Solutions and explanation to the <u>aircraft related</u> part of the June 2010 exam

The correct answer(s) is (are) indicated with **red bold fonts.** [Some extra explanations are added in between squared brackets and using blue fonts].

Grading notes:

- Question 2 was excluded from the grading process, because the multiple choices in the Dutch version were incorrectly formulated.
- The text of Question 9 was formulated as multiple choice (i.e., one single possible answer), but, actually, it was a multiple answer (with two possible answers). In this case, full points have been granted to those who indicate at least one of the two correct answers.

1. Which of the following statements are correct (more correct statements are possible)?
   a. The climb gradient affects the maximum altitude that can be achieved in the shortest time [this is true for the climb rate]
   b. Certification regulations generally stipulate limits to the maximum value of climbing gradient [certification generally stipulates limit to minimum climbing gradient]
   c. Certification regulations generally stipulate limits to the minimum value of climbing gradient
   d. Higher aspect ratio values are beneficial to lower the thrust over weight ratio required to achieve a certain climb gradient. [the opposite is generally true, see example of F104 interceptor on slides]
   e. Interceptor aircraft are characterized by very high climb rate performance

2. Which one of the following statement is correct? When an aircraft deploys flaps at landing…..
   a. The friction drag decreases but the span efficiency factor (i.e. the Oswald factor) increases [the opposite of these two statements is true]
   b. The friction drag decreases, such that the aircraft can better climb
   c. The aircraft stability changes, whereas drag is not affected [drag is definitely affected]
   d. **Friction drag increases and Oswald factor decreases** [the Oswald factor decreases, because the lift distribution is drastically modified by the deflected flaps (it becomes quite different than the optimal elliptical or quasi elliptical distribution)]
   e. Both statement a. and b. are true

3. Which one of the following statements is correct?
    a. A very high stall speed is beneficial in view of reaching higher cruise speed [stall speed and cruise speed are not directly related. High cruise speed can be reached independently of the stall speed. Low stall speed can be achieved using high lift devices, without affecting the cruise speed]
    b. A low stall speed allows lowering the approach speed
    c. The stall speed depends on the weight of the aircraft but not on the size of the wing [stall speed depends both on the wing size and the aircraft weight (which must be equal to the lift) See mathematical definition of stall speed]
    d. The stall speed depends on the size of the wing but not on the weight of the aircraft
    e. The stall speed at sea level is higher than at high altitude [the opposite is true because of the higher air density at sea level than at high altitude]

4. Chose the right statement to complete the sentence below:
   In the typical organization of the aircraft development process in industry, the *baseline design* is...
    a. The main input to enter the detail design phase of an aircraft development program
    b. The main input to the preliminary design phase [see slides on the various phases of the design process]
    c. The result of the base drag computation process [nothing to do with drag]
    d. The configuration of a reference aircraft selected for the development of the new aircraft [nothing to do with reference aircraft]
    e. None of the statements above

5. Which one of the statements below is correct?
    a. The longer the fuselage length, the higher the pressure drag contribution [the drag of the fuselage is due to two main contributions: pressure drag and friction drag. The pressure drag is the dominant drag contribution for bluff bodies (short and thick); the friction drag contribution is dominant for slender bodies (with a very large wet surface with respect to the enclosed volume). Hence the contrary is true]
    b. The shorter the fuselage length, the higher the friction drag [the contrary is true. See above]
    c. The higher the slenderness ratio of the fuselage (L/d, where L is the total fuselage length and d is the average cross section diameter), the higher the ratio between pressure drag and friction drag [The opposite is true, because the pressure drag contribution gets lower with slender bodies, hence the ratio of pressure and friction drag decreases]
    d. The higher the slenderness ratio of the fuselage (L/d, where L is the total fuselage length and d is the average cross section diameter), the higher the ratio between friction drag and pressure drag.
    e. None of the statements above is correct

