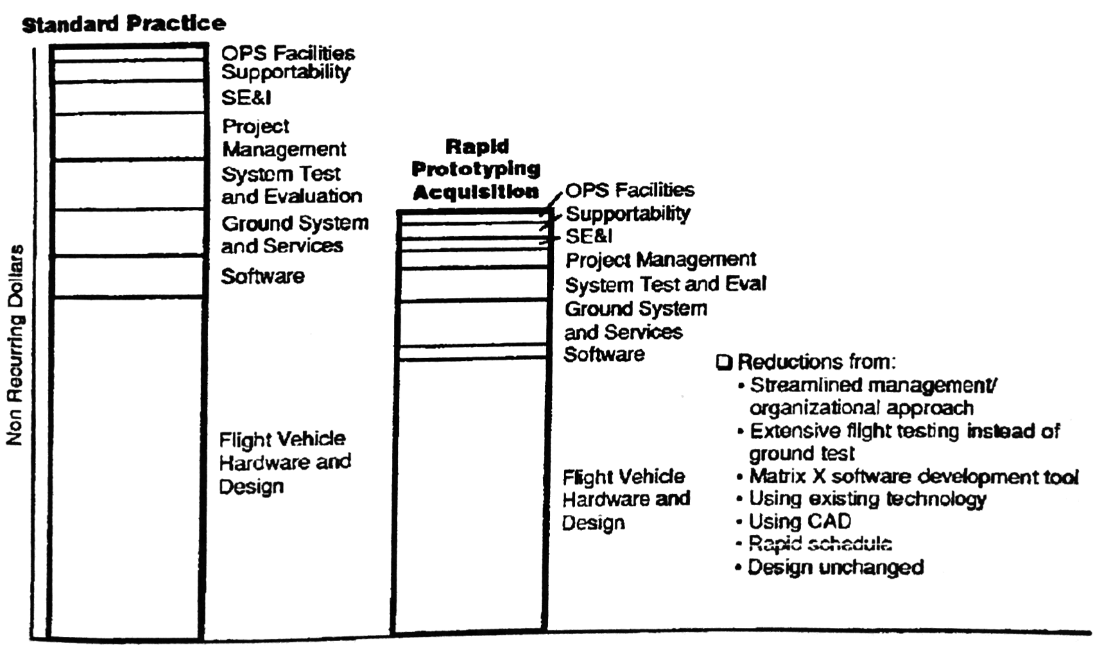


FIG.2-65: **Cost Comparison of „Rapid Prototyping" and Standard Strategy**

was launched the first time only 13 month after contract award in Dec. 1955 (ref. 48), but resulting in this case in a high test failure rate.

SUMMARY:

Essential development cost reductions can be achieved by:

- Reduction of project bureaucracy and paperwork (reporting, reviews),
- Maximum use of existing elements (COTS),
- Reduction of theoretical analyses in favour of „rapid prototyping" and testing,
- Minimum project team size and development schedule,
- Maximum use of existing components.

2.63 Experimental and Demonstration Vehicles

It is on open issue whether subscale experimental vehicles or „demonstrators" lead to a cost reduction or a cost increase of the complete development program, taking into account also the delay of the real size vehicle's operational readiness. Nevertheless such a vehicle can be an important programmatic step if the real-size program cannot yet be funded, or technology verification by a flight vehicle seems to be indispensible.

While the preferred commercial/industrial approach is „rapid prototyping" of the actual vehicle (Thor IRBM, SR-71, Kistler K-1 RLV) the preferred governmental research and development approach is to build subscale test vehicles. This is a typical step-by-step method in order to minimize risk. Such experimental vehicles can be low-cost or very expensive.

A low-cost example is the DELTA CLIPPER **DC-X** experimental vehicle, developed and flight tested by McDonnell Douglas at the White Sands Missile Range under an SDOI-Contract of only 70 Mio.$. This is equivalent to the normal TFU recurring cost - after vehicle development. The total cost including the industrial contributions may have been some 100 M$, or 600 MYr. This test vehicle with 18 Mg GLOW and 9850 kg dry mass was using existing technology, however, with some interesting new features as the complete external structure made of CFC, the pneumatic landing gear system, a CFC hydrogen tank and a gaseous hydrogen/ LOX thruster system for attitude control . It was designed for low-speed operation only since the main purpose was the demonstration of altitude maneuverability with the RCS, the vertical landing with rocket power, and quick turnaround ground operations with a staff of only 15 people. 12 successful test flights of up to 142 sec duration with successful demonstration of maneuverability in an altitude of 3 000 m have been performed. Vertical landing with thrust modulation of its four P&W RL-10 engines on four extendible legs was a new first-time demonstrated vehicle design feature, until a maintenance fault led to an accident and the end of the program in Sep.1996.

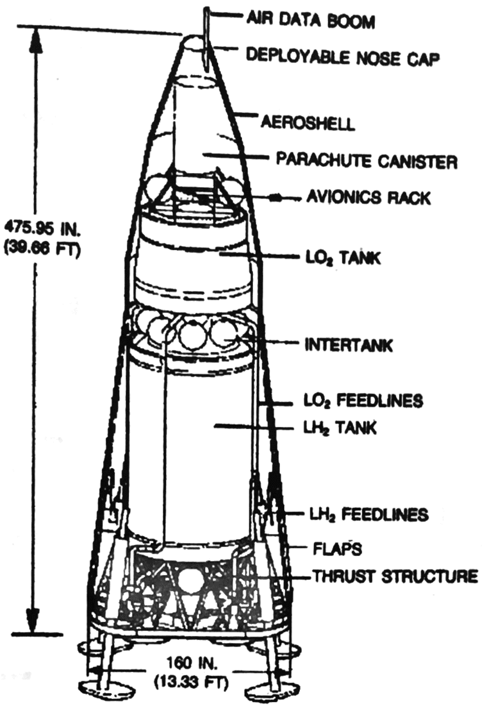
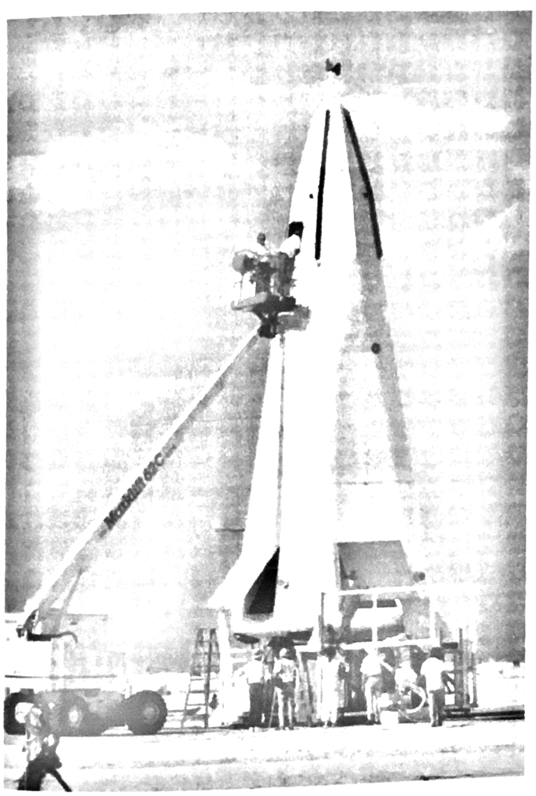


