General performance data — Chapter 2

2.1. Introduction

This section provides the information necessary to make preliminary performance assessments for the launch vehicles of the ARIANE 4 family.

The ARIANE 4 launch vehicle is able to meet the need of a wide range of missions from Low Earth Orbits to escape trajectories.

Over the past ten years ARIANE 4 has set up a high degree of reliability and excellence in particular for the launch of geostationary satellites into transfer orbit.

The data presented here reflects this flight proven hardware / software performance knowledge for the major missions flown by ARIANE.

This document is a guide for mission design, and predictions are presented with a slight degree of conservatism to make sure that the launch vehicle will meet the expectations.

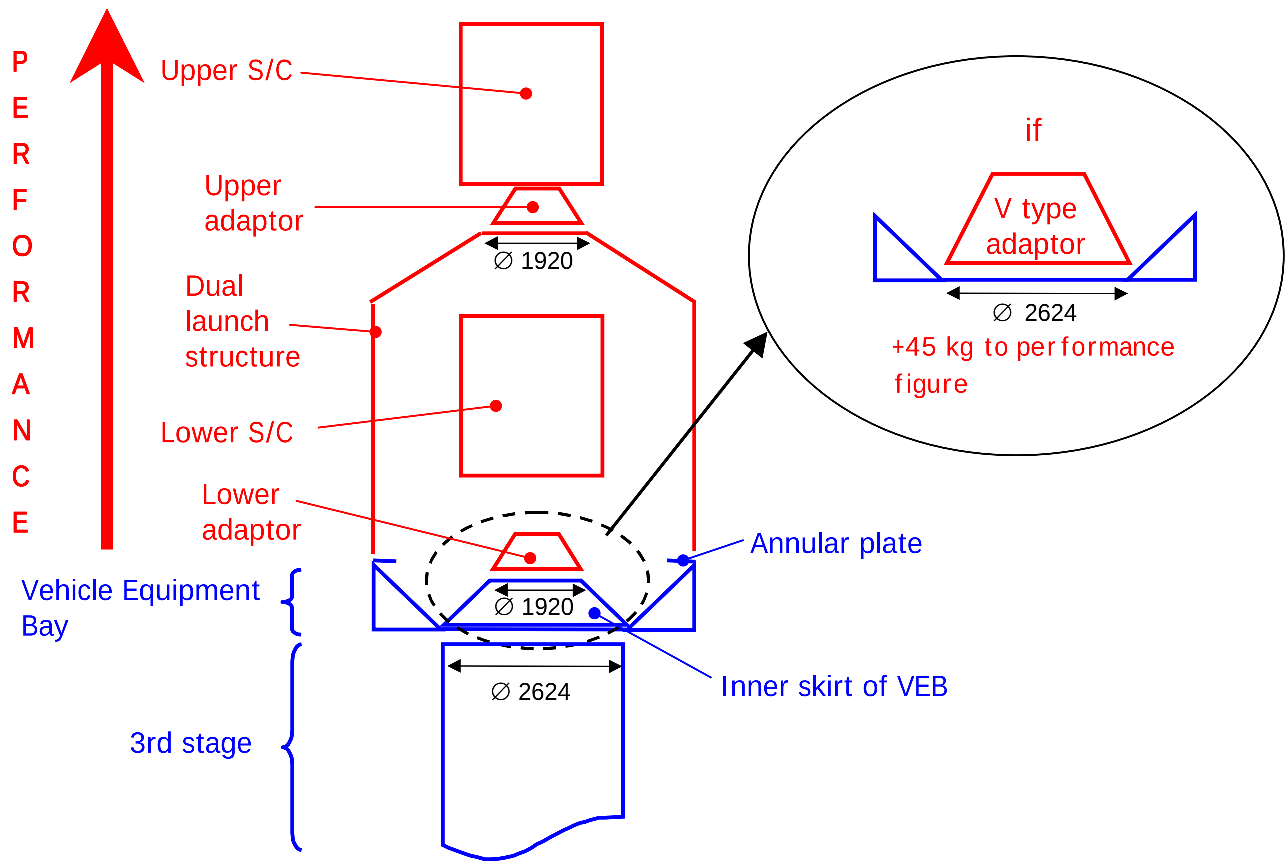
Beyond these performance figures, customized and innovative methods and solutions can also be proposed to Customer, to individually optimize their mission or even increase the launch vehicle standard capability.

ARIANE 4 can perform the launch in the single payload configuration, that allows to specifically tailor the mission to the customer need, or, through the use of various carrying structures, in the dual payload configuration that has demonstrated throughout the years to be a unique way to rapidly and economically deploy satellites onto the geostationary transfer orbit.

2.2 Performance definition

The performance figures given in this chapter are expressed in terms of payload mass. The mission performance includes the mass of :

- the spacecraft(s)
- the dual launch system (if used), which mass is mission dependant and approximately of :
  - mini SPELDA 320 kg
  - stretched mini SPELDA 350 kg
  - short SPELDA 410 kg
  - SYLDA 4400 200 kg
- the multiple launch system, dispenser (if used) : mass, mission-specific
- the adaptor(s) : see annexes for the mass characteristics. In case of use of a V type adaptor, which replaces part of the vehicle equipment bay structure, 45 kg has to be added to the performance figure



Performance computations are based on the following main assumptions :

- Third stage carrying sufficient propellant to reach the targeted orbit with the specified probability of 99% except otherwise specified
- Aerothermal flux at fairing jettison less or equal to 1135 W/m²
- Altitude values given with respect to a spherical earth radius of 6378 km
- Launch from the CSG (French Guiana), taking into account the relevant safety requirements

2.3 Launcher propellant reserve management

The reference performance corresponds to a 99% probability to reach the targeted orbit before depletion of one of the third stage propellants.
A certain amount of these propellants loading called the "performance reserve" accounts for statistical dispersions of the launch vehicle parameters.
It can be decided to make use of part of this "reserve" to impart an additional delta-velocity to the payload. This allow :

- to inject a higher mass on the standard GTO than would otherwise be possible (see paragraph 2.4.2). This is particularly interesting for a payload which mass is higher than the launch vehicle performance.
- to increase the apogee altitude compared to the reference 99% mission in the case of a sub-synchronous or super-synchronous injection (see paragraphs 2.5.3 and 2.5.4)

The use of the “performance reserve” reduces the probability of third stage non-depletion cut-off and may lead to stage exhaustion before reaching the targeted orbit (or, in other words, before guidance instruction to cut thrust). The final injection apogee can therefore be lower than predicted, with a minimum equal to the 99% reference performance apogee, the loss being directly proportional to the propellant deficit. If a lower than GTO apogee is achieved, the spacecraft then performs a subsynchronous transfer procedure (see paragraph 2.5.4) raising the apogee with a Perigee Velocity Augmentation (PVA) maneuver.

2.4 Geosynchronous transfer performance

2.4.1 Ariane standard Geostationary Transfer Orbit (GTO)

Most of the communications satellites in orbit have been launched by ARIANE into the Geostationary Transfer Orbit (GTO). These satellites have benefited of the unique location of the Kourou Europe Spaceport : its low latitude minimizes the satellite on-board propellant needed to reach the equatorial plane. Based on the satellite constraints, the resulting optimized dual launch ARIANE 4 standard Geostationary Transfer Orbit, defined in terms of osculating parameters at injection, is :

- Inclination i = 7 °
- Altitude of perigee Zp = 200 km
- Altitude of first apogee Za = 35786 km
- Argument of perigee ω = 178 °

Injection is defined as the end of third stage thrust-decay following shut down.

The above mentioned orbital parameters are not significantly influenced by events occurring between the third stage shut down and the spacecraft separation.

The longitude of the first descending node, as defined in the figure 2.4.1.a, depends upon the particular ARIANE 4 configuration and lies around 7° West.

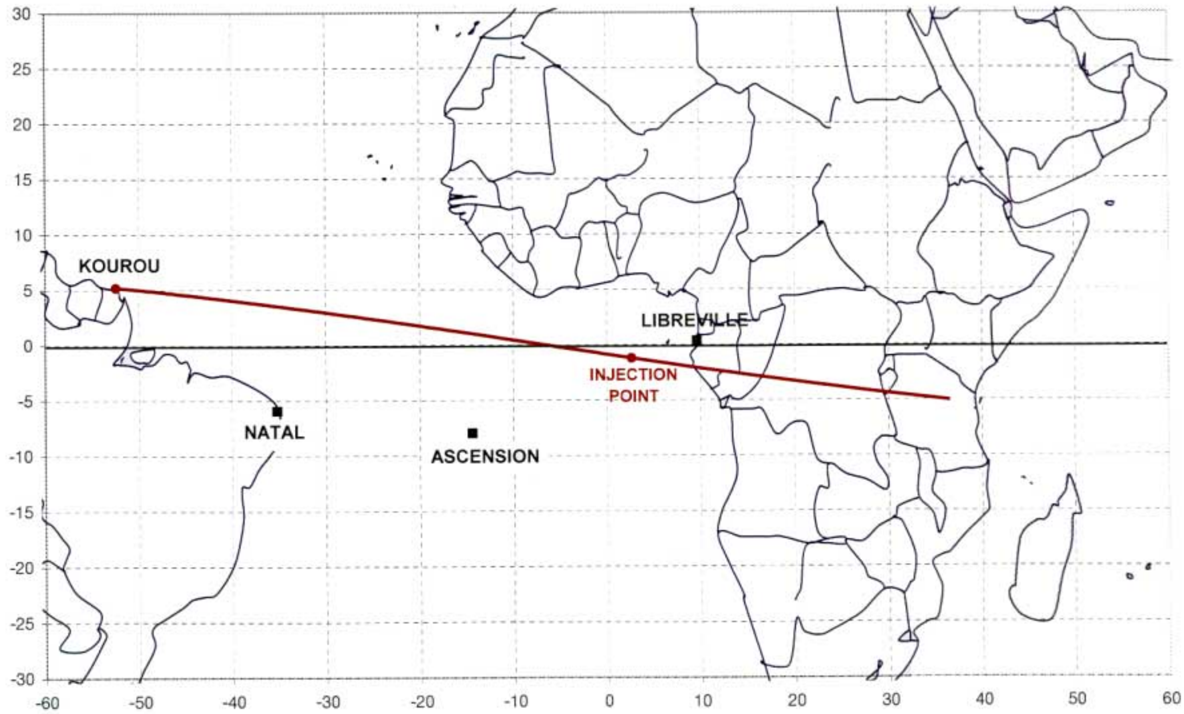


Figure 2.4.1.a : Typical ground track

2.4.2 Performance on the standard Geostationary Transfer Orbit

Through extensive flight experience, enlarged at each flight through a precise monitoring of all systems and subsystems, Arianespace has developed an intimate knowledge of the vehicle and of its performance. Therefore systems performance assumptions can be adjusted to less conservative but flight-demonstrated set of parameters for the Customer benefit. Arianespace is therefore proposing as standards, beside the 99% probability reference performance, the 95% and 50% improved performance data.