6. Why can an auxiliary power unit (APU) be a critical safety device on a twin-engine, extended-range aircraft?
    a. Two engines cannot generate enough power to operate all electric systems on board the aircraft
    b. The APU provides critical power for the emergency systems
    c. The APU provides additional thrust in case of a one-engine-inoperative (OEI) condition
    d. **The APU provides energy for all the systems in case of a one-engine-inoperative condition (OEI)** [see lecture on aircraft system]
    e. None of the above

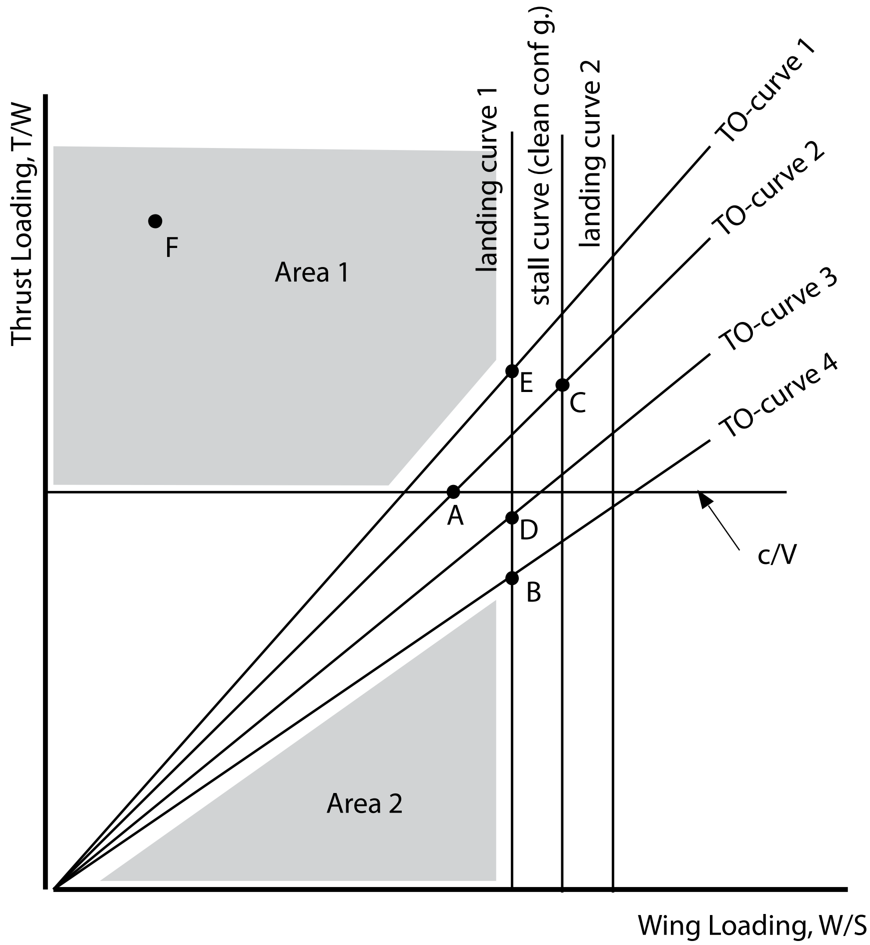
7. Which of the statements below are correct (more correct statements are possible. You can make use of the table 1 from CS.25)?
    a. Two Type I and two Type III exits (per side) should be accounted for during the development of a 170 passengers aircraft (pax distributed in 2 classes) [this amount of emergency exits would turn out to be insufficient in the moment that the same aircraft must be used in a single, high density configuration (hence with many more passengers than 170)]
    b. **More than 2 Type I and Two Type III exits (per side) should be accounted for during the development of a 170 passengers aircraft (pax distributed in 2 classes)**
    c. Less than 2 Type I and Two Type III exits (per side) can be accounted during the development of a 170 passengers aircraft (pax distributed in 2 classes), if it can be demonstrated that, in case of emergency, all passengers can be evacuated within 90 seconds. [Regulations do not allow discounts! A lower amount of exits than stipulated by the CSs would not even allow a company to proceed with any demonstration phase]
    d. The requirements concerning the minimum size for ejection seats for small military trainers can be found in CS23. For larger military trainers (MTOW>19000lb), they can be found in CS25. [The EASA, CSs do not apply for military aircraft, but for civil aviation]

e. **The amount and type of emergency exits can influence the number of flight attendants required to operate the aircraft** [indeed the type of exits, hence their size and passengers exit rate, affects the amount of flight attendants that must be in charge of each door. See slide 39 on the first lecture on fuselage]