FIG. 2-66a: **DC-X VTOL Experimental Vehicle**



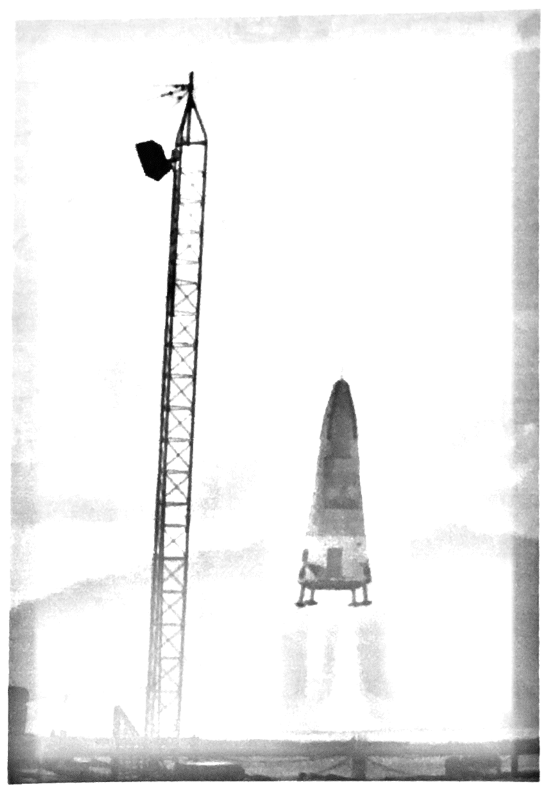


FIG. 2-66b and c: **DC-X Launch Preparations and Vertical Landing Operation**

Another low-cost project was the air-launched **X-34** Winged Test Vehicle (FIG. 2-67) with 21 Mg launch weight and 7700 kg dry mass as a Mach 8 demonstrator. The contracted cost of 70 M$ (Orbital Sciences Corp.), however, did not include the NASA-MSFC-developed Fastrac rocket engine (ca. 40 M$) and other NASA contributions. The industry involved had accepted a 120 M$ cost share, so that the total cost projection was about 250 M$ or 1250 MYr (This is much less than for the manned X-15 Rocket Aircraft Program 1957-59 with similar objectives : 163 M$, equivalent to 5800 MYr at a dry mass of 6700 kg). The low funding level, technical problems and the complex project organization led to a critical situation so that the X-34 project was terminated by NASA in March 2001.

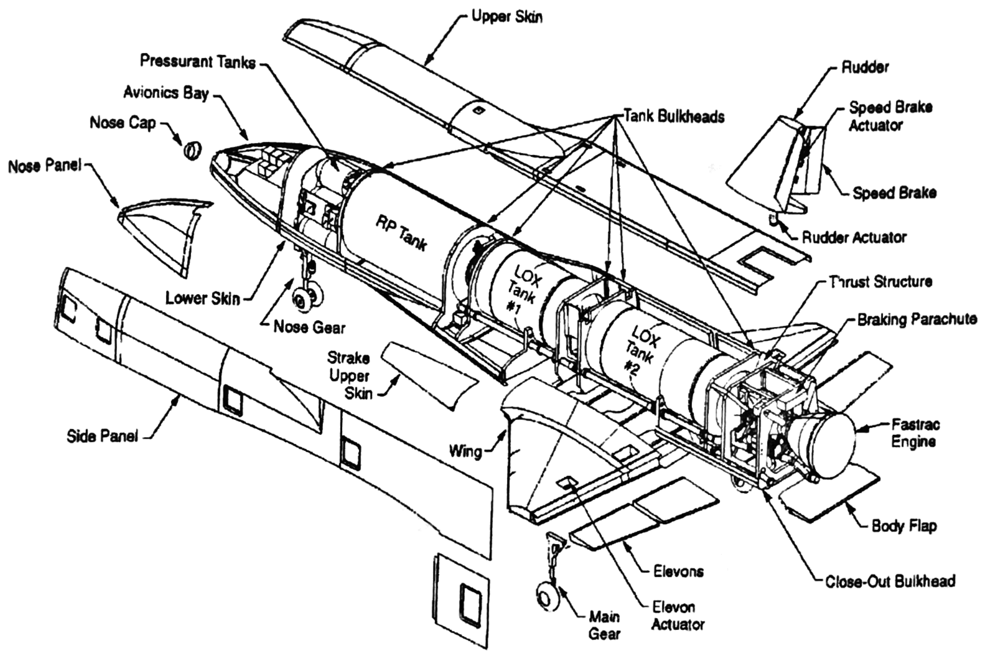


FIG. 2-67: **X-34 Winged Test Vehicle Vehicle**

A much higher budget was planned for the unmanned X-33 Lifting Body Test Vehicle as demonstrator for the Lockheed Martin VENTURE STAR reuable launch vehicle. The 135 Mg GLOW vehicle with 96 Mg propellants (LOX/Hydrogen) and about 36 000 kg dry mass had a NASA budget of 941 M$ plus the industrial contribution from Lockheed Martin and Boeing-Rockwell of originally 231 M$. This was growing to some 400 M$ due to the technical difficulties and the resulting launch delay of some 3 years . The total cost could be estimated to at least 1.4 Billion $ or 7000 MYr. This is due to the difficult technical concept selected (lifting body with complicated tank configurations made from CFC) and the technologies which were not yet verified at project start (tanks, TPS and linear aerospike engine). Unforeseen engine problemsonly caused a cost increase of 36 M$ according to Rocketdyne[3]. The

[3] Aviation Week, 2.Nov.98

weight growth would have allowed only a maximum speed of Mach 11 to 13, compared to the original goal of Mach 15. Due to the technical problems and further financing the project was terminated by NASA in March 2001.