Actual mission may be defined to other values depending on customer need.

In order to protect inhabited areas of French Guyana, safety criteria taking into account the wind speed and direction, the nature and quantity of toxic products on board of the launch vehicle and spacecraft, the destruction devices set up, may constrain the launch azimuth and therefore the performance.
Performance values are presented for three spacecraft propellant masses per tank, of the most critical product. It corresponds to the maximum mass contained in the largest individual tank (S/C propellant loading can be higher depending on the number of tanks).

	600 kg oxidizer or 450 kg hydrazine			1200 kg oxidizer or 1100 kg hydrazine			1800 kg oxidizer or 1800 kg hydrazine		
Probability	**99 %**	**95 %**	**50 %**	**99%**	**95 %**	**50 %**	**99 %**	**95 %**	**50 %**
A44L*	4768	4810	4894	4748	4790	4874	4660	4702	4786
A44LP	4290	4327	4398	4270	4307	4378	4167	4204	4275
A42L	3591	3623	3689	3581	3613	3679			
A44P	3465	3496	3561	3446	3477	3542			
A42P	2891	2922	2986						
A40	2174	2213	2268						

* For dual launch, subject to payload compartment configuration [+ 0, - 50 kg]

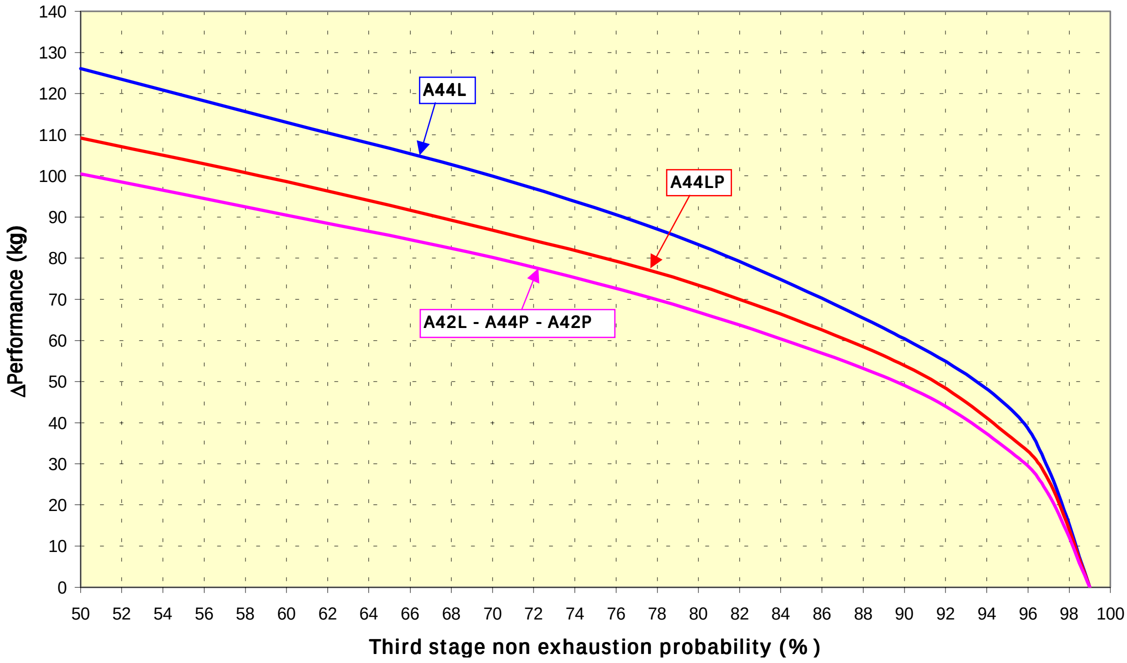


Figure 2.4.2.a - Performance gain versus probability of third stage non exhaustion

Probability (%)	Δmass (kg)		
	A44L	A44LP	A42L
99,00	0	0	0
97,00	29	26	23
95,00	44	37	34
90,00	60	54	49
80,00	83	73	67
70,00	100	87	80
60,00	113	99	90
50,00	126	109	100

2.4.3 Injection Accuracy

This paragraph applies for guidance commanded shut-down.

The covariance matrix of injection errors on the orbital parameters is as follows (for standard geostationary transfer orbit) :

	a	e	i	ω	Ω
a	676	0.00751	-0.01202	-0.0665	0.0812
e		$8.41\ 10^{-8}$	$-1.01\ 10^{-7}$	$-2.47\ 10^{-7}$	$7.82\ 10^{-7}$
i			$3.20\ 10^{-4}$	0.00250	-0.00251
ω				0.0202	-0.0201
Ω					0.0202

Nota : this matrix is applicable for ARIANE 44LP. However values differ very little from one launch vehicle version to the other.
Following standard deviation values are to be used for all L/V configurations :

a	semi major axis (km)	26
e	eccentricity	0.00029
i	inclination (°)	0.018
ω	perigee argument (°)	0.14
Ω	ascending node (°)	0.14
Zp	perigee Altitude (km)	1.0
Za	apogee Altitude (Km)	52

Parameter Mo (mean anomaly at injection) depends on the flight sequence and its accuracy is not correlated with the other parameters.

2.4.4 Typical launch vehicle trajectory

Figures 2.4.4.a to 2.4.4.c show the main trajectory parameters as a function of flight time, for the ARIANE 44LP.

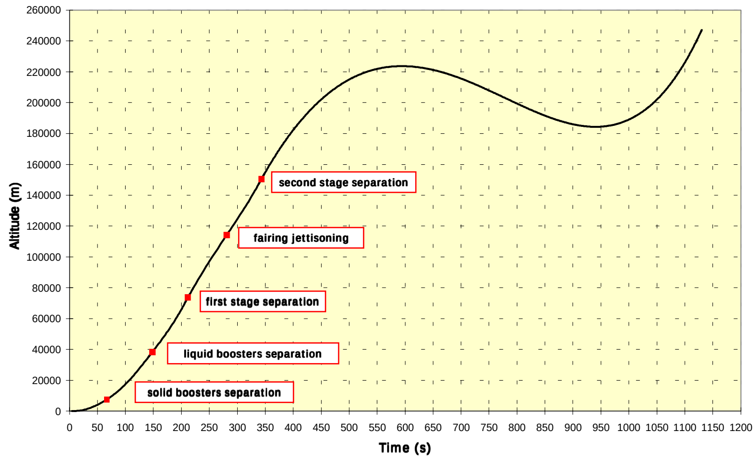


Figure 2.4.4.a - Ariane 44LP trajectory - Altitude versus Time

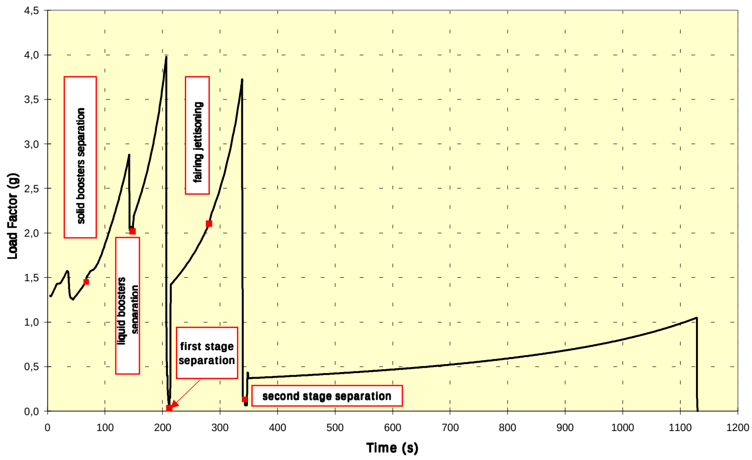


Figure 2.4.4.b - Ariane 44LP trajectory - Load Factor versus Time

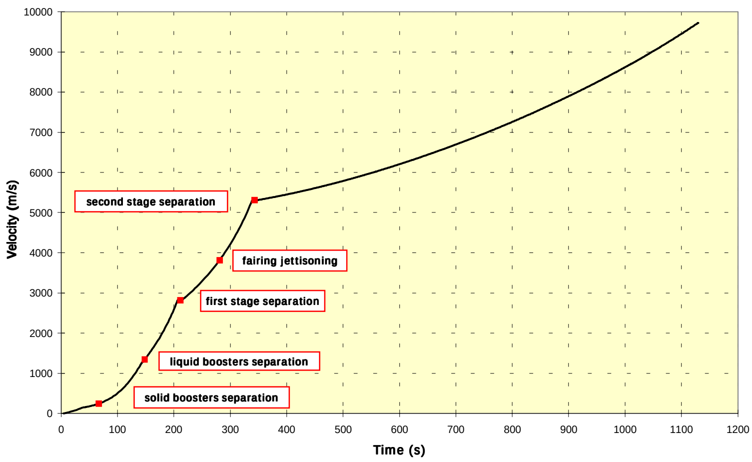


Figure 2.4.4.c - Ariane 44LP trajectory - Relative Velocity versus Time

2.5 Tailored GTO mission

Although the ARIANE 4 standard Geostationary Transfer Orbit, with its low inclination, does already provide a good answer to achieve the lifetime requirement of the Customer, Arianespace can propose a wide range of options to further increase the spacecraft beginning of life propellant mass and Customer's revenue.

2.5.1 Customized mission

The ARIANE mission can be customized by some adjustments to the transfer orbit or to the launch vehicle upper part (vehicle equipment bay and adaptors) :

- Aerothermal flux at fairing jettisoning can be set differently from the standard. The figure 2.5.1.a shows the influence of the aerothermal flux at fairing jettisoning on the performance and on the jettisoning time. The "zero" reference corresponds to the standard 1135 W/m² value. Increasing this value, reduces the time during which the fairing is carried and therefore increases the final launch vehicle performance.

- The perigee can be reduced down to 180 km, the associated gain of performance is shown on figure 2.5.1.b

- Depending on the flight configuration single vs. dual launch, the launch vehicle payload compartment can be customized. For instance, removing the stiffening annular plate of the Vehicle Equipment Bay external cone in dual launch (when it is unnecessary) provides a gain of 20 kg.