No. Pass.	Type I	Type II	Type III	Type IV
1–9				1
10–19			1	
20–39		1	1	
40–79	1		1	
80–109	1		2	
110–139	2		1	
140–179	2		2	
180–299	Add exits so that 179 plus "seat credits" ≥ passenger number.			

	Seat Credit	Exit Type
	12	Single Ventral
	15	Single Tailcone
	35	Pair Type III
	40	Pair Type II
	45	Pair Type I
	110	Pair Type A
≥ 300	Use pairs of Type A or Type I with the sum of "seat credits" ≥ passenger number.	

8. Looking at the W/S-T/W plot below, indicate which of the following statements are correct (more correct statements possible)
   a. A design point could be picked both in *Area 1* and *Area 2*, but design points in *Area 1* would demand a too high thrust loading [no design points allowed in area 2, where the requirements on take off distance and climb gradient are not satisfied]
   b. A design point could be picked only in *Area 2*, elsewhere the take off constraint would be violated [all points in area 2 violates the take off requirements].
   c. A design point could only be picked in *Area 1* in order to comply with the takeoff, landing and wing loading requirements [there is no such "wing loading" requirement! Wing loading value can be selected in order to fulfill certain requirements]
   d. **In point D the climb gradient requirement is violated** [indeed point D has a too low thrust loading]
   e. **In point E, the take off constraint is satisfied for a lower CLmax value than Point A** [indeed the 4 take off curves are plotted for increasing values of maximum lift coefficient, being TO-curve 1 the one with lowest CLmax and TO-curve 4 the one with the highest CLmax]



9. Looking at the same W/S-T/W plot as for the previous question, indicate which one of the statements below is wrong.
    a. Point E is an allowed design point [correct: it fulfils the requirements on take off, climb gradient and the most restricting one on landing (i.e., landing curve 1)]
    b. Point A can be an allowed design point only for a higher $CLmax_{take\text{-}off}$ value than that used to build *TO-curve 1* [This is correct: assuming the use of more complex high lift devices, hence a higher CLmax value than used to plot TO-curve1, Area 1 would expand and include point A as possible design point]
    c. **Point C violates the landing constraint indicated by *Landing-Curve 2*** [wrong! point C does not violate the landing constraint indicated by *Landing-curve 2*: the wing loading is lower than the allowed limit. Point C would violate the constraint indicated by *landing curve 1*, being the wing loading too high in that case]
    d. **Allowing a lower stall speed, point C would shift towards higher wing-loading values** [lowering the allowed stall speed would demand a larger wing surface for a given aircraft weight, hence a lower wing loading. Point C would then shift to the left]
    e. Point F is a possible design point

10. A propeller driven airplane must have a power off stall speed of no more than 50 knots at sea level (air density 1,23 Kg/m^3) with full flaps down (CLmax = 2 at landing). With flaps up (CLmax=1.6) the stall speed is to be less than 60 Knots (1 Knot = 0.514444m/s).
    In order to meet both requirements at take-off gross weight, it is necessary that …
    a. W/S < 95.6 Kg/m^2 [by using the formula for stall speed calculation with the two values of CLmax and speed, it can be easily derived that for W/S <95,6 it is possible to satisfy only the stall speed requirement in clean wing. The flapped stall speed requirement would be violated]
    b. **W/S < 813.81 N/m^2**
    c. 82.98Kg/m^2 < W/S < 937.50N/m^2 [in order to satisfy both requirements, the maximum allowed wing loading must be the lowest of the two values and not one value in between]
    d. W/S > 937.50N/m^2 [in this case both the stall speed requirements for clean and flapped wing would be violated]
    e. CS23 does not allow measuring the stall speed in power off condition [the contrary is true. All stall speed measurements should be based on power off condition. The operating engine might cause an increase in the lift, hence induce not conservative measurement of the stall speed]