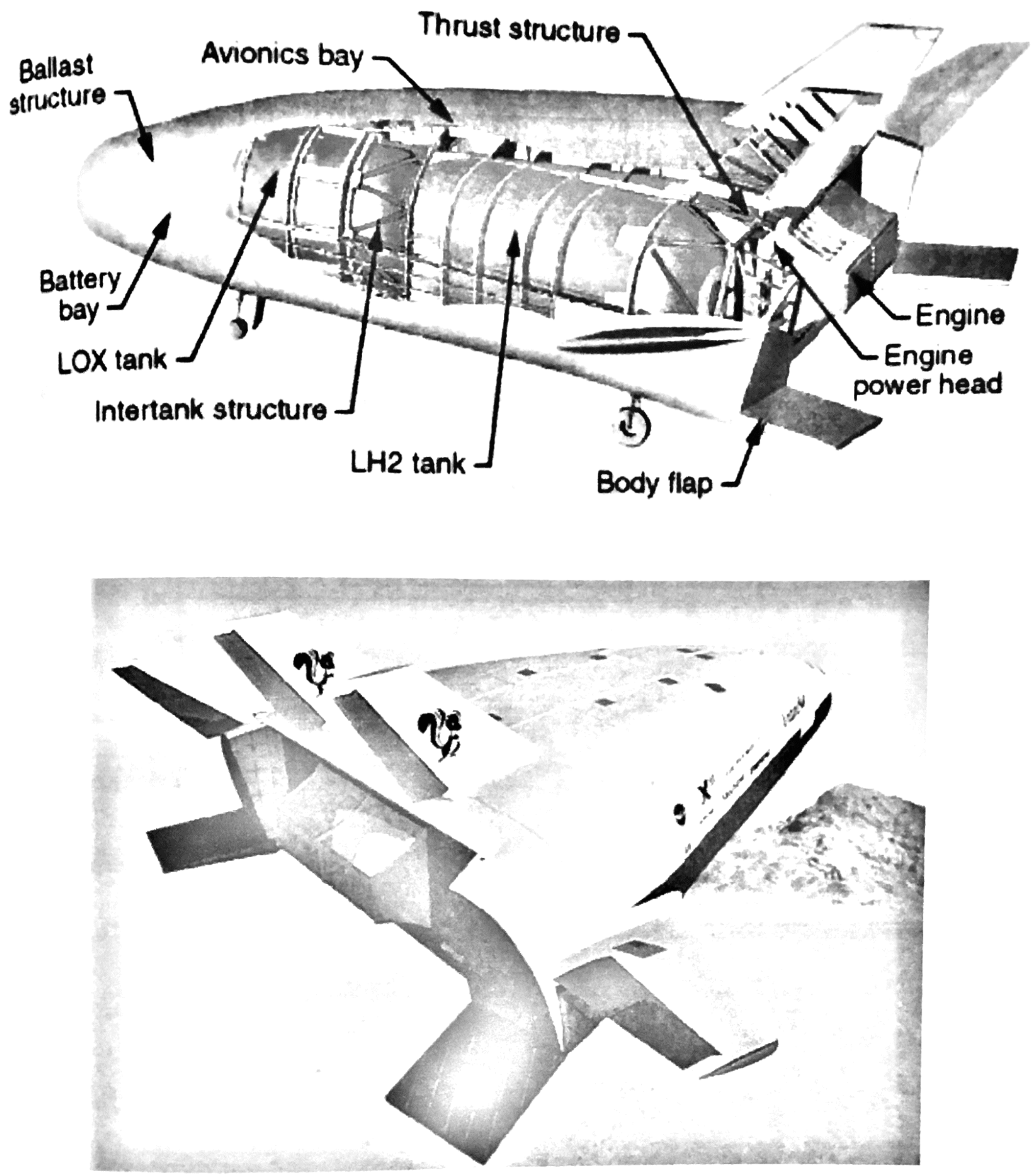


FIG. 2-68: **X-33 Lifting Body RLV Demonstrator Vehicle**

2.7 Cost Estimation Accuracy, Uncertainties and Risks

2.71 *TRANSCOST* -Development Cost Estimation Accuracy

The accuracy of the TRANSCOST NRC data can be very high as shown by examples. It depends very much on

(1) the careful consideration of *all* development cost criteria, as listed, and

(2) on realistic input data for the different vehicle and engines' mass values, as well as for the schedule (program duration).

For a more extensive discussion of the accuracy problems of cost estimation see ref.12.

2.72 Development Cost History / Statistics

There have been cases were development costs have been considered as matter of physics with a statistical cost probability profile as shown in FIG. 2-69. Here it was suggested that there is a 50 % chance of lower than nominal costs as well as a 50 % chance for higher costs. The reality, however, is different.

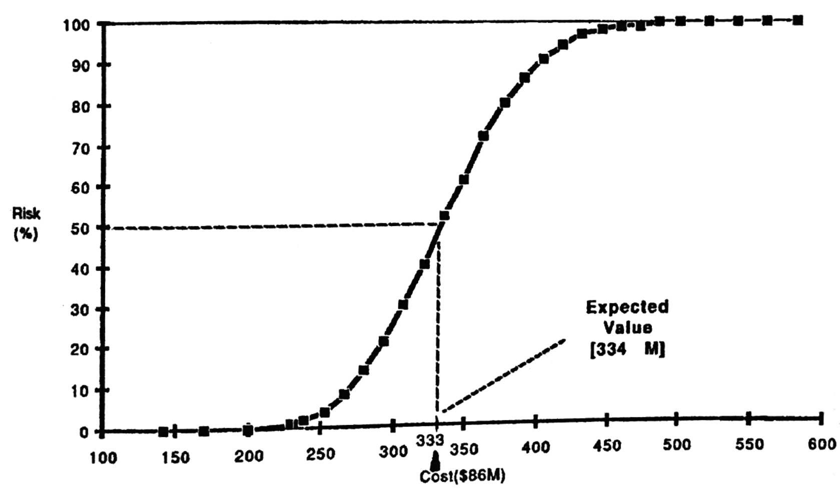


FIG. 2-69 : **Unrealistic Statistical Cost Distribution Model**

A GOA Report to the US Congress in 1993 shows the results of a cost survey on 29 NASA Programs:

- -- 14 % have been at or below the original cost estimate,
- -- 53 % have superceeded the cost estimation by up to 100 %, and
- -- 33 % exhibited more than 100 % cost growth.

This result demonstrates poor cost estimation capability and poor cost management / cost control. The psychological / political background, however, is the fact that

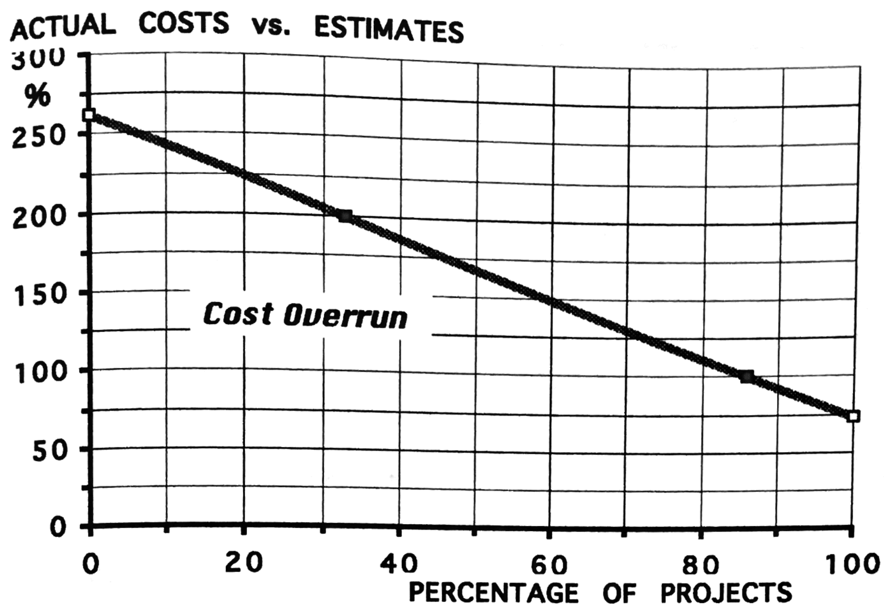


FIG. 2-70: **Effective Development Cost Results of 29 NASA Programs (1993)**

originators of project proposals have to be afraid of presenting very realistic cost estimates which could cause proposal rejection.

2.73 The Major Development Cost Risks

The major risks regarding the potential development cost growth are as follows:

(1) The technology employed was not fully qualified at the beginning of the program. Technical changes and additional qualifications do not only need additional funding but often mean a delay with a resulting costly schedule extension.

(2) The vehicle specifications were not complete and not frozen at the start of the program. Any change of specs or additional requirements will drive up costs and lead to difficult contract modifications.