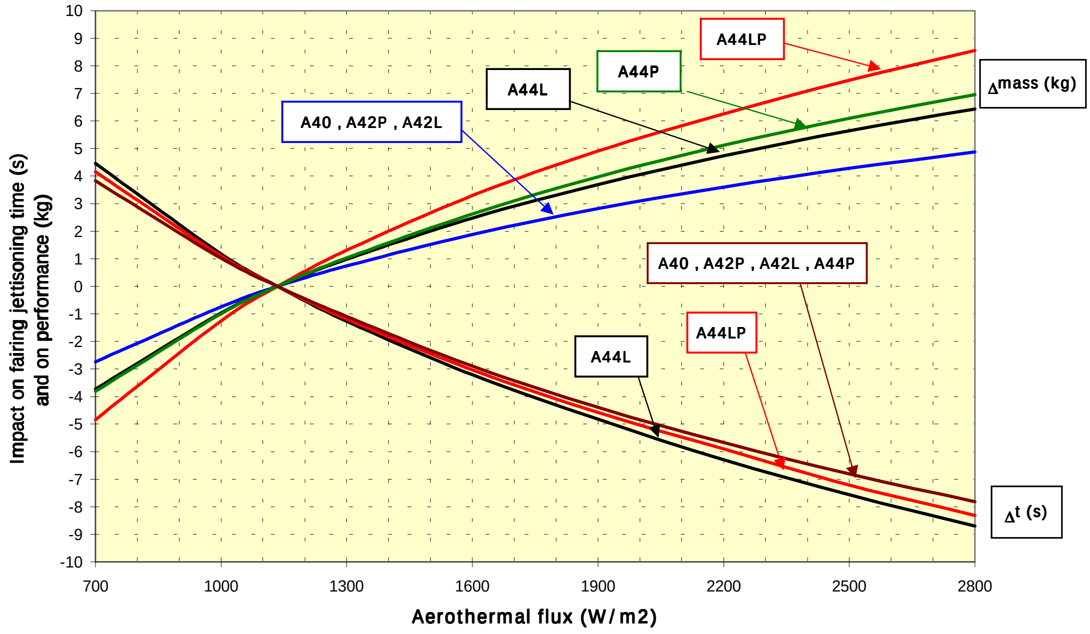


Figure 2.5.1.a - Impact of aerothermal on fairing jettisoning time and on performance

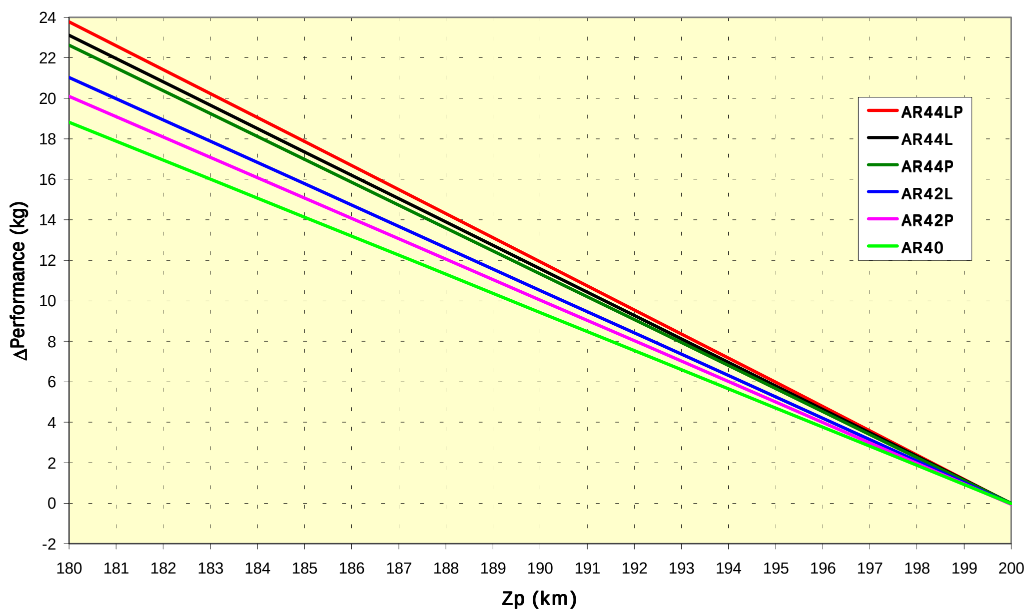


Figure 2.5.1.b - Ariane 4 - Performance gain versus perigee altitude
Za = 35786 km - i = 7° - ω = 178°

2.5.2 Altitude of perigee, argument of perigee and inclination adaptation

This launch strategy can be proposed when the spacecraft mass is below the standard GTO performance of the launcher version considered. Spacecraft lifetime can be increased by furthermore reducing the transfer orbit inclination and / or increasing the perigee altitude. The spacecraft propellant mass necessary to circularize the orbit and lower the inclination is accordingly reduced for the benefit of the spacecraft lifetime.
The curves 2.5.2.a to 2.5.2.e present the launch vehicle performance for perigee ranging from 200 to 250 km and inclination from 3 to 7°.

Argument of perigee and inclination can also be optimized to customer needs. The curve 2.5.2.f shows the performance impact for inclinations from 1 to 10° and argument of perigee from 10 to 230°.

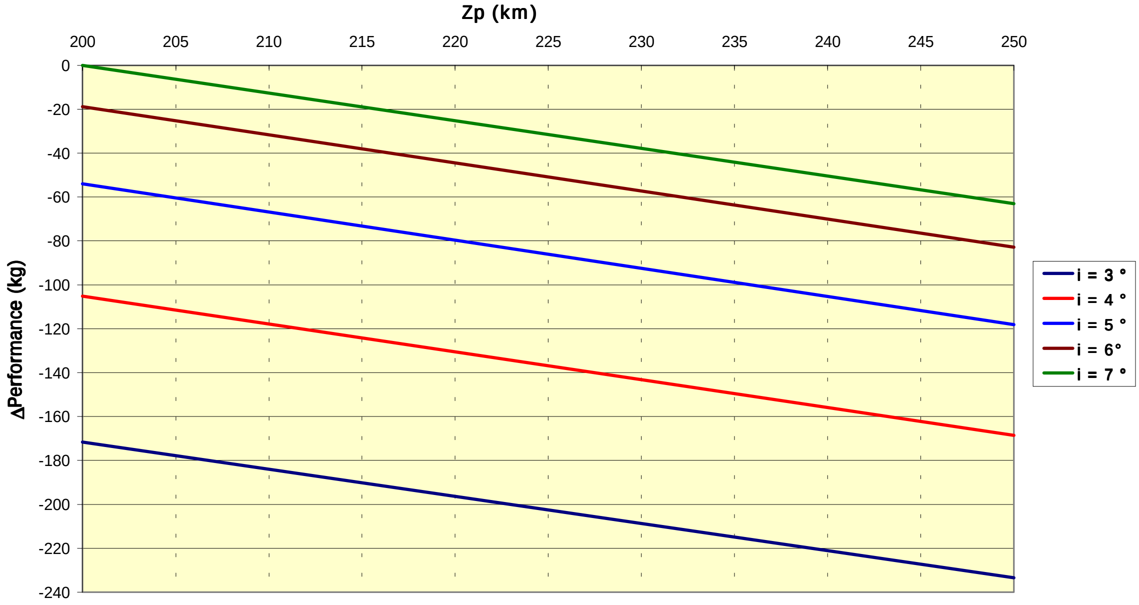


Figure 2.5.2.a - Ariane 44L - Performance impact versus perigee altitude
Za = 35786 km at injection, ω = 178 °

A44L		
Inclination (°)	Zp (km)	Δmass (kg)
3	200	-172
3	250	-233
4	200	-105
4	250	-169
5	200	-54
5	250	-118
6	200	-19
6	250	-83
7	200	0
7	250	-63

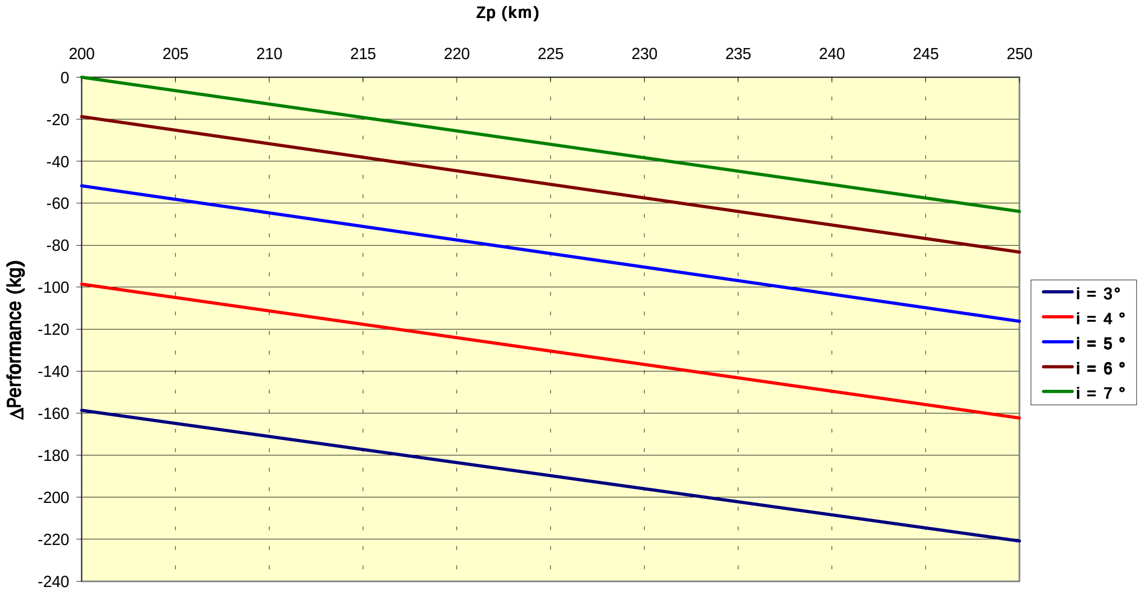


Figure 2.5.2.b - Ariane 44LP - Performance impact versus perigee altitude
Za = 35786 km at injection, ω = 178 °

A44LP		
Inclination (°)	Zp (km)	Δmass (kg)
3	200	-159
3	250	-221
4	200	-99
4	250	-162
5	200	-52
5	250	-116
6	200	-19
6	250	-83
7	200	0
7	250	-64