You are a conceptual aircraft designer at AeroIndustries. Together with your team, you are currently working on the development of a new generation military jet transporter. The following mission has been specified to drive the design the aircraft:

1 Engine start and warm up
2 Taxi
3 Take off at sea level (air density 1,23 Kg/m^3)
4 Climb to cruise altitude
5 Cruise for 5000Km at service ceiling, with a speed of 920Km/h (L/D = 15)
6 Descend at loiter altitude on destination airport
7 Loiter for 20 minutes (L/D = 18)
8 Descend
9 Land, taxi and shut down engines

The aircraft will be operated by a typical crew of 8, i.e., 3 pilots, 2 flight engineer and 3 loadmasters (assume 100Kg including baggage per crew member), and will be able to fly the mission specified above carrying a payload of 122500Kg.

Your first task is to compute the **maximum take off weight of the aircraft, the aircraft empty weight and the fuel weight.**

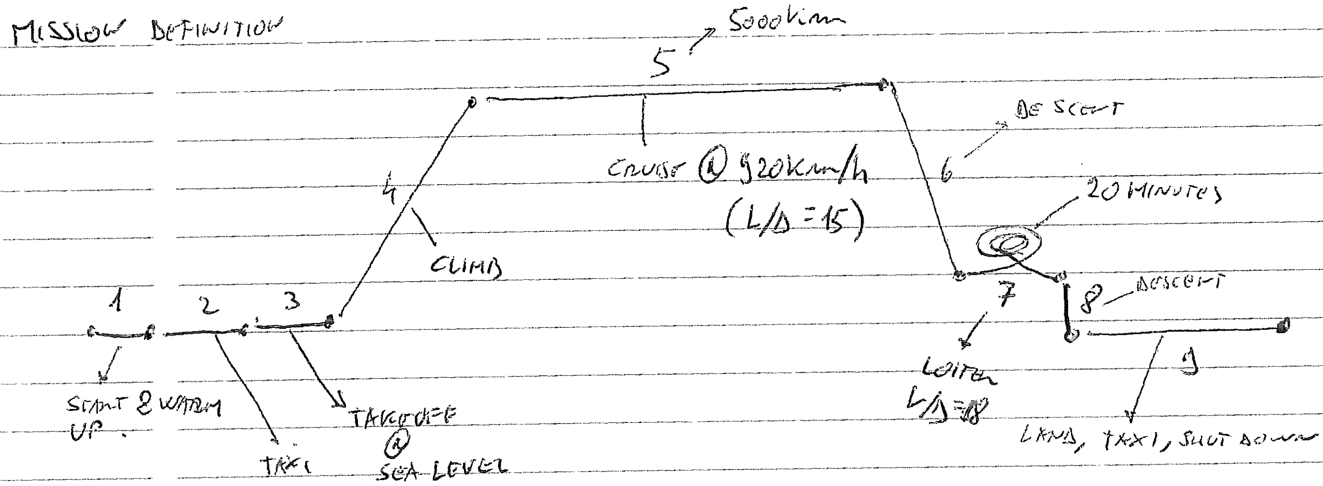
- Based on statistics you can assume the empty weight/takeoff weight ratio equal to 0,45.
- For this preliminary calculation you can ignore the weight of oil and trapped fuel.
- There is no need to include any reserve fuel for this type of mission.