(3) The vehicle and engines' dry mass values have been underestimated. There is a great risk of optimistic assumptions and/ or incomplete mass values. Without a sufficient initial mass margin the vehicle payload will decrease and/ or a costly redesign is required.

(4) A project schedule has been assumed which assumes that everything goes according to (the original) plan. However, mishaps and delays always ocur. If no time margin has been built into the original schedule it will be extended with the unavoidable cost increase.

2.8 Verification of the *TRANSCOST* - NRC Submodel

2.81 Space Shuttle Main Engine (SSME) Development Cost

The SSME is an advanced technology high-pressure rocket engine, originally specified for 55 flights on the Shuttle Orbiter. The development was performed in the 19972 to 1982 period which is 11 years instead of the initially projected 9 years. The engine dry mass increased during development from 3010 to 3180 kg.

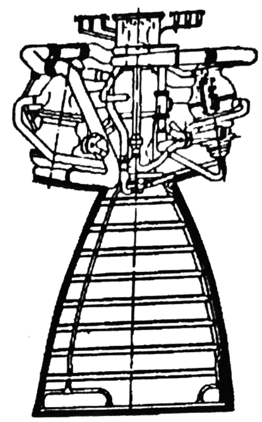


FIG. 2-71: **SSME**

The *TRANSCOST*-CER from chapter 2.32 results in an estimate of

$$H_E = 277 \cdot M^{0.48} \; f_1 \; f_2 \; f_3 \quad \text{MYr}$$
$$= 277 \cdot 3180^{0.48} \cdot 1.30 \cdot 1.22 \cdot 0.85 \quad \text{MYr}$$
$$= \underline{17\,921 \;\; \text{MYr}}$$

The f_1- Factor is 1.3 because the SSME is a „first of its kind" project with new technology. The Technical Quality Factor f_2 is 1.22 (cf. FIG. 2-11) since the number of tests at the end of 1982 was about 900 (730 at the time of the first STS-1 flight in April 1981). The team experience factor f_3 can be assumed with 0.85 due to the previous Rocketdyne experience with the F-1 and J-2 engines.

The actual expenditures according to the published NASA budget values (not „planned cost") and the related MYr-values are:

YEAR	72	73	74	75	76	TQ	77	78	79	80	81	82	Total
Mio.US $	45	41	56	95	135	36	182	197	173	141	134	127	**1362**
MYr	1023	820	1018	1597	2045	545	2528	2472	2005	1530	1357	1206	**18146**

The comparison between the actual cost and the *TRANSCOST*- Model value shows that the CER results are very close to the actual development cost (- 1,2 %)

The original development cost estimate of the competing companies in 1971, however, was only 500 Mio.$ = <u>12 500 MYr</u> [4]). Some 15% of the cost increase can be allocated to underfunding in the 2nd, 3rd and 4th year, causing the 2-year longer schedule (cf. FIG. 2-55) . The residual difference of 30 % has probably

4 Aviation Week, 21 June 1971

been caused by unforeseen technical problems with mass increase, and a proposal price at the lowest limit.

2.82 ARIANE 5 Development Cost

The ARIANE 5 development estimate and comparison is an example for a complete vehicle system. The different vehicle elements have been costed as shown in TABLE 2-V, representing the actual technical status of the ARIANE 5 launch vehicle as of 1999.
The individual element's cost may differ from the actual values, also because of a different definition of the scope, but the overall result seems to fit well with the cost stated in 1996 - if there would have been no program extension or schedule delay.

TABLE 2-V : ***TRANSCOST*-Development Cost Estimate for the ARIANE 5**

ELEMENT	REFERENCE MASS	CER-MYr	f_1	f_2	f_3	f_8	MYr
SRBs (EAP)	39 300 kg	5 938	1.1	—	0.9	0.86	5 056
Core Veh.(EPC)	11 210 kg	19 692	1.1	1.16	0.85	0.86	18 368
+ Shroud	2 400 kg						
Vulcain Engine	1 685 kg	9 800	1.1	0.79	0.9	0.86	6 592
3rd stage(EPS)	1 200 kg	7 689	0.9	1.09	0.9	0.86	5 838
+ Equipment Bay	1 300 kg						
Aestus Engine	119 kg	890	0.8	—	0.8	0.77	438

TOTAL Elements' Development Effort	36 292 MYr
with System Engineering Factor f_0 (= 1.04^3)	**40 827 MYr**
Total Program Cost	**46 500 MYr**

The official ESA development cost value for ARIANE 5 as of March 1987 was 3496 MAU (86) according to ESA Document BP-Ariane(87)WP/22. This is equivalent to 30 600 MYr (see TABLE 1-II). In April 1992 the total cost were stated to be 4747 MAU(91) in Doc. ESA-BP Ariane(92)42, equivalent to 32 500 MYr. In May 1996 a value of 6.58 Billion ECU was published[5], equivalent to some 39 300 MYr.

[5] Aviation Week, 6.May 1996, referring to ESA Program Manager J.Durand

The main cost increases resulted from technical changes during development, such as the thrust upgrade of the Vulcain engine from 770 to 870 kN , the increase of the propellant mass in the third stage from 5 to 9.7 Mg), as well as other unforeseen problems and incidents.

The development cost difference between the TRANSCOST-estimate of 40 800 MYr and the 46 500 MYr-value of 1999 can be accounted to the program schedule extension by more than one year until the first flight compared to the original plans, as well as the redesign and qualification of the flight control system after the catastrophic failure of the first test flight in June 1996. The required additional two test fligts took another two years more.

The resulting final program cost (CTC) are about 7.4 B. Euro, including the annual cost increases or inflation. By the year-to-year budget figures' translation into MYr this results in a total of 46 500 MYr, about 18 % more than the 1996 estimate, or 52 % higher than the initial estimate of 1987.

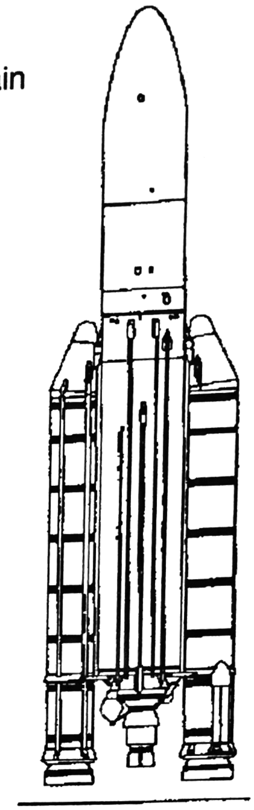


FIG. 2-72: **ARIANE 5**

2.83 Comparison of Vehicle Development Costs (Cross-Check)

For verification of the Net Mass and Dry Mass Charts for the different vehicle configurations FIG. 2-73 has been prepared showing the dry mass vs. (ascent) propellant mass for

-- Expendable Launch Vehicles (ELVs),

-- Ballistic Reusable Vehicles,

-- Winged Orbital Vehicles, and

-- Fly-back Boosters.

The values are taken from the model curves individually derived from the available reference projects. Apparently the curves show a good coherence. It must be taken into account, however, that due to the higher net mass of RLVs also a higher propellant mass is required for achieving the same performance (delta-V).

With the dry mass data from FIG. 2-73 and the individual CERs for ELVs and the three RLV configurations, FIG. 2-74 has been established. The result looks plausible.