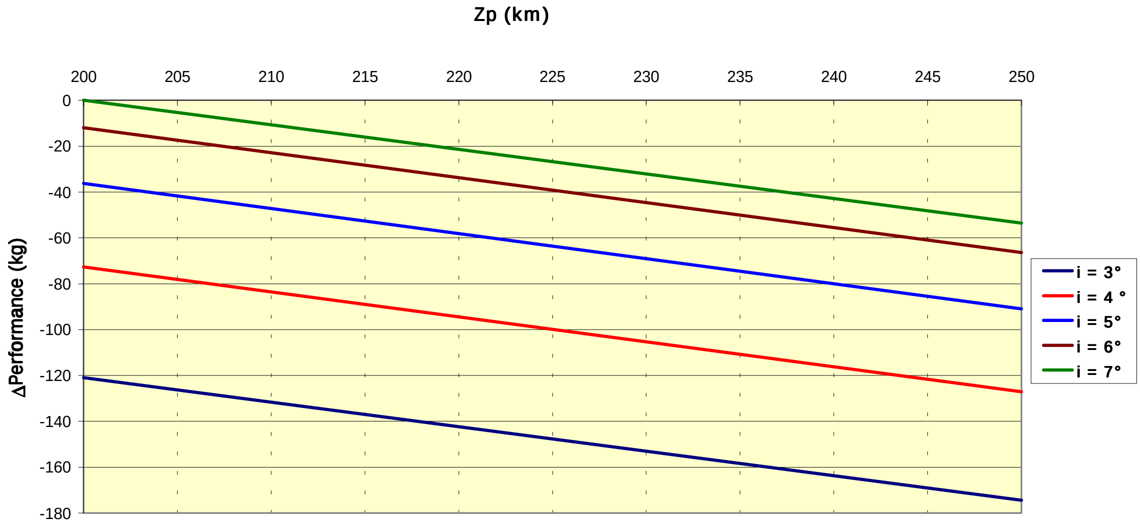


Figure 2.5.2.c - Ariane 42L - Performance impact versus perigee altitude
Za = 35786 km at injection, ω = 178 °

A42L		
Inclination (°)	Zp (km)	Δmass (kg)
3	200	-121
3	250	-174
4	200	-73
4	250	-127
5	200	-36
5	250	-91
6	200	-12
6	250	-66
7	200	0
7	250	-54

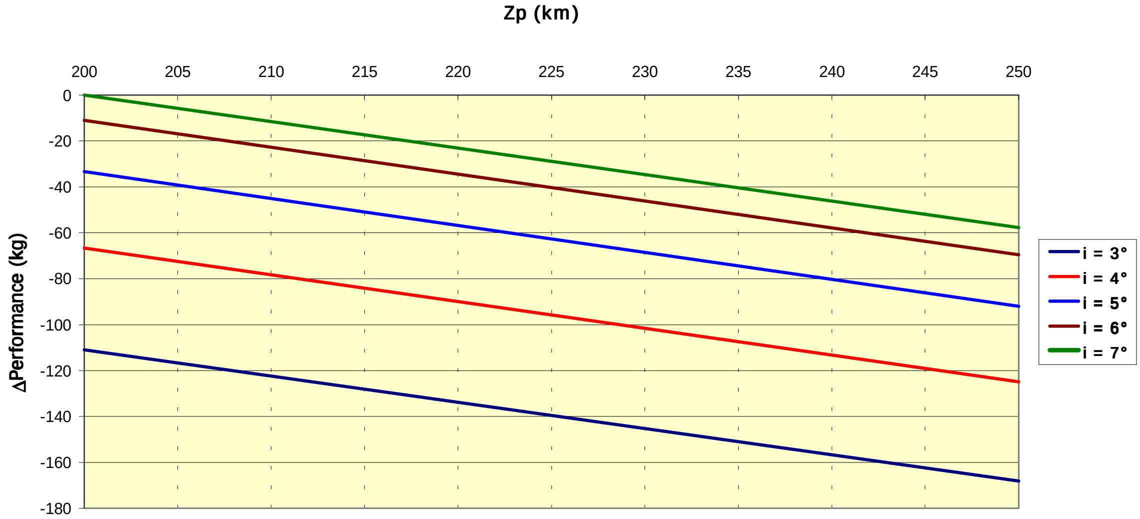


Figure 2.5.2.d - Ariane 44P - Performance impact versus perigee altitude
Za = 35786 km at injection, ω = 178 °

A44P		
Inclination (°)	Zp (km)	Δmass (kg)
3	200	-111
3	250	-168
4	200	-67
4	250	-125
5	200	-33
5	250	-92
6	200	-11
6	250	-70
7	200	0
7	250	-58

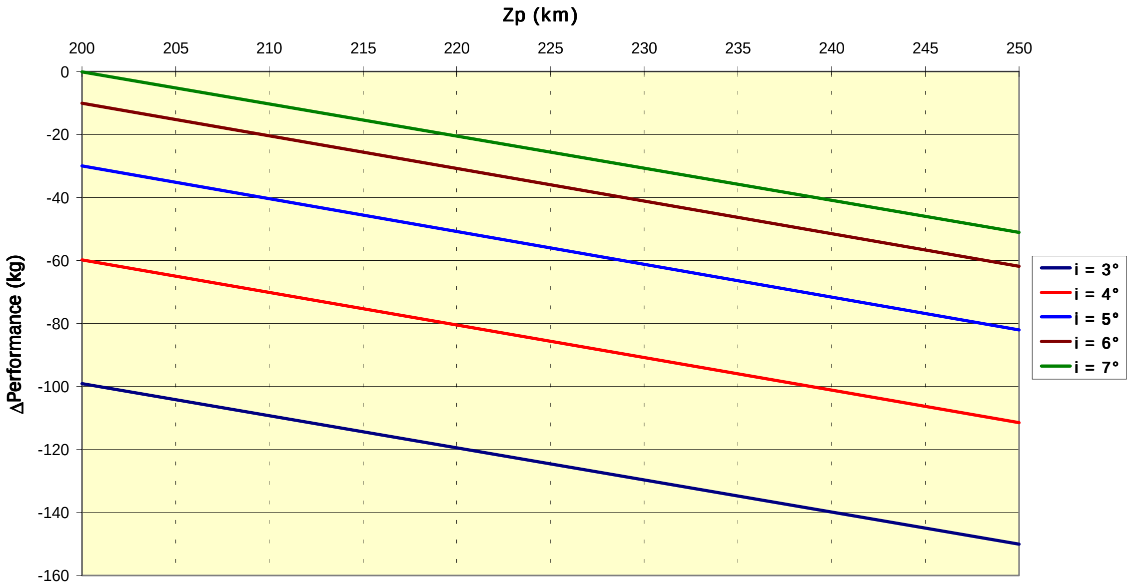


Figure 2.5.2.e - Ariane 42P - Performance impact versus perigee altitude
Za = 35786 km at injection, ω = 178 °

A42P		
Inclination (°)	Zp (km)	Δmass (kg)
3	200	-99
3	250	-150
4	200	-60
4	250	-111
5	200	-30
5	250	-82
6	200	-10
6	250	-62
7	200	0
7	250	-51