- You can use the fuel fractions values reported in the *data and formula sheet* for your aircraft category.
- The engine specific fuel consumption can be assumed equal to c_J= 0,000019 Kg/Ns, constant and independent of the altitude.
- The air density at sea level can be assumed equal to 1,23 Kg/m^3

PRELIMINARY WEIGHT ESTIMATION FOR A

MILITARY JET TRANSPORTER

MISSION DEFINITION



($W_{fuel,res} = 0$)

- CREW = 8 x 100 kg = 800 kg
- NO RESERVE, NO TRAPPED OIL/FUEL ($W_{TFO} = 0$)
- PAYLOAD = 122500 Kg
- W_E/MTOW = 0,45

$$MTOW = \underbrace{W_{EM} + W_{crew}}_{W_{OEW}} + W_{payload} + W_{fuel} + \cancel{W_{TFO}}$$

$$MOTW = 0{,}45\ MTOW + 800\,kg + 122500\,kg + \underbrace{W_{fuel,used} + \cancel{W_{fuel,reserve}}}$$

$$MTOW = 0{,}45\ MTOW + 123300\,kg + (1 - M_{ff}) \cdot MTOW$$

$$M_{ff} = \frac{W_1}{W_{TO}} \prod \frac{W_{i+1}}{W_i} = \frac{W_1}{W_{TO}} \cdot \frac{W_2}{W_1} \cdot \frac{W_3}{W_2} \cdot \frac{W_4}{W_3} \cdot \frac{W_5}{W_4} \cdot \frac{W_6}{W_5} \cdot \frac{W_7}{W_6} \cdot \frac{W_8}{W_7} \cdot \frac{W_{final}}{W_8}$$

FROM STATISTICS: 0,990 0,990 0,995 0,980 | 0,990 | 0,990 0,992

W_5/W_4 → BREGUET CRUISE

W_7/W_6 → BREGUET ENDURANCE

(*) IN THIS CASE 2 DESCENT PHASES HAVE BEEN CONSIDERED:
IT'S OK IF JUST ONE DESCENT PHASE IS ACCOUNTED

kladpapier/rough-work paper

ESTIMATION OF CRUISE FUEL FRACTION $\frac{W_5}{W_4}$

THE ~~...~~ RANGE BREGUET EQUATION FOR JET AIRCRAFT CAN BE USED

$$R = \frac{V}{g\,C_J} \frac{L}{D} \ln\left(\frac{W_4}{W_5}\right)$$

WHERE:

$R = 5000\,\text{Km} \cdot 1000 = 5000000\ m$ RANGE IN CRUISE

$V = \frac{920\ \text{Km/h}}{3{,}6} = 255{,}55\ m/s$ CRUISE SPEED

$\left(\frac{L}{D}\right)_{cruise} = 15$ AERODYNAMIC EFFICIENCY IN CRUISE CONDITION

$C_{JET} = 0{,}000019\ Kg/Ns$ SPECIFIC FUEL CONSUMPTION OF SELECTED JET ENGINES

$$\ln\left(\frac{W_4}{W_5}\right) = \frac{R \cdot g \cdot C_J}{V \cdot L/D} = \frac{5\cdot10^6 \cdot 9{,}80665 \cdot 1{,}9\cdot10^{-5}}{255{,}55 \cdot 15} \frac{[m]\,[m/s^2][Kg/Ns]}{[m/s]}$$

$$\text{"} \quad \text{"} = 0{,}243034$$

$$\left(\frac{W_4}{W_5}\right) = e^{0{,}243034} = 1{,}275112$$

$$\boxed{\frac{W_5}{W_4} = 0{,}784245} \simeq 0{,}784$$

kladpapier/rough-work paper

ESTIMATION OF LOITER FUEL FRACTION $\left(\frac{W_7}{W_6}\right)$

THE ENDURANCE BREGUET EQUATION FOR JET A/C CAN BE USED

$$E_{JET} = \frac{1}{g \cdot C_J} \frac{L}{D} \ln\left(\frac{W_6}{W_7}\right)$$

WHERE
$$\begin{cases} E = 20_{m} \cdot 60 = 1200 \text{ s} & \text{(LOITERING TIME)} \\ C_J = 1{,}9 \cdot 10^{-5} \text{ Kg/Ns} & \text{(ASSUMED THE SAME AS IN CRUISE)} \\ \left(\frac{L}{D}\right)_{LOIT} = 18 & \text{(HIGHER THAN IN CRUISE)} \end{cases}$$