For about the same vehicle performance the resulting development cost relation is shown in TABLE 2-VI. The development cost level according to the CERs only (assuming the same dry mass as for the ELV example) is shown in brackets. The higher cost level for RLVs is due both to the higher specific CER costs as well as to the inherently higher dry mass.

TABLE 2-VI : **Dry Mass- and Development Cost-Values of RLVs in Relation to ELVs**

Vehicle Concept	Dry Mass Factor	Dev.Cost Factor
-- Expendable Vehicles	1.0	1.0
-- Ballistic RLVs	2.2	2.4 (1.6)
-- Winged Orbital RLVs	4.1	4.0 (2.1)
-- Fly-back Boosters	5.7	3.4 (1.8)

Although FBB vehicles have the highest dry mass and need the largest propellant mass the development costs are somewhat lower than for Winged Orbital RLVs. This can be explained by the lower structure and thermal loads caused by the re-entry maneuver of orbital vehicles, as well as by the shorter mission duration

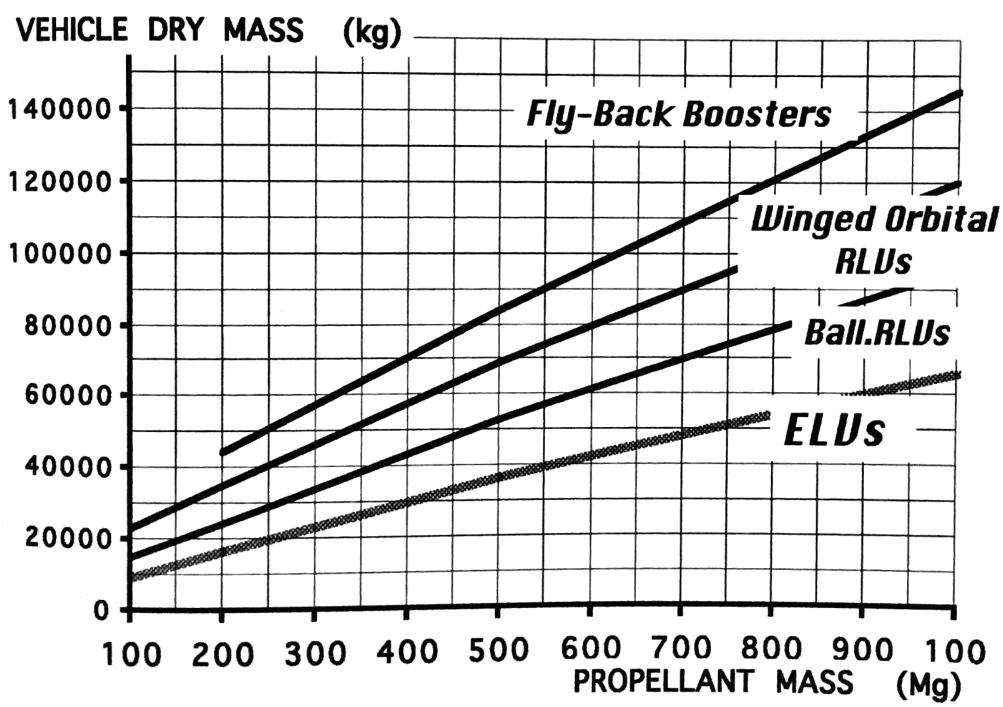


FIG. 2-73 : **Dry Mass of RLV-Concepts vs.Propellant Mass in Comparison to ELVs**

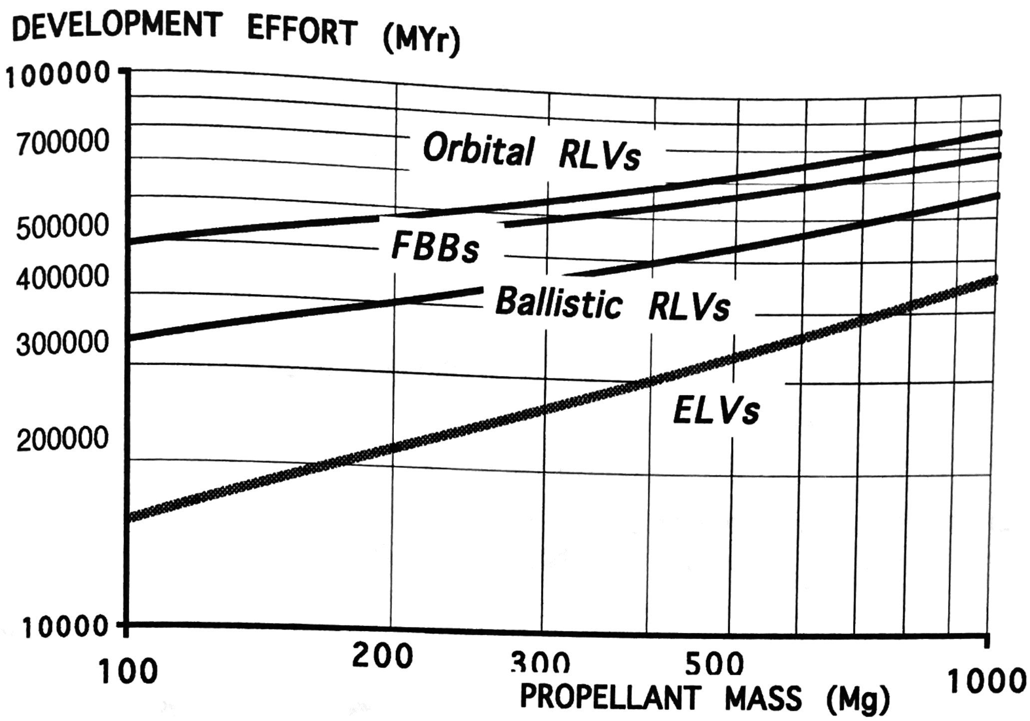


FIG. 2-74 : **Nominal Development Cost Comparison of RLVs vs. ELVs** (excluding engine development)

2.9 Applications of the *TRANSCOST* Development Cost Submodel for Future Project Studies

2.91 Development Cost Comparison of Different RLV Concepts

It is important - but rarely being done - to perform a cost comparison of different launch vehicle options with the same costing methodology and assumptions, before making a concept selection / decision. The TRANSCOST-Model allows to perform such a cost estimation process for different launch system concepts without a detailed subsystem design.