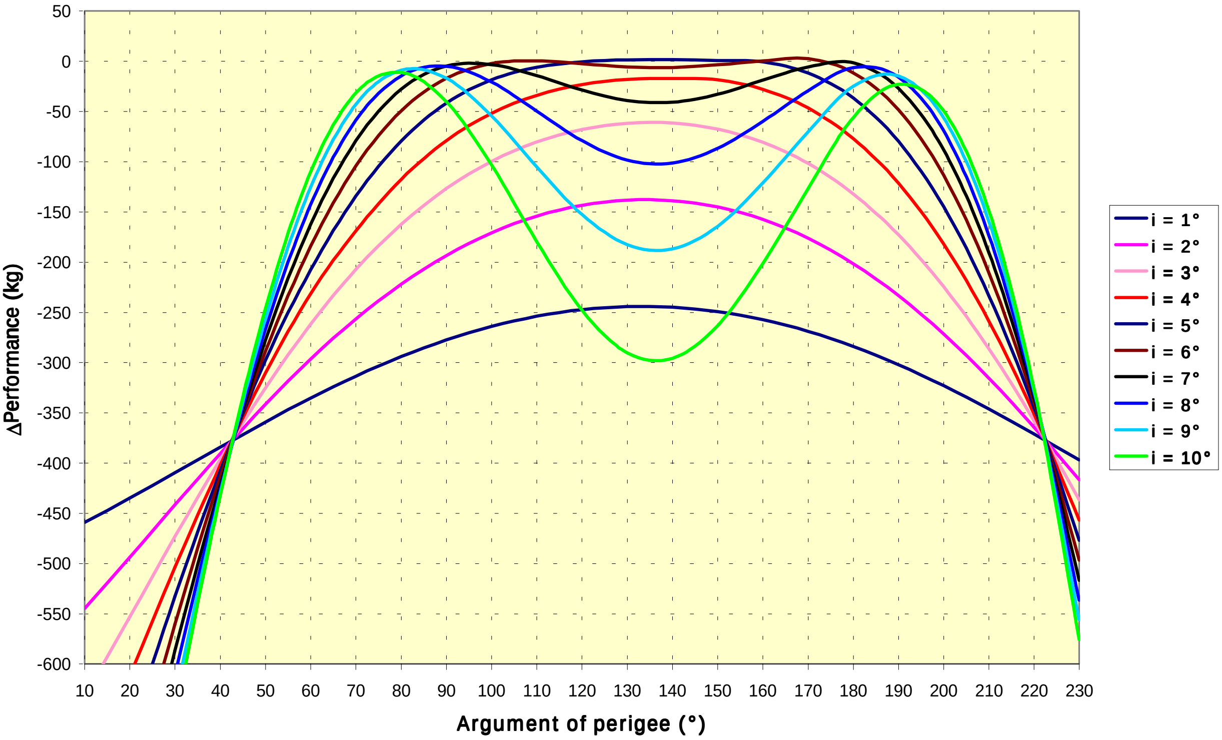


Figure 2.5.2.f - Ariane 44L - Performance versus argument of perigee

	Δmass (kg)									
ω (°)	i = 1°	i = 2°	i = 3°	i = 4°	i = 5°	i = 6°	i = 7°	i = 8°	i = 9°	i = 10°
10	-459	-545	-633	-721	-809	-892	-971	-1041	-1103	-1155
20	-435	-494	-553	-612	-668	-723	-773	-820	-861	-898
30	-410	-442	-473	-503	-533	-560	-586	-611	-635	-655
40	-384	-390	-396	-402	-407	-413	-421	-426	-429	-432
50	-359	-341	-325	-310	-297	-288	-277	-266	-256	-246
60	-335	-296	-262	-231	-208	-184	-162	-143	-125	-110
70	-313	-256	-207	-169	-134	-104	-79	-59	-43	-32
80	-294	-222	-162	-117	-79	-49	-28	-14	-9	-11
90	-277	-193	-127	-79	-42	-17	-5	-5	-17	-40
100	-264	-171	-100	-52	-18	-2	-3	-21	-54	-103
110	-254	-154	-80	-34	-6	1	-14	-50	-105	-180
120	-247	-143	-68	-23	0	-2	-29	-79	-153	-248
130	-244	-138	-62	-18	1	-6	-39	-98	-183	-290
140	-245	-139	-62	-17	2	-6	-41	-101	-187	-296
150	-249	-145	-68	-18	1	-3	-33	-87	-164	-263
160	-257	-157	-81	-28	-1	0	-19	-60	-121	-201
170	-269	-176	-102	-47	-12	3	-5	-29	-70	-127
180	-284	-201	-132	-77	-37	-11	-2	-6	-25	-56
190	-302	-233	-173	-121	-80	-48	-27	-16	-15	-23
200	-323	-272	-224	-182	-145	-114	-88	-69	-56	-49
210	-346	-316	-286	-259	-233	-211	-191	-173	-160	-149
220	-371	-364	-358	-351	-345	-340	-335	-331	-328	-326
230	-397	-417	-437	-457	-477	-497	-517	-536	-556	-576

2.5.3 Supersynchronous transfer

Supersynchronous transfer can be proposed when the spacecraft mass is below the standard GTO performance. Spacecraft lifetime can be increased by reducing the propellant needed to circularize the orbit and lower the inclination. These maneuvers are performed at a higher than geosynchronous altitude, where the velocity is decreased. The curves on figure 2.5.3.a present the ARIANE 4 performance for apogee altitude ranging from 36 000 to 100 000 km. The others parameters of the orbit remain unchanged :

Inclination	7 °
Perigee altitude	200 km
Argument of perigee	178 °

2.5.4 Subsynchronous transfer

Subsynchronous transfer consists in injecting the spacecraft on a transfer orbit with an apogee lower than geosynchronous. The ARIANE 4 performance curves for apogee altitude between 20 000 km and 36 000 km are presented in figure 2.5.4.a.

This strategy is of interest when the satellite mass exceeds the launch vehicle performance on the standard GTO and when the spacecraft tank capacity is large with respect to the standard GTO injection and lifetime needs. The spacecraft propulsion system is used at the perigee of transfer orbit to raise the apogee, this maneuver being known as the Perigee Velocity Augmentation (PVA). The overall propulsion budget of the mission translates in a benefit for the spacecraft in terms of lifetime (for a given dry-mass) or in terms of dry mass (for a given lifetime) compared to the standard GTO injection limited to the launch vehicle capability.

For a given satellite mass, it can be decided to make use of the part of the "performance reserve" (see paragraph 2.3) to increase the apogee altitude compared to the reference 99% mission.

Flight 114 (December 1998) gives a pertinent illustration of the above :

- 4217 kg required performance on an ARIANE 42L, with a predicted apogee altitude at launcher injection of :
  - 19459 km on a 99% basis
  - 21606 km on a 50% basis
- the apogee altitude achieved was 21586 km (commanded cut-off)

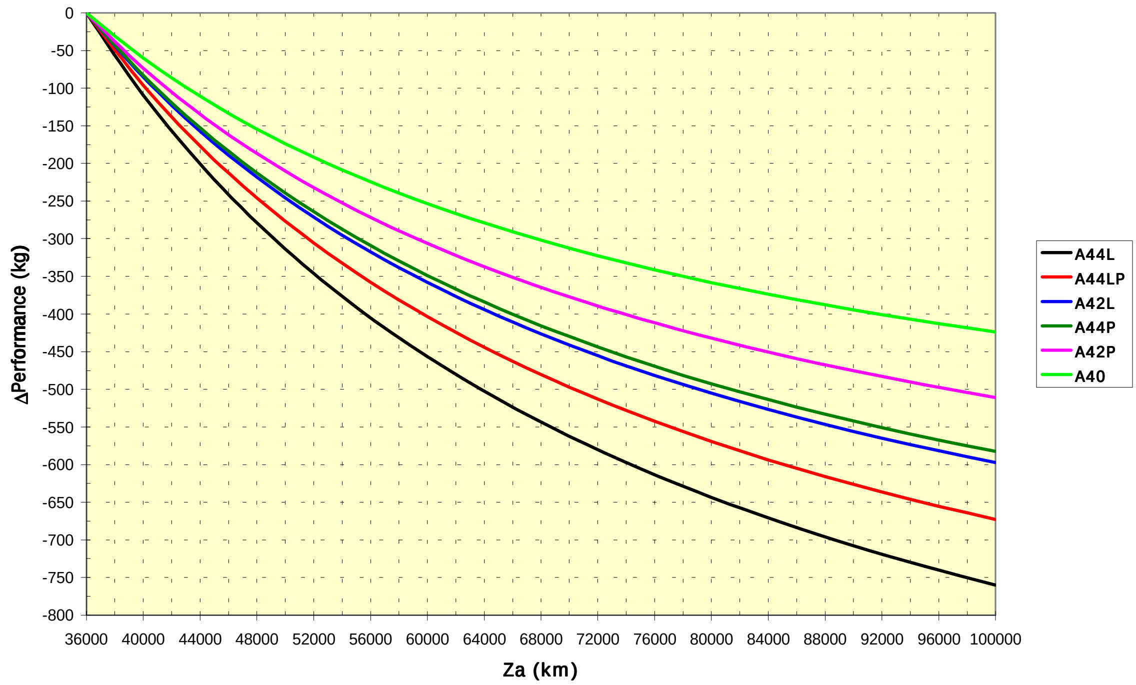


Figure 2.5.3.a - Ariane 4 - Impact on performance of the increase of the apogee altitude
Zp = 200 km, i = 7°, ω =178°

	Δmass (kg)					
Za (km)	A40	A42P	A44P	A42L	A44LP	A44L
36000	0	0	0	0	0	0
40000	-60	-73	-83	-85	-96	-109
44000	-111		-153	-157	-177	
45000		-148				-221
48000	-154		-212	-218	-246	
50000		-210				-314
52000	-192		-264	-271	-306	
55000		-262				-391
56000	-225		-309	-317	-358	
60000	-253	-306	-349	-358	-404	-457
64000	-279		-384	-394	-444	
65000		-344				-513
68000	-302		-416	-426	-480	
70000		-377				-562
72000	-323		-444	-455	-513	
75000		-406				-605
76000	-341		-469	-482	-543	
80000	-358	-432	-493	-505	-569	-643
84000	-374		-514	-527	-594	
85000		-455				-677
88000	-388		-533	-547	-616	
90000		-475				-708
92000	-401		-551	-565	-636	
95000		-494				-735
96000	-413		-567	-582	-655	
100000	-424	-511	-583	-597	-673	-760

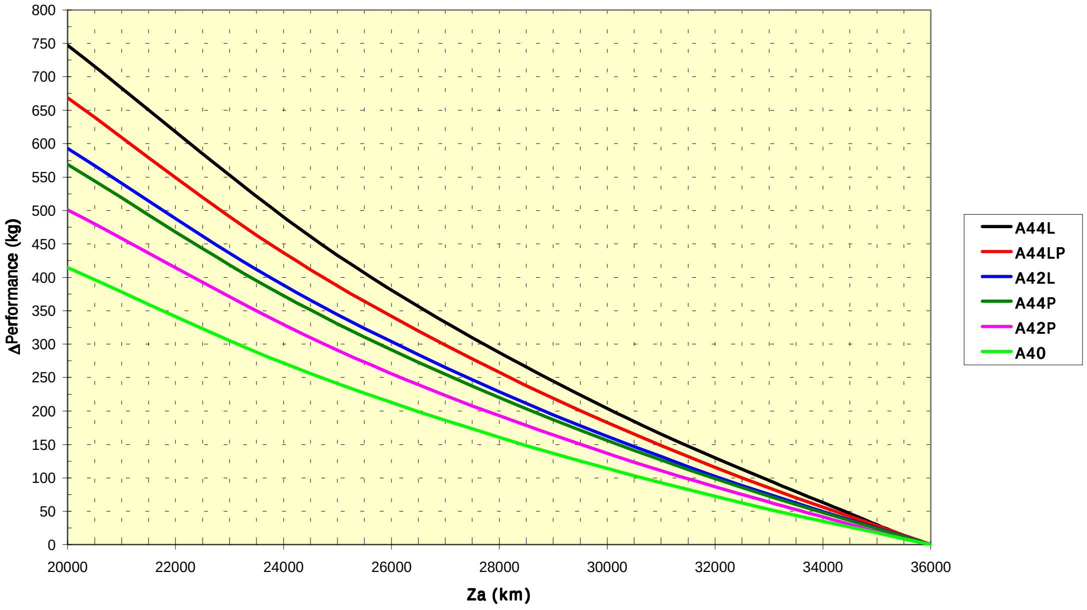


Figure 2.5.4.a - Ariane 4 - Gain of performance versus decrease of apogee altitude
Zp = 200 km, i = 7°, ω =178°

	Δmass (kg)					
Za (km)	**A44L**	**A44LP**	**A42L**	**A44P**	**A42P**	**A40**
20000	747	669	593	569	501	415
25000	433	437	388	372	291	272
30000	204	258	229	220	137	161
35000	30	116	103	99	19	72
36000	0	0	0	0	0	0

2.5.5 ARIANE 4 performance enhancements example

Flight 113 October 1998

Demonstrated performance **4947 kg** with ARIANE 44L, achieved through :

- inclination and perigee altitude adaptation :
  i = 6.5°
  Zp = 185 km
- removal of the stiffening annular plate on the external cone of the vehicle equipment bay (this piece of structure can only be removed in case of dual launch)
- deletion of one battery out of three. The third battery is not needed when the vehicle equipment bay is equipped with gyrolaser type inertial platforms
- engine tuning (slight increase of chamber pressure, mixture ratio and propellant loading optimization) associated with hardware availability and operational constraints
- flight performance reserve associated with a probability to reach the intended orbit before propellant exhaustion of 97 % (99 % as a reference)

2.6 Sun Synchronous Orbit (SSO)

The launch vehicle performance is presented on figure 2.6.b for circular orbits ranging from 400 to 1400 km.
For each altitude the corresponding sunsynchronous inclination is presented in figure 2.6.a.