$$\ln \frac{W_6}{W_7} = \frac{E_J \cdot g \cdot C_J}{(L/D)_{LOIT}} = \frac{20 \cdot 60 \cdot 9{,}80665 \cdot 1{,}9 \cdot 10^{-5}}{18} \quad \cancel{s} \cdot \frac{m}{s^2} \cdot \frac{kg}{N\cancel{s}}$$

$$\ln \frac{W_6}{W_7} = 0{,}012422$$

$$\frac{W_6}{W_7} = e^{0{,}0124224} = 1{,}012499$$

$$\boxed{\frac{W_7}{W_6} = 0{,}987655 \simeq 0{,}988}$$

$$1 - M_{FF} = 0{,}280262$$

$$= 0{,}272992 \quad \text{(IF ONLY ONE DESCENT ACCOUNTED)}$$

~~OPTIMISTIC~~

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$\dot{M}TOW = 0{,}45\ MTOW + 123\,300 + 0{,}280262\ MTOW$

$(1 - 0{,}280262 - 0{,}45)\ MTOW = 123300$

- $MTOW = \frac{123\,300}{0{,}269738} = 457109{,}7\ Kg$

  (445112,9 Kg for 1 descent)

- $M_{EW} = 0{,}45\ MTOW = 205699{,}4\ Kg$

  (200300,8 Kg for 1 descent)

- $M_{FUEL} = 128110{,}3\ Kg$

  (121512,1 Kg for 1 descent)

Answers June 21, 2010 exam, problems 11-20

11c: The acronym SMART stands for specific, measurable, achievable, realistic, time-bound.

12d
Δv = 450m/s = 2000m/s x ln($M_{o,1}/M_e$). Since M_e is given, it follows for $M_{o,1}$ = 1.3 x M_e = 626.2kg
Δv = 1000m/s = 3000m/s * ln($M_{o,2}/M_{o,1}$). Using the above calculated value for $M_{o,1}$, we find $M_{o,2}$ = 1.4 x $M_{o,1}$ = 873.9kg.

13c: Nadir direction is defined as pointing down to Earth

14d: Angular acceleration follows by dividing the worst case disturbance torque (0.05 Nm) by the MMOI. This gives an angular acceleration of 5 x 10^{-5} rad/s^2. For a constant acceleration we find for the angle over which the vehicle is rotated in 5 minutes time:

$$0.5 \times 5 \times 10^{-5} (5\text{min} \times 60\text{s})^2 = 4.5 \text{ rad} = 128.9 \text{ deg}$$

15e: Maximum orbital period is 90 minutes. During these 90 minutes, we have a data rate of 749 Mbps. This gives a total amount of data of 749 Mbps x 90min x 60s = 4.04E12 bits or 505 GByte.

16e: Given the solar intensity and the conversion efficiency, we have 1400 W/m^2 x 0.12 = 168 W/m^2 available. Correcting for the inherent degradation (distance between cells) and the life degradation, we find as a result 168 x (0.80 x 0.85) = 114.2 W/m^2. We need to produce 1000 W, hence we need a surface area of 1000/114.2 = 8.75 m^2.

17b: For area moment of inertia of thin-walled cylinder it follows a value of 2.51E-5 m^4. It follows for the Euler buckling load using relation [39] (reader) a value of π^2 x 70E9 x 2.51E-5/(4 x 0.5^2) = 17.6 MN.

18c

19b: We need to find a balance between the heat absorbed by the satellite and the heat radiated away. It follows:

$$J_s \times \alpha \times A = \varepsilon \times \sigma \times T^4 \times 6A$$

Here we have taken the area that absorbs equal to "A" and the area that emits equal to "6A". We only need to determine the solar flux at 1.5 AU. This is equal to the solar constant, divided by the distance in AU squared. This gives 622 W/m^2. So we now have:

$$622.2 \text{ w/m}^2 \times 0.15 \times A = 0.9 \times 5.67\text{E-8} \times T^4 \times 6A$$

After dividing both sides by A, we are left with 1 equation with 1 unknown (temperature T). Solving for the unknown gives: T = 132.1 K

