Name: **Enrolment No:**	UPES UNIVERSITY OF TOMORROW

UPES

End Semester Examination, May, 2025

Course: Aerodynamics – I **Semester: IV**
Program: B.Tech ASE **Time 03 hrs.**
Course Code: ASEG 2020 **Max. Marks: 100**

SECTION A

S. No.		Marks	CO
Q1.	Define streamlines, pathlines, and streaklines.	**4**	**CO1**
Q2.	Provide a reasoned explanation for the generation of form drag on an aircraft	**4**	**CO2**
Q3.	Explain the role of conformal transformation in the aerodynamic analysis of airfoils.	**4**	**CO3**
Q4.	Discuss the aerodynamic advantages provided by multi-element airfoils.	**4**	**CO4**
Q5.	Interpret the influence of Aspect Ratio of wing on its aerodynamic efficiency.	**4**	**CO5**

SECTION B

Q6.	Derive the momentum equation in its integral conservative form. **OR** Discuss in detail the differences between the Eulerian and Lagrangian approaches for describing fluid motion. Illustrate your explanation with a suitable example.	**10**	**CO2**
Q7	Define a source flow and derive the expression for its stream function and velocity potential function.	**10**	**CO3**
Q8	Consider a vortex filament of strength Γ in the shape of a closed circular loop of radius R. Obtain an expression for the velocity induced at the center of the loop in terms of Γ and R.	**10**	**CO4**
Q9	Derive the fundamental equation of Prandtl's lifting line theory for a finite wing.	**10**	**CO5**

SECTION-C			
Q 10	Superimpose uniform flow with a vortex sheet placed along the camber line of an airfoil. Derive an expression to estimate the circulation of the vortex sheet. Apply this expression to a symmetric airfoil to obtain: i. Lift coefficient as a function of angle of attack ii. Location of the aerodynamic center **OR** The NACA 4412 airfoil has a mean camber line given by $\frac{z}{c} = 0.25\left[0.8\left(\frac{x}{c}\right) - \left(\frac{x}{c}\right)^2\right]$ for $0 \le \frac{x}{c} \le 0.4$ $\frac{z}{c} = 0.111\left[0.2 + 0.8\left(\frac{x}{c}\right) - \left(\frac{x}{c}\right)^2\right]$ for $0.4 \le \frac{x}{c} \le 1$ Using thin airfoil theory calculate: i. Zero lift angle of attack ii. C_L when angle of attack is 3^0 iii. C_m at quarter chord when angle of attack is 3^0 iv. "x_{cp}/c" when angle of attack is 3^0	**20**	**CO4**
Q 11	An aircraft has a wing area of 15.8 m^2 and a wingspan of 9.75 m. Its maximum gross weight is 10898 N. The wing uses an NACA 65-415 airfoil, which has a lift slope of 0.1033 /degree and zero lift angle of attack of -3^0. Assume $\tau = 0.12$. If the airplane is cruising at 53.64 m/s at standard sea level condition at its maximum gross weight and is in straight flight at constant altitude. Calculate the geometric angle of attack of the wing.	**20**	**CO5**