The main RLV options based on present or near-term technology are:

(A) **Winged Rocket Configuration** (VTO-HL) - with vertical take-off and horizontal landing, as the Space Shuttle Orbiter

(B) **Ballistic Rocket Configuration** (VTOL) - with vertical take-off and landing, as tested by the DC-X, the Delta Clipper Demonstrator Vehicle,

(C) **Parallel-staged Winged TSTO Rocket Configuration** - sometimes with equal-sized stages, VTO-HL, (FSSC-9), and

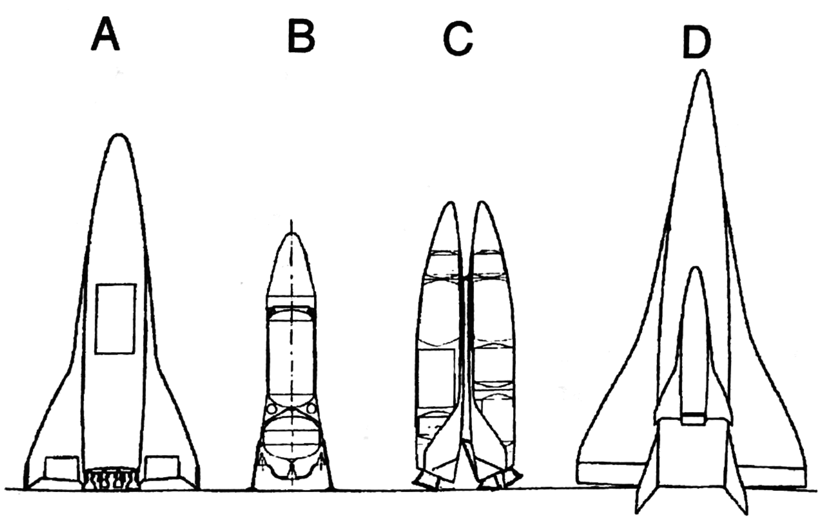
(D) **Horizontally launched TSTO** with airbreathing propulsion in the first stage and rocket propulsion for the second stage - with the German SÄNGER Concept as example.

The concepts are shown in FIG. 2-75 to about the same scale for a payload of 7000 kg to the ISS-orbit. Since most missions are going to higher orbits than just LEO two stages are required normally. Therefore, it is not a question of SSTO vs. TSTO, but whether the first stage should have

- orbital capablity,
- suborbital capability (with flyback provisions), or
- only boost capability (allowing for glide-back to the launch site).

The four RLV options are characterized as follows:

CONCEPT A: The winged RLV with rocket propulsion and orbital capability has an external similarity with the Shuttle Orbiter and implies horizontal landing which is a well demonstrated mode. Wings and the related power supply for the aerodynamic surfaces require the highest launch mass of all concepts, as indicated in FIG. 2-75.



	A	B	C	D
M_0 =	900 Mg	650 Mg	550 Mg	360 Mg
M_{DRY} =	101 Mg	62 Mg	56+29 Mg	132 Mg
H =	62 m	43 m	50 m	75 m

FIG. 2-75: **Major Reusable Launch Vehicle Concepts' (RLVs) Mass and Size**

75. A vehicle with suborbital capability would require additional turbojet engines and design for cruise flight, adding substantially to the dry mass, but reducing the total vehicle size.

CONCEPT B: The ballistic reusable vehicle concept as such is not new - it has been conceived already in 1963/64 by Krafft Ehricke (NEXUS) and Phil Bono (RHOMBUS).

The technology level at that time,as well as the assumed payload size of 450 tons resulted in vehicle sizes of several thousand tons, as did the Chrysler Shuttle option of 1972 (SERV). Nowadays with existing technology and a payload of some 15 tons a ballistic SSTO Launch Vehicle can be built with 500 to 900 tons launch mass, depending on the payload and mission orbit. In 1994/95 the technique of vertical takeoff and landing, as well as a short turnaround time with minimum ground staff was demonstrated by the DC-X experimental vehicle.The ballistic SSTO Vehicle (to LEO) has a lower launch mass and a lower dry mass requires, however, a higher net mass due to the propellant demand for the vertical landing mode.

CONCEPT C: The two-stage winged rocket vehicle has been studied in many variations. Shown is a special design with externally equal-shaped stages which are non-optimum from the performance standpoint, but may reduce development cost. Internally the vehicles have to be different. The first stage in this case has a low suborbital velocity capability, allowing for a glide-back to the launch site.

However, two complete systems have to be developed, plus the mechanical and dynamic interstage problems. The total launch mass is relatively low; however, not the combined dry mass.

CONCEPT D: The launch sytem with horizontal take-off using an advanced aircraft design with turboramjet engines, plus a relatively small winged orbital rocket vehicle as second stage, minimizes the total launch mass but not the vehicle size: due to the use of hydrogen in the first stage with its low density the vehicle is the largest one. The combined dry mass is the highest of all concepts, but it has the greatest mission flexibility by take-off from conventional airfields and its inherent cruise capability of few thousand kilometers.

FIG. 2-76 shows the resulting development costs of the four launch systems of FIG. 2-75 if those are developed under the usual government agencies‘ contract conditions („Business-as-Usual“). The differences are substantial and should normally be a major decision factor instead of (only) personal preferences. By application of cost engineering principles, such as

- -- no performance optimization,
- -- use of existing technologies, and
- -- use of existing rocket engines, if possible,

as well as reduction of the costly „micromanagement“ by the customer‘s organization. the development cost could be reduced to some 50 %, as detailed in ref. 78 for the example of a winged SSTO launch vehicle. In case of a (less probable) industrial-commercial development the costs could be reduced to 30 % or less of the values shown in FIG. 2-76.

The great development cost differences of the launch vehicle options clearly show how important the role of cost engineering is in the selection of the vehicle concept to be developed in the future. For the more conventional TSTO vehicle, the cost will be some 30 % higher than for an SSTO vehicle (in spite of a 40 % lower launch mass). For a winged vehicle with the more conventional horizontal landing mode the development cost will be some 50 % higher than for a ballistic vehicle with vertical

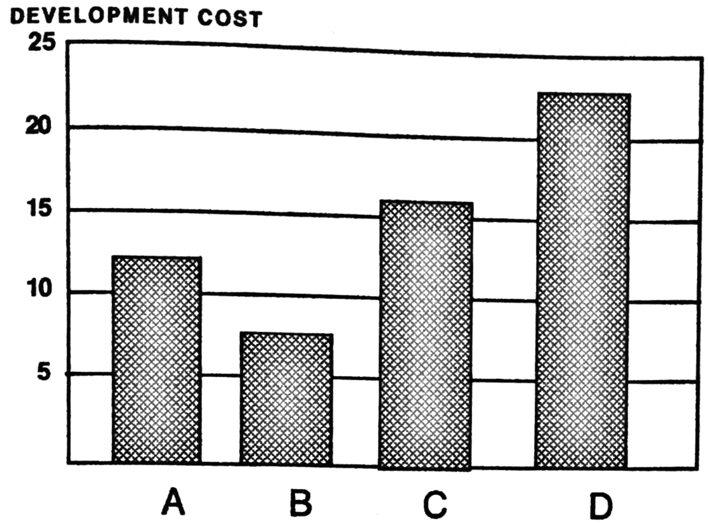


FIG. 2-76 : **Development Costs (Billion USD, 2000) of the four RLV-Concepts shown in FIG. 2-75**

landing by rocket power. Although this has been demonstrated successfully, it meets scepticism because it is not the traditional mode of landing.

For most missions, especially to geostationary transfer orbit or for Earth-escape missions, two-or three-stage vehicles will be required. Therefore, the real question is ***NOT*** SSTO vs. TSTO but whether the first stage should have orbital capability and can return to the launch site without additional fly-back engines. Those features are required in case the first stage is designed for suborbital velocity: in most cases a winged vehicle is proposed with airbreathing turbojet engines for the return flight to the launch site. This, however, results in a complex first stage design which has to fulfill the requirements of a VTO launch vehicle stage and of a cargo aircraft. Experience has shown that the aircraft design requirements are mostly under-estimated by launch vehicle designers, while the launch vehicle problems are normally under-estimated by aircraft designers. In any case it is a complex and expensive vehicle development. In addition there is the problem of stage separation and the increased effort for fly-back operations.