Performance computations are based on the following assumptions :

- Aerothermal flux at fairing jettison less than 500 W/m²
- Launch azimuth of 0° (due North)
- Inertial node control on a 20 min launch window

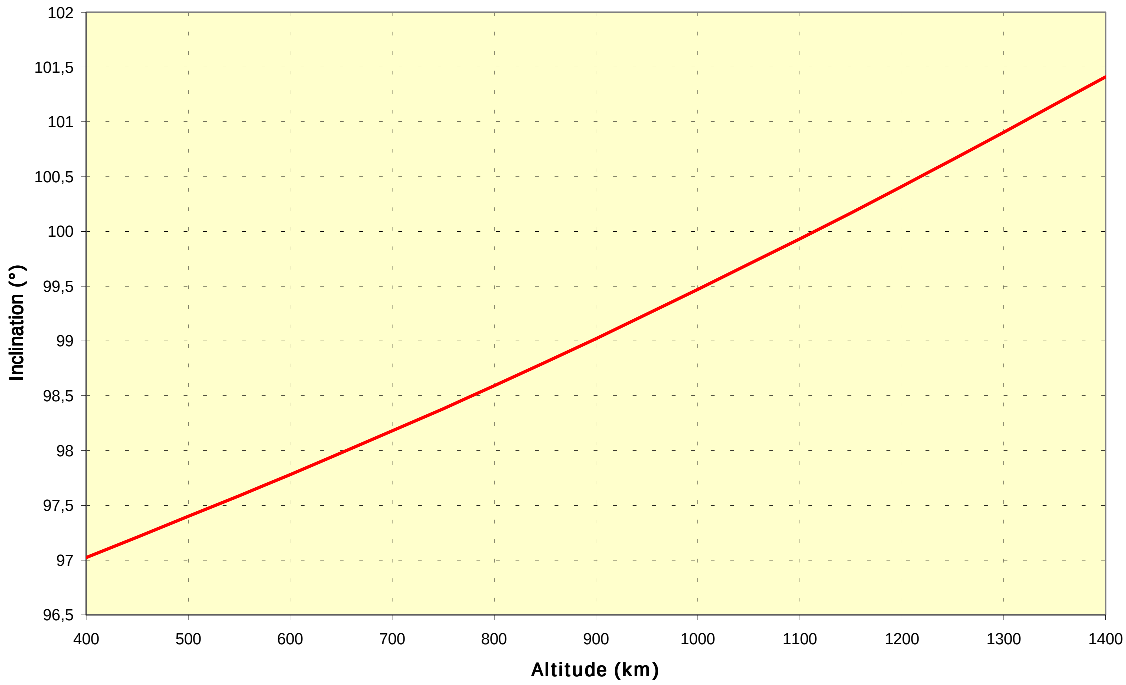


Figure 2.6.a - Altitude and inclination of SSO

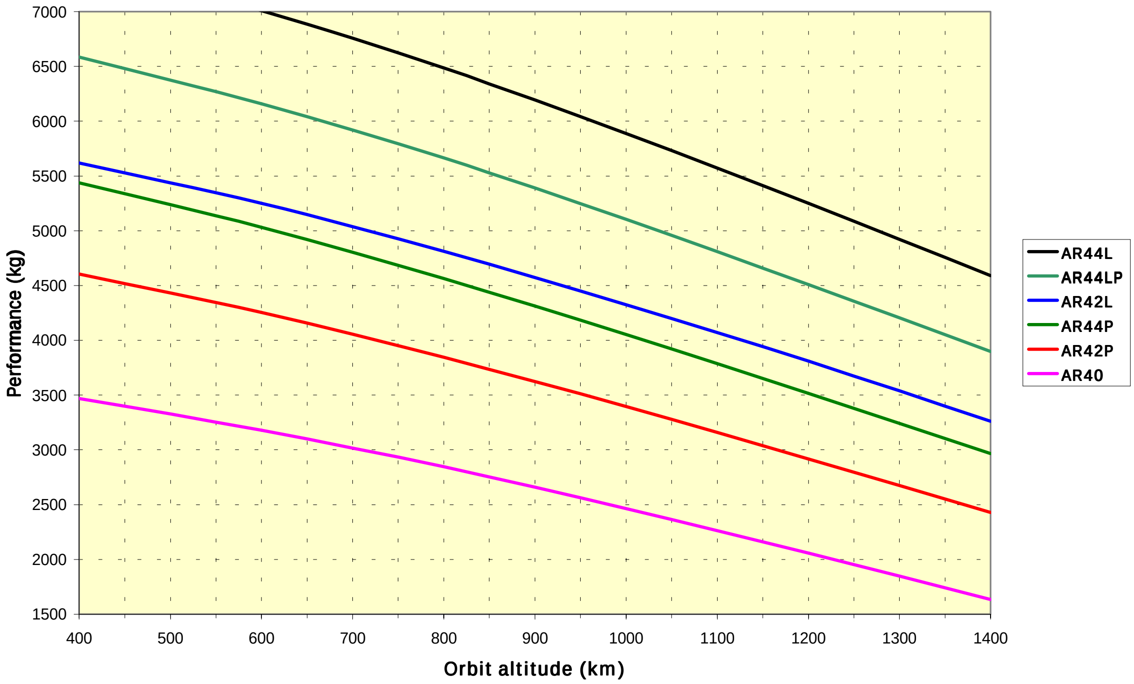


Figure 2.6.b - Performance for SSO

	Mass (kg)					
Altitude (km)	A44L	A44LP	A42L	A44P	A42P	A40
400	7446	6584	5618	5438	4604	3469
600	7009	6159	5250	5031	4253	3178
800	6485	5663	4813	4563	3844	2846
1000	5888	5103	4324	4052	3394	2464
1200	5249	4507	3809	3514	2917	2056
1400	4590	3896	3261	2965	2430	1634

2.7 Low Earth Orbits (LEO)

Among the wide range of orbits that can be served by ARIANE 4, the 60° inclined circular orbits have been chosen to illustrate the ARIANE 4 performance.
For launches on other orbits users are invited to contact Arianespace.

Performance computations are based on the following assumptions :

- Aerothermal flux at fairing jettison less than 500 W/m²
- Launch azimuth of 60°
- Inertial node control on a 20 min launch window

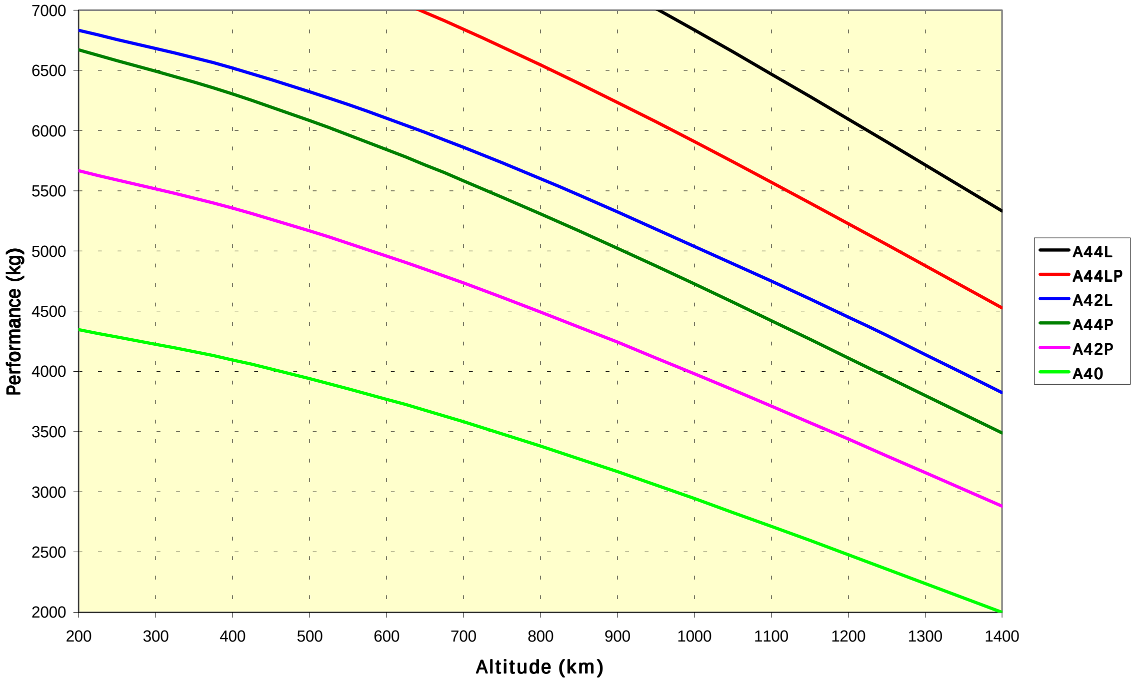


Figure 2.7.a - Perfomance for LEO - Circular orbits - i = 60°

	Mass (kg)					
Altitude (km)	**A40**	**A42P**	**A44P**	**A42L**	**A44LP**	**A44L**
200	4347	5667	6670	6833	7964	8987
400	4096	5355	6304	6519	7597	8622
600	3769	4958	5843	6102	7117	8125
800	3380	4493	5309	5601	6546	7524
1000	2944	3981	4726	5038	5909	6834
1200	2479	3438	4113	4451	5226	6095
1400	1998	2882	3488	3825	4526	5332

2.8 High elliptical Orbits (HEO)

The 65° inclined orbits have been chosen to illustrate the ARIANE 4 performance. For other orbits, customers are invited to contact Arianespace to obtain precise data.