20a: First use rocket equation to calculate propellant mass and equation for rocket thrust to calculate propellant mass flow rate. Using the rocket equation and using the given values for ΔV and effective exhaust velocity gives a vehicle mass ratio of 6.52. Given the initial mass of 100 ton, we find an empty mass of 15.34 ton and hence a propellant mass of 84.66 ton. Given the maximum allowed acceleration of 7g, we find a maximum acceleration of 68.7 m/s^2. For constant thrust we find acceleration is maximum at end of burn. This then defines a thrust level of 68.7 m/s^2 x 15.34 ton = 1.025 MN. Dividing the thrust by the effective exhaust velocity gives a mass flow rate of 329 kg/s. Dividing the propellant mass of 84.66 ton by the propellant mass flow rate of 329 kg/s gives the required time: t = 257.3 s.

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		studienummer
		vak / code
		datum / fac.

Gebruik voor elke opgave een afzonderlijk vel papier !

Open problem 2 : Space shuttle id vel. change

1) Effective exhaust velocity $(V_e)_{eff} = I_{sp} \cdot g_0$

SSME

$= 455\,(9{,}81)$

$$\boxed{(V_e)_{eff} = 4464 \text{ m/s}}$$

Massflow rate $\dot{m} = \dfrac{F}{(V_e)_{eff}} = \dfrac{1.752 \times 10^6 \text{ N}}{4464}$

$$\boxed{\dot{m} = 392.5 \text{ kg/s}}$$

2) Effective exhaust velocity SRB

$M_F = 590.000 - 63272 \text{ kg} = 526728 \text{ kg}$

$\dot{m} = \dfrac{M_F}{t_b} = \dfrac{526728 \text{ kg}}{124 \text{ s}} = 4248 \text{ kg/s}$

$$\boxed{(V_e)_{eff} = \frac{F}{\dot{m}} = \frac{12.5 \times 10^6 \text{ N}}{4248 \text{ kg/s}} = 2943 \text{ m/s}}$$

3) First sub-rocket

Initial mass $(M_o)_1 = 110 \text{ ton} + 756 \text{ ton} + 2(590) \text{ ton}$

$$\boxed{(M_o)_1 = 2046 \text{ ton}}$$

Empty mass $(M_e)_1 = (M_o)_1 - (2\dot{m}_{SRB} \times 124 s) - (3\dot{m}_{ssme} \cdot 124 s)$

$= 2046 - (2 \times 527) - (146)$

$$\boxed{(M_e)_1 = 846 \text{ ton}}$$

4/ Average exhaust velocity 1st sub-rocket

$$\frac{N_{SSME} \cdot F_{SSME} \cdot t_b + N_{SRB} \cdot F_{SRB} \cdot t_b}{M_F}$$

$$\overline{(V_e)_{eff}} = \frac{3 \times 1.752 \times 10^6 \cdot 124 + 2(12.5E6) \cdot 124}{(2046 - 846)\ ton}$$

$$\boxed{\overline{(V_e)_{eff}} = 3126.5\ m/s}$$

5/ $(M_o)_2 = (M_e)_1 - 2(M_e)_{SRB}$

$$\boxed{(M_o)_2 = 846 - 2(63.2) = 719.6\ ton}$$

$(M_e)_2 = (M_e)_{orbiter} + (M_e)_{ET}$

$$\boxed{(M_e)_2 = 110 + 26.5 = 136.5\ ton}$$

6/ $(\Delta V)_1 = \overline{(V_e)_{eff}}\ \ln \frac{(M_o)_1}{(M_e)_1} = 3126.5\ m/s \cdot \ln \frac{2046}{846} = 2.76\ \frac{km}{s}$

$(\Delta V)_2 = (V_e)_{eff\ SSME} \cdot \ln \frac{(M_o)_2}{(M_e)_2} = 4464\ m/s\ \ln \frac{719.6}{136.5} = 7.42\ \frac{km}{s}$

$10.2\ \frac{km}{s}$