In the third case where a more simple first stage is used just as booster with low delta-V, it can glide-back to the launch site. However, the second stage in this case is relativley large and needs the same systems and technology as a vehicle with direct orbital (LEO) capability. This is then only a matter of sizing.

Avoiding an additional vehicle system development and the inherent problems of vehicle assembly, stage separation and flyback operations of the first stage is certainly a desirable feature. For this reason, and from the cost and economics standpoint, a vehicle with orbital capability is clearly the preferred choice, compared to a two-stage vehicle to LEO (and three-stage beyond).

2.92 Ballistic SSTO vs. TSTO RLV Development Cost Comparison

This analysis is a cost comparison of an SSTO-Vehicle (Single-stage-to-Orbit -LEO) with two possible TSTO Vehicle (Two-stage-to-Orbit - LEO) options: using either Hydrogen / LOX or Kerosene / LOX as propellants (ref .97).

All three vehicles have been sized so that 7 Mg payload can be delivered to the ISS-Orbit (450 km/ 51.6°). In all cases existing rocket engines are being used: the Russian D-57 engine (marketed by Aerojet) for the Hydrogen/LOX vehicles and the NK-33, resp. RD-120 engines for the Kerosene/ LOX option.

The basic SSTO Vehicle design is shown in FIG. 2-73: it is a BETA-type concept with a plug cluster engine arrangement. The TSTO concepts are similar to the Kistler Aerospace K-1 RLV (cf. FIG. 2-64).

The main mass data are shown in TABLE 2-VII. The net mass and dry mass values (without engines) of the Hydrogen/LOX vehicles are consistent with the chart of FIG. 2-33. The mass values for the Kerosene/LOX stages are based on FIG. 6 of ref. 97. The different optimum launch acceleration of 1.27 for TSTO vehicles and 1.4 for the SSTO Vehicle has been taken into account for the number of engines required.

The development cost calculation is made by application of the basic CER for ballistic RLVs as shown in chapter 2.43, plus the system engineering factor of 1.04 for the SSTO vehicle and 1.08 for the TSTO RLVs. The results presented in FIG. 2-78 are based on a reduced BAU-strategy : 50% of the historic cost, as discussed in chapter 2.61. The SSTO-RLV - although being the largest vehicle geometrically and weight-wise - shows <u>the lowest development cost.</u>

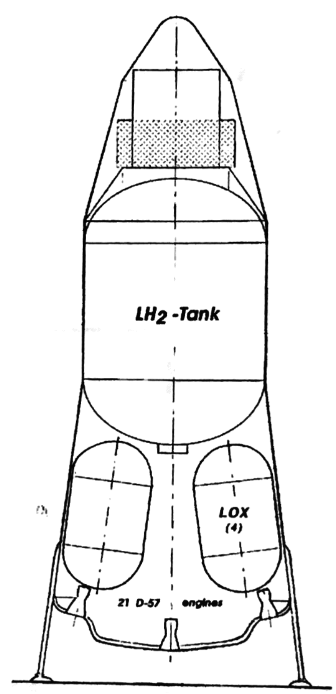


FIG. 2-77: **SSTO-Vehicle Concept (BETA IIA)**

The TSTO Hydrogen/LOX-Vehicle with the lowest GLOW and dry mass comes off with some 36 % higher development cost. The reason is that two different systems have to be developed, plus the stages' integration and in-flight separation. Surprisingly the TSTO Vehicle with the conventional propellants is the most expensive to develop (plus 45 %) . The combined dry mass is higher than for the Hydrogen version and the propellant mass is relatively high due to the lower performance (I_{sp}) of the rocket engines.

TABLE 2-VII: **SSTO and TSTO-RLV Mass Data**

	SSTO-RLV	TSTO Launch Vehicles	
PROPELLANTS:	LOX / LH2	LOX / Kerosene	LOX / LH2
GLOW	456 Mg	405 Mg	241 Mg
PAYLOAD - ISS Orbit	7 Mg	7 Mg	7 Mg
PROPELLANT MASS			
STAGE 1	400 Mg	285 Mg	150 Mg
STAGE 2	—	81 Mg	51 Mg
DRY MASS w/o Engines			
STAGE 1	29 000 kg	17 700 kg	13 700 kg
STAGE 2	—	7 130 kg	6 600 kg

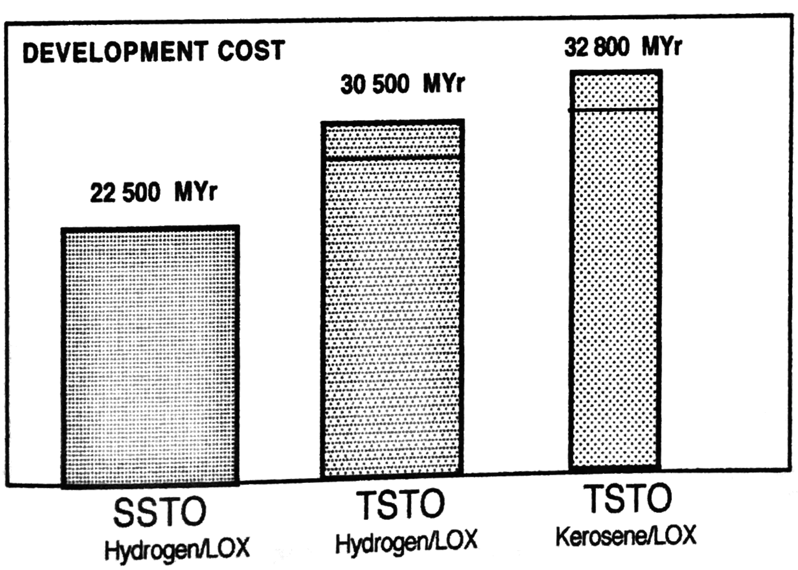


FIG. 2-78: **Development Cost of a Ballistic SSTO-RLV vs. TSTO-RLVs**

2.93 Potential Development Cost Range for a Winged Rocket SSTO Launch Vehicle

Development Cost for the same launch vehicle, resp. a launch system with the same performance, can vary in a wide range. For a more detailed analysis the example of a Winged VTO Rocket Vehicle was chosen as depicted in FIG. 2-79. The reference vehicle (FSSC-1, ref. 78) has a launch mass (GLOW) of 916 Mg and a dry mass of 97 760 kg inclusive the engines' mass of 25 400 kg. New rocket engines were included: a 150 bar staged-combustion cycle engine with 1637 kN SL thrust level. Four of the 8 engines were equipped with larger nozzles to achieve an I_{sp} of 448 sec in vacuum. The total propellant mass (LH_2 /LOX) amounts to 794 Mg (+ 6.8 Mg residuals, margin, RCS propellants). The LEO mass for payload, a CTV or OTV is 17.5 Mg at a 250 km/ 5° orbit.

The TRANSCOST-Model CERs establish the „most probable development cost" under BAU (Business-as Usual) conditions. This means the cost include 15 to 20 % cost for unforeseen technical problems and delays which is usually the case for such vehicle programs. This does not mean that the cost can't be higher: that occurs in case of catastrophic incidents or a major change of specifications.