Performance computations are based on the following assumptions :

- Argument of perigee around 52 °
- Ascending node free
- Aerothermal flux at fairing jettisoning less than 1135 W/m²

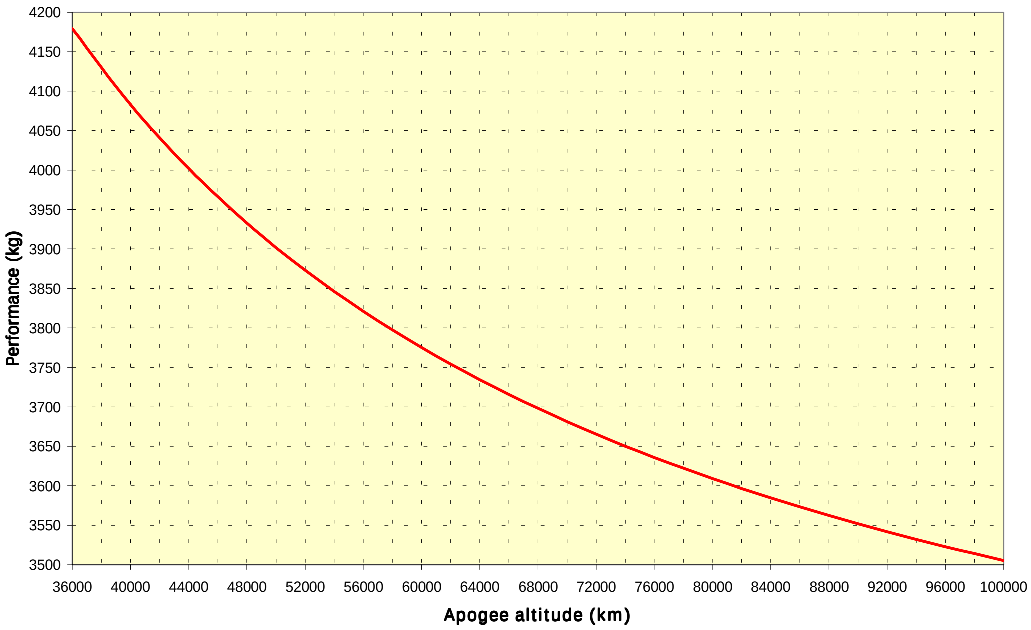


Figure 2.8.a - Ariane 44L - Zp = 200 km - i = 65° - ω free

Altitude (km)	Mass (kg)
36000	4179
40000	4083
44000	4002
48001	3933
52000	3873
56000	3821
60000	3775
64000	3735
68000	3698
72000	3665
76000	3636
80000	3609
84000	3585
88000	3562
92000	3542
96000	3523
100000	3505

2.9 Escape missions

The launch vehicle performance for an ARIANE 44L is presented on figure 2.9.a for energy expressed in terms of excess velocity ranging from 2000 to 5000 m/s and for declination from –5 to 5 °.

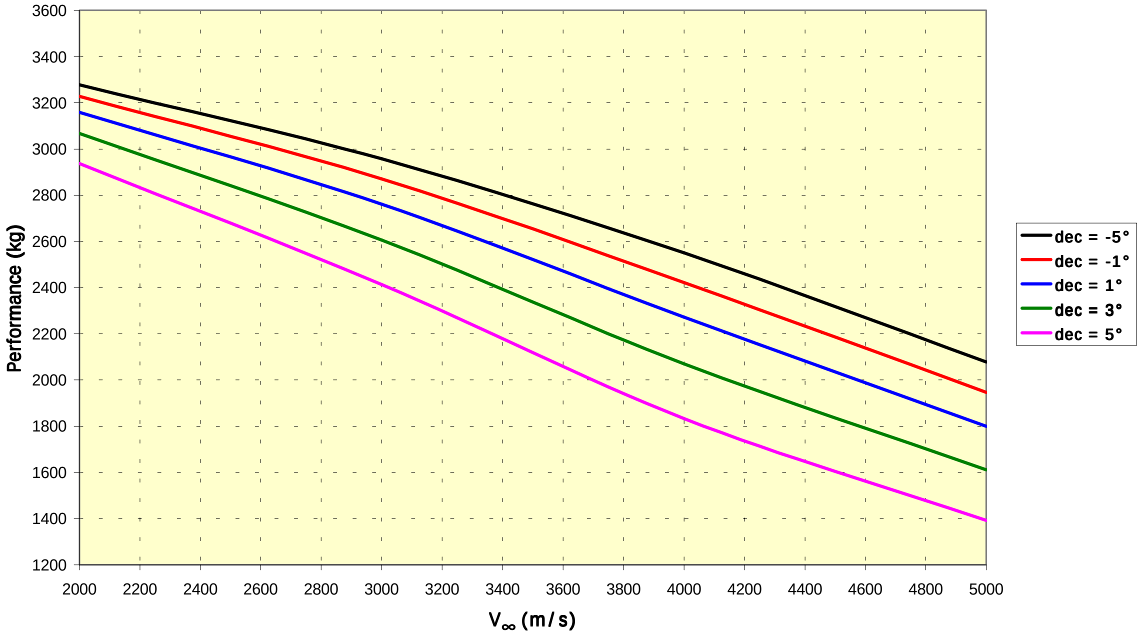


Figure 2.9.a - Ariane 44L - Launch towards East

	Mass (kg)				
V∞ (m/s)	dec = - 5 °	dec = - 1 °	dec = 1 °	dec = 3 °	dec = 5 °
2000	3277	3228	3159	3068	2936
3000	2957	2871	2761	2605	2413
4000	2550	2422	2272	2070	1833
5000	2078	1947	1800	1611	1391

2.10 Spacecraft orientation and separation

2.10.1 General description

After injection into the desired orbit, the launch vehicle Attitude Control System provides the specified orientation and spin (if required) for each spacecraft before its separation. After completion of the separation(s), the third stage is sequenced to carry out a maneuver to avoid subsequent collision.

Typical sequences of events are shown in figures 2.10.1.a (SYLDA), 2.10.1.b (short SPELDA), 2.10.1.c (mini and stretched mini SPELDA).

The time available for the above maneuvers is about 400 seconds.

2.10.2 Orientation performance

The desired orientation at separation should be specified by the User with respect to the following system of axes :

- u : radius vector with its origin at the centre of the Earth, and passing through the intended orbit perigee.
- v : perpendicular to u in the intended orbit plane, having the same direction as the perigee velocity.
- w : perpendicular to u an v, such that u, v, w form a direct trihedron (right handed system).

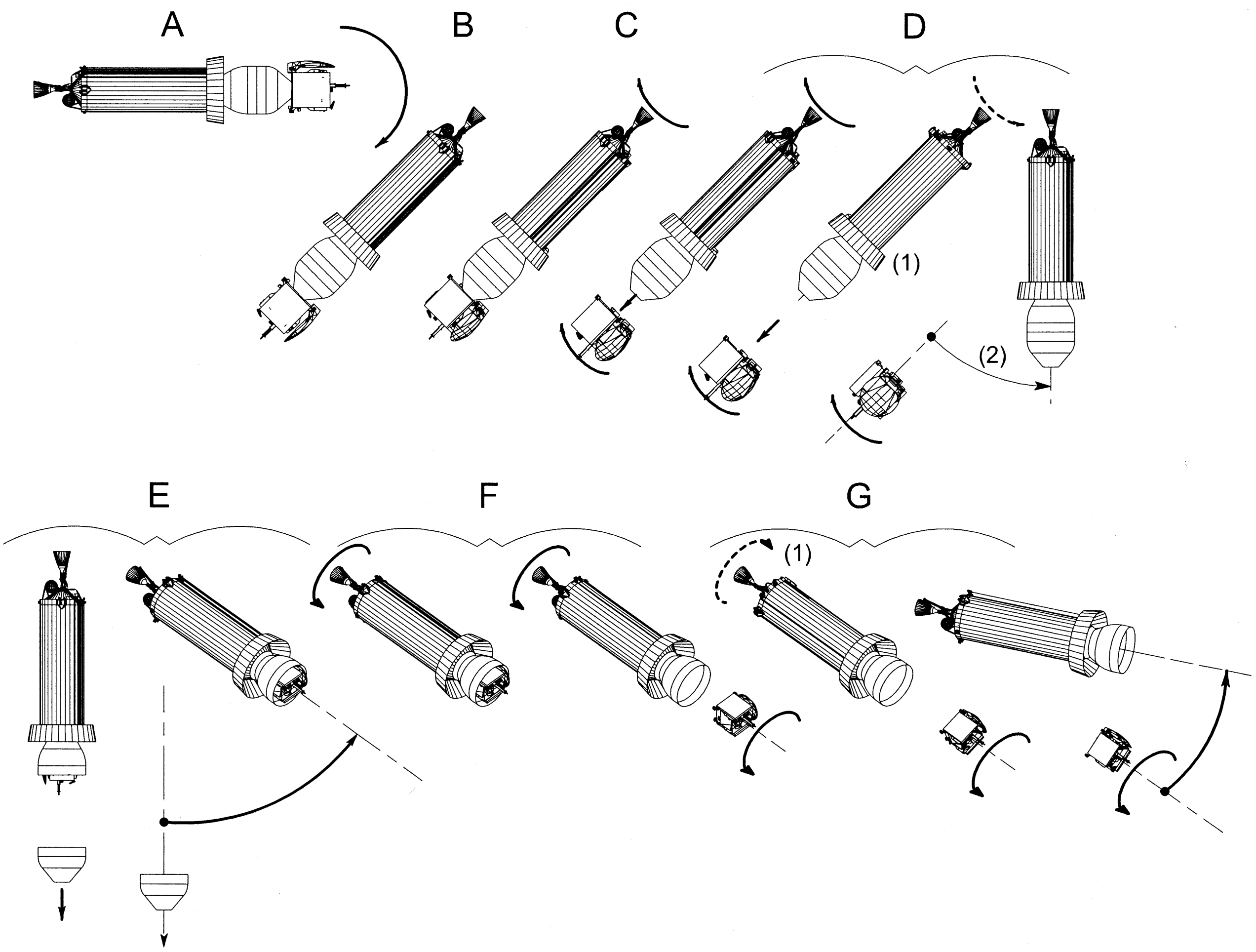
Note : In the case of 3-axis stabilized spacecraft, the desired orientation of the spacecraft longitudinal axis and of one transverse axis should be given.

For specific dual launches, Mission Analysis may lead Arianespace to request a slight adjustment of the desired orientation.

The orientation required by the user can be adjusted to take into account the launch time. In this case, the User must supply details of the orientation required at the start and the end of the intended launch window. This orientation will be intepolated between these two values, for a given launch time.