The most probable BAU development cost according to the TRANSCOST CER's for the vehicle system are

$$H_{VW} = 1421 \cdot M^{0.35} \cdot f_1 \quad f_2 \quad f_3 \quad f_8 \qquad \text{MYr}$$
$$= 1421 \times 72\,350^{0.35} \times 1.2 \times 1 \times 0.9 \times 0.86$$
$$= 66\,270 \text{ MYr,}$$

and for the rocket engine with 3176 kg mass and 600 qualification test runs:

$$H_{EL} = 277 \cdot M^{0.48} \cdot f_1 \quad f_2 \quad f_3 \quad f_8$$
$$= 13\,285 \times 1.0 \times 1{,}07 \times 0.8 \times 0.86$$
$$= 9\,780 \text{ MYr.}$$

Both values added and multiplied with the system integration factor of 1.04 results in total cost of about 79 100 MYr, or 16 Billion Euro (2003) for a European (ESA) development program.

The „Ideal Development Cost" as they would be derived by a „bottom-up" cost estimate or the use of a subsystem based cost model - which would normally be used as the

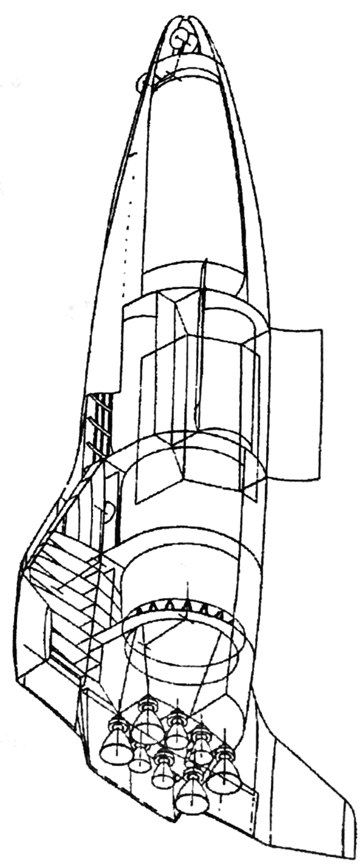


FIG. 2-79 : **Reference Launch System**

„Official Program Cost" at the start of the development would be 15 to 20 % lowe i.e. about 65 000 MYr , or some 13.5 Billion Euros (2003 value).

The concurring NASA Langley estimate in the „Access-to-Space Study" (ref. 98) for a similar Winged SSTO Vehicle (890 Mg GLOW, 790 Mg propellants) with BAU-methodology was **14.5 B.$**(94) = 81 800 MYr, which is eqivalent to 70 400 MYr under European conditions.

The first major cost reduction can be achieved by application of cost engineering, by example, by the use of existing components and subsystems: i.e. the use of existing rocket engines. In fact, the Russian RD-0120 would fit in this case. The engines development cost of 9 780 MYr could be reduced to some 1000 MYr for necessary adaptations. In this case the total (ideal) development cost would be only 58 000 MYr, or 12 Billion Euros (2003).

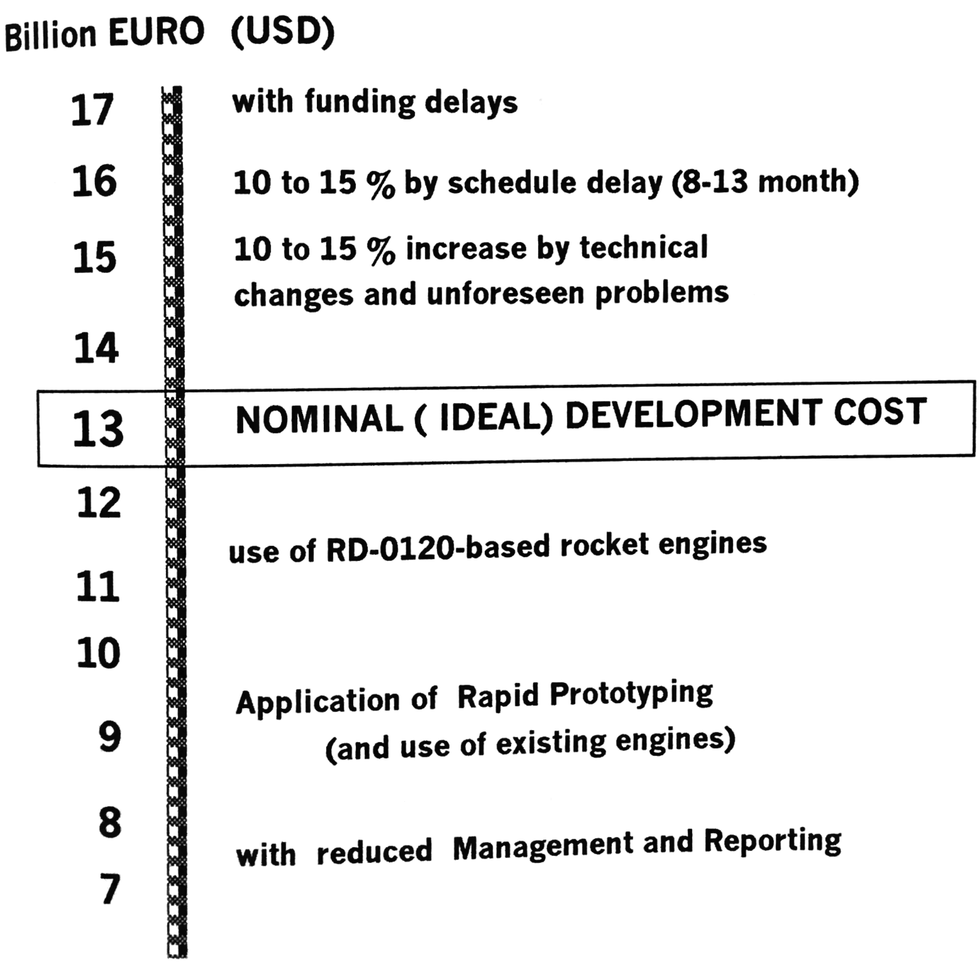


FIG. 2-80 : **Example for the Potential Development Cost Range for a Winged SSTO Launch Vehicle** (ref. 78)

A further development cost reduction is feasible by application of „rapid prototyping", the use of computerized design and integration techniques to some estimated 50 000 MYr or 10 Billion Euros (2002).

Finally, reduced customer „micro-management" and reporting/ review activities as discussed in chapter 2.61 can reduce the required manpower to some estimated 36 000 MYr or 7.2 Billion Euros. This is 52 % of the „Ideal Development Cost", in agreement with the examples mentioned in chapter 2.61.

In summary, the development cost of a large winged SSTO launch vehicle in Europe (under ESA contract) could require **between 16 and 7 Billion Euro**, depending on the quality of project preparation, the application of cost engineering, the development and test strategy and the management / contract conditions of the customer.

In case of a purely commercial / industrial development of such a winged SSTO Vehicle in the United States (taking into account the different MYr costs and productivity) the cost could be further reduced down to a level of 21 500 to 24 000 MYr, equivalent to **4.5 to 5 Billion US$ (2000).** This is the cost range which has been indicated by Lockheed Martin for the commercial development of the „Venture Star" Project ; however, the more complex nature and technology required for the lifting body configuration with a linear aerospike engine system (compared to the more conventional reference configuration of FIG. 2-79) would make realistic commercial development cost of 6 to 7 Billion $ in this case more probable.