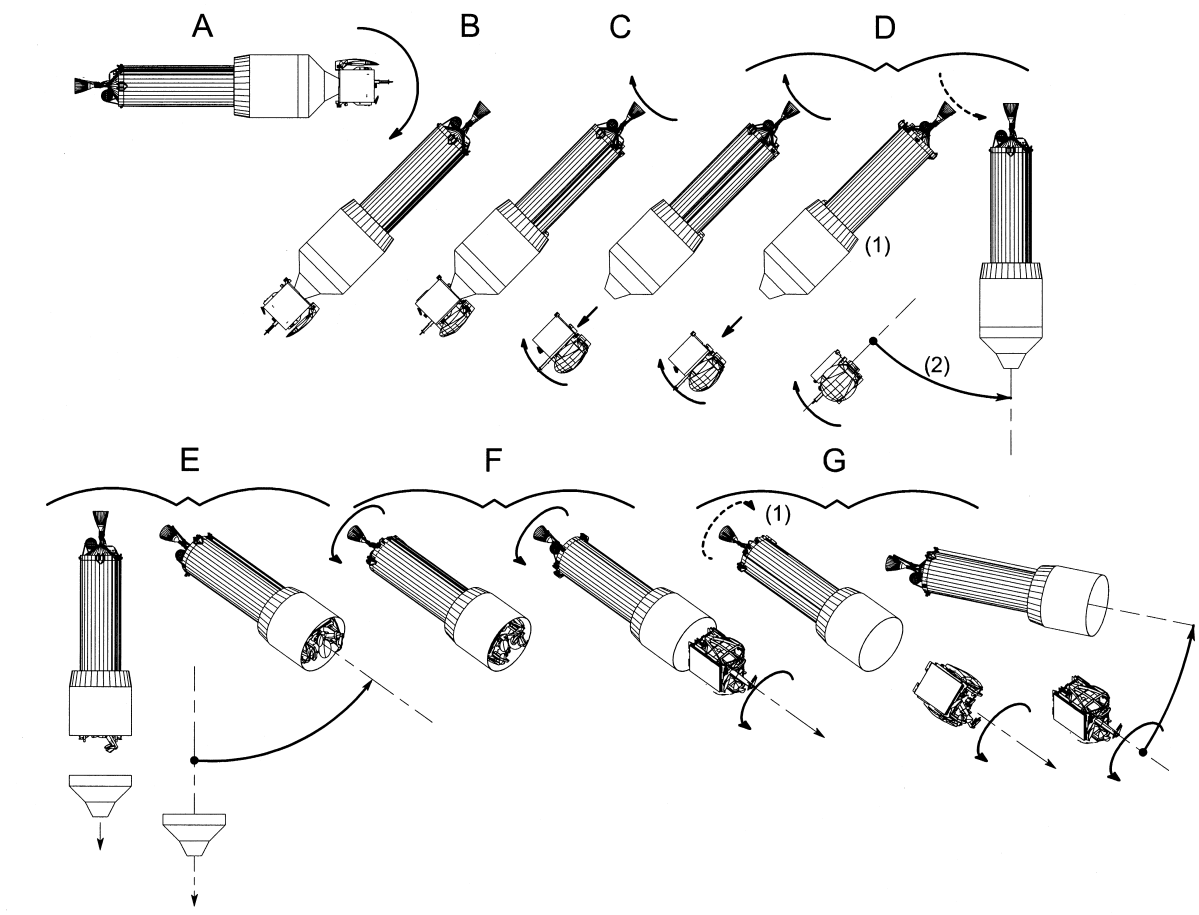
2.10.3 Spin-up performance

The Ariane spin-up system, produces a spin rate less than or equal to 5 rpm, counterclockwise looking aft ; the clockwise spin can be made available if requested. For a single launch, the Preliminary Mission Analysis (see para. 6.4) may show that a highter spin rate could be provided.



A and B	:	Orientation of composite (3^{rd} stage + payload) by 3^{rd} stage roll and attitude control system (SCAR)
C	:	Spin up by action of SCAR
D	:	Separation of upper spacecraft. Then spin down (1) and attitude deviation by action of SCAR (2).
E	:	Upper SYLDA jettisoning. Reorientation as requested by inner spacecraft
F	:	Spin up and separation of inner spacecraft.
G	:	3^{rd} stage avoidance maneuver. (Spin down, attitude deviation, Lox valves opening).
Note	:	Spacecraft separations can also be accomodated under a 3 axis stabilized configuration.

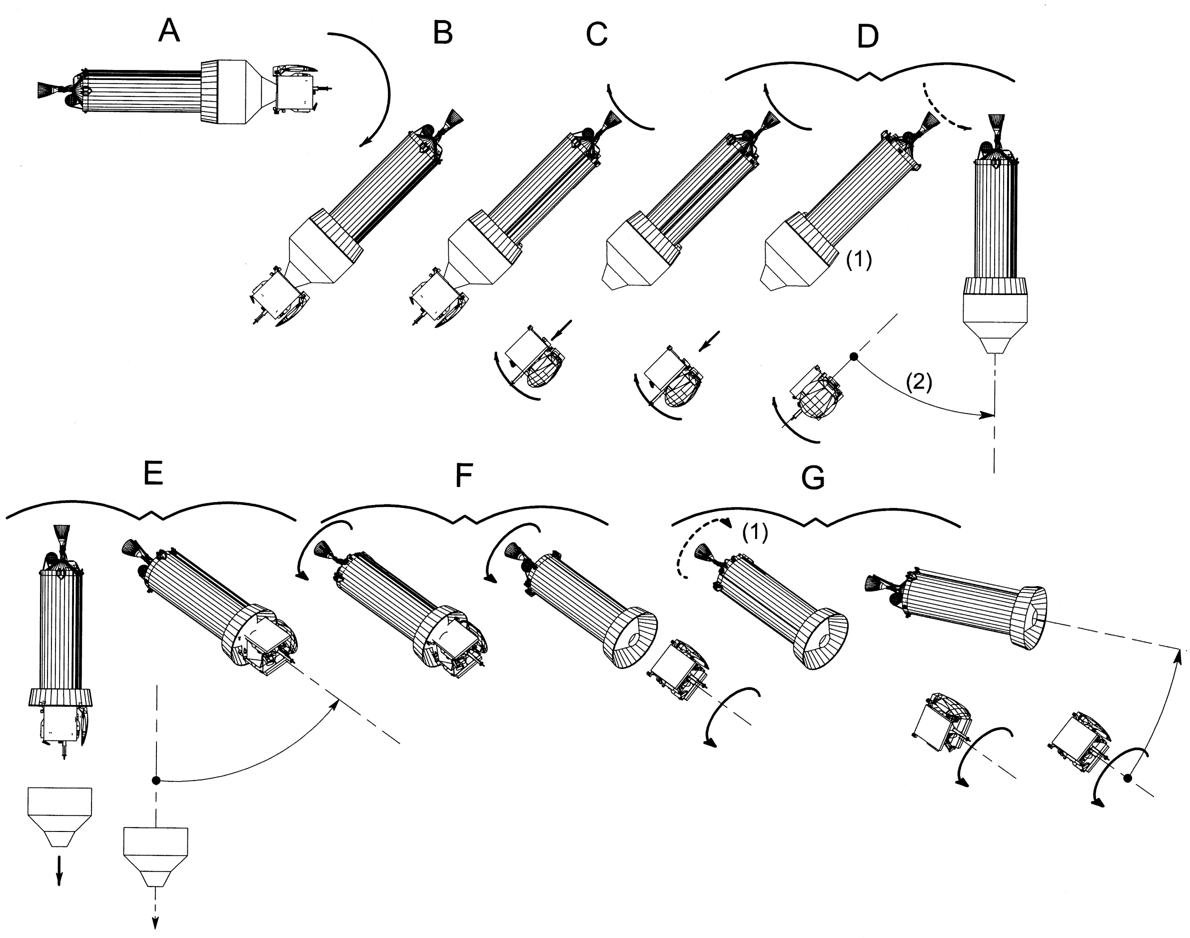
Figure 2.10.1.a : Typical spacecraft/SYLDA separation sequence for various attitude and spin requirements



A and B	:	Orientation of composite (3rd stage + payload) by 3rd stage roll and attitude control system (SCAR)
C	:	Spin up by action of SCAR
D	:	Separation of upper spacecraft. Then spin down (1) and attitude deviation by action of SCAR (2).
E	:	Upper SPELDA jettisoning. Reorientation as requested by inner spacecraft
F	:	Spin up and separation of inner spacecraft.
G	:	3rd stage avoidance maneuver. (Spin down, attitude deviation, Lox valves opening).

Note : Spacecraft separations can also be accomodated under a 3 axis stabilized configuration.

Figure 2.10.1.b : Typical spacecraft/short SPELDA separation sequence for various attitude and spin requirements



A and B	:	Orientation of composite (3rd stage + payload) by 3rd stage roll and attitude control system (SCAR)
C	:	Spin up by action of SCAR
D	:	Separation of upper spacecraft. Then spin down (1) and attitude deviation by action of SCAR (2).
E	:	SPELDA jettisoning. Reorientation as requested by inner spacecraft
F	:	Spin up and separation of inner spacecraft.
G	:	3rd stage avoidance maneuver. (Spin down, attitude deviation, Lox valves opening).
Note	:	Spacecraft separations can also be accomodated under a 3 axis stabilized configuration.

Figure 2.10.1.c : Typical spacecraft/mini and stretched mini SPELDA separation sequence for various attitude and spin requirements

2.10.4 Spacecraft pointing accuracy

The actual values will result from the Preliminary Mission Analysis (see para. 6.4). The following values cover a large range of Ariane 4 compatible spacecraft.

2.10.4.1 Spin-up spacecraft

Values are given along the spin-axis for a 5 rpm spin rate and a probability of 99 %.

- immediately before separation
  transverse angular velocity
  $\omega_t \leq 2°/s$ (in the main inertia axis)
- immediately after separation
  depointing of kinetic momentum
  vector $\delta_H \leq 6°$

2.10.4.2 Three-axis stabilized spacecraft

- immediately before separation
  attitude error along each spacecraft
  geometrical axis : $\Delta_X \leq 3°$
  with probability of 99 %
- immediately after separation, the transverse angular velocity depends mainly upon :
  . Spacecraft static inbalance,
  . Separation system perturbations,
  . Torque induced by the spacecraft sloshing masses.

Note : The values are in accordance with spacecraft balancing characteristics given in para. 4.5.2 and assume the adaptor is supplied by Arianespace.
Possible perturbations induced by spacecraft sloshing masses are not considered in the above figures.

2.10.5 Separation velocities

Each separation system is designed to deliver a minimum relative velocity of 0.5 m/s between the two separated bodies.

For each mission, Arianespace will verify that distances between orbiting bodies are satisfactory until the first spacecraft motor firing.

For this analysis, the User has to provide Arianespace with its orbit and attitude maneuver flight